DEPARTMENT OF DEFENSE
OFFICE OF FREEDOM OF INFORMATION
1155 DEFENSE PENTAGON
WASHINGTON, DC 20301-1155

January 24, 2014
Ref: 14-F-0337

Mr. John Greenewald, Jr.
[redacted]

Dear Mr. Greenewald,

This responds to your attached Freedom of Information Act (FOIA) request, for a copy of *Ten Steps Into Space*, which you submitted to the Defense Technical Information Center (DTIC) on December 14, 2013. DTIC subsequently referred your request, along with the responsive document, to this office for processing and direct response to you. We assigned your request FOIA case number 14-F-0337.

As this document is also publicly available online, we are releasing the document provided by DTIC to you in full. It is enclosed. There is no fee for this information in this instance.

This action closes your request with this office.

Sincerely,

Susan Aldorfer
for Paul J. Jacobsmeyer
Chief

Attachments:
As stated

Hamilton, Mike CIV DTIC R 14-F-0337

From: John Greenewald, Jr. <john@greenewald.com>
Sent: Saturday, December 14, 2013 12:18 PM
To: FOIA
Subject: FOIA REQUEST

To whom it may concern,

This is a non-commercial request made under the provisions of the Freedom of Information Act 5 U.S.C. S 552. My FOIA requester status as a "representative of the news media." I am a freelance television producer often working on documentaries related to my FOIA requests, my work is commonly featured throughout major news organizations, and I freelance writer for news sites as well. Examples can be given, if needed.

I prefer electronic delivery of the requested material either via email to john@greenewald.com or via CD-ROM or DVD via postal mail. Please contact me should this FOIA request should incur a charge.

I respectfully request a copy of the following document:

Title: (U) Ten Steps into Space

Accession Number: ADB193716

Corporate Author: FRANKLIN INST PHILADELPHIA PA

Report Date: Dec 1958

Descriptive Note: Monograph no. 6

Pages:212 Page(s)

Report Number: XD - XD (XD)

Monitor Series: XD

Neither DOD nor NARA could locate the document, so I thought I would forward the request to DTIC.

Thank you so much for your time, and I am very much looking forward to your response.

Atc 1

Sincerely,

John Greenewald, Jr.

[redacted]

Sincerely,

John Greenewald, Jr.

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14-F-0337

DEFENSE TECHNICAL INFORMATION CENTER
8725 JOHN J. KINGMAN RD, STE 0944
FT. BELVOIR, VA 22060-6218

IN REPLY REFER TO: DTIC-R (FOIA 2014-57)

DEC 31 2013

Mr. John Greenewald
[redacted]

Dear Mr. Greenewald:

This is in response to your email dated December 14, 2013, received in this office December 16, 2013, requesting information under the Freedom of Information Act (FOIA) (enclosure 1). Under Department of Defense rules implementing the FOIA, published at 32 CFR 286, your request was categorized as "other."

The document that you have requested, ADB193716, entitled, "*Ten Steps Into Space*" is limited to U.S. Government agencies and their contractors only; therefore, your request has been forwarded to the organization below for processing and direct response back to you. Please direct all future correspondence related to document ADB193716 to:

Defense Freedom of Information Division
1155 Defense Pentagon
Washington, DC 20301-1155

To date, there are no assessable fees for services from DTIC. Please understand that other members of the public may submit a FOIA request for copies of FOIA requests received by this office or the names of those who have submitted requests. Should such occur, your name and, if asked for, a copy of your request will be released; however, your home address and home telephone number will not be released. Other private citizens who have obtained your name by using such a request may contact you. However, correspondence from the DoD about your request will be on official letterhead. If you have any questions, please call me at (703) 767-9204. Thank you for your interest in obtaining information from DTIC.

Sincerely,

MICHAEL HAMILTON
FOIA Program Manager

Enclosure

Atc 2

57

TEN STEPS INTO SPACE

[redacted]

211pg

MONOGRAPH NO. 6

Published at Lancaster, Pa., under the auspices of
JOURNAL OF THE FRANKLIN INSTITUTE
Philadelphia, Pa.

December, 1958

PRICE $4.00

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TEN STEPS INTO SPACE

TEN STEPS INTO SPACE

A Series of Lectures
sponsored by The Franklin Institute
March-May, 1958, in Philadelphia

MONOGRAPH NO. 6

Published at Lancaster, Pa., under the auspices of
JOURNAL OF THE FRANKLIN INSTITUTE
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December, 1958

Printed in the United States of America

FOREWORD

The Franklin Institute has for many years been deeply concerned with missiles, Earth satellites and space travel. Its Laboratories for Research and Development have been steadily engaged in vital tasks which helped produce working mechanisms for rockets. In 1956 it sponsored a symposium on "Earth Satellites as Research Vehicles," which, in published form, is one of the two or three authoritative, pre-Sputnik books on the subject. On a popular level, the staff of the Planetarium has operated on the premise that man would eventually travel in space. We have offered planetarium demonstrations which, by taking advantage of the planetarium illusion techniques, served to alter our sense of space and time and permitted us to move ahead into the future. Such demonstrations have included "trips" to the moon, to Mars and to Saturn. Thus, the Institute has kept abreast of latest developments in the rocket and missile field.

With the advent of the eminently successful V-2 rocket, the mechanism was available which could ultimately be used by man for the exploration of space. Little by little, the necessary steps to achieve this goal were taken. Finally, on the fourth day of October, 1957, the launching of the first Earth satellite fired the imagination of an unprepared world. Those of us who had been involved in some facet of this adventure were called upon to explain the avalanche of fast-moving events occurring at that time. Science fiction had suddenly become scientific fact—and everyone wanted to know something about rockets, satellites and space travel.

Against this background, the Institute, seeking to serve engineers and scientists of the Philadelphia area, sponsored a series of ten semi-technical lectures on astronautics, in the spring of this year. Ranking workers in the space travel field were invited to the Institute to lecture on their specialties. The Dolfinger-McMahon Foundation made it financially possible for these outstanding speakers to travel to Philadelphia from all parts of the United States. The Institute wishes publicly to acknowledge its gratitude, both to the men who took their valuable time to lecture, and to the Foundation, without whose help the series could not have been carried out.

The lecture course proved so successful that it was decided to publish the series as a JOURNAL Monograph, in order that a wider audience might be reached. The present volume represents essentially the verbatim remarks of the lecturers—condensed slightly and edited to make the material suitable for written rather than oral presentation. It is our hope that *Ten Steps into Space* will serve to clarify the basic principles and problems of space travel for those who seek to understand the events of the past year in the astronautics field.

The Franklin Institute may have similar Astronautics Series in the future, but in no case will it be able to present a sequence of more timely lectures. To all who helped make this possible—our sincerest thanks.

I. M. LEVITT

October 15, 1958
Philadelphia, Pa.

CONTENTS

The Authors . . .

WILLY LEY, space travel historian and author, was one of the founders of the German Rocket Society which pioneered rocket research in 1927. From that time on, his interest in rockets has been alternately his profession and his hobby. He has served as science editor for the newspaper *PM*, as a research engineer for the Washington (D. C.) Institute of Technology, and as consultant to the Office of Technical Services of the Department of Commerce. He is the author of many recent and popular books on space travel.

KURT R. STEHLING, head of the Naval Research Laboratory's Vanguard Rocket Division, is consulting editor on missiles for *Aviation Age* and a contributing editor for *Astronautics*. He was President of the Canadian Rocket Society in 1947 and is a Director of the American Rocket Society. He was a member of the ARS Space Flight Committee which recommended a minimum satellite program to the National Science Foundation, and Project Farside is a derivative of two of his papers on high-altitude balloon rocket launchings. Prior to his present assignment, he was group leader in the Rocket Fluid Mechanics & Heat Transfer Group at Bell Aircraft, and a Research Associate in jet propulsion at Princeton University.

H. W. RITCHEY, Vice President of Thiokol Chemical Corporation, has guided propellant development work and rocket engine design in Thiokol's Rocket Division for eight years. Under his direction, rocket engine performance has greatly improved From 1938 to 1941 he was a petroleum chemist for the Union Oil Company, where he explored the possibilities of non-corrosive, detergent lubricating-oil additives. During World War II he taught courses in chemistry and jet propulsion at the U. S. Naval Postgraduate School. Prior to joining the Thiokol staff, he was an atomic reactor engineer for General Electric Company.

S. FRED SINGER, Associate Professor of Physics at the University of Maryland, is a member of the Technical Panel on Rockets and the Technical Panel on Cosmic Rays of the U. S. National Committee for the International Geophysical Year. He is also a member of the Space Flight Technical Committee of the American Rocket Society. From 1950-1953 he was Scientific Liaison Officer with the Office of Naval Research at the U. S. Embassy in London, during which time he studied European research programs in upper atmosphere physics. With the Johns Hopkins University from 1946 to 1950, he engaged in high altitude research with rockets.

PAUL HERGET, staff head of the Vanguard Computer Center and Director of the University of Cincinnati Observatory, is responsible for predictions and calculations of the orbits of U. S. artificial satellites. He was consultant to the Manhattan Project and is Chairman of the Astronomy Panel for the National Science Foundation. For thirteen years he has been directing a nationwide project in which twelve major observatories carry on research on the minor planets. In 1954 he rediscovered the minor planet Athalia, "lost" for half a century.

GERHARD HELLER is Deputy Director of the Research Projects Laboratory, U. S. Army Ordnance Missile Command, at Redstone Arsenal. He worked on the development of the Explorer from its beginnings in 1953, and was in the control room at the launching of the United States' first successful artificial satellite.

KRAFFT A. EHRICKE, Assistant to the Technical Director of the Astronautics Division of Convair, has been working on rockets and space travel since 1942. As a research engineer for German Army Ordnance, he helped develop the V-2 rocket. After the war he came to the United States where he worked on advanced V-2 systems analysis and ramjet projects at Fort Bliss, Texas. Before joining Convair in 1954, he served at Redstone Arsenal as Chief of the Gasdynamics Section, and was a preliminary design engineer with Bell Aircraft Corporation.

J. I. F. KING is Manager of Physics Operation in the Aero Physics Laboratory of the Aero Sciences Laboratories, at General Electric Company's Missile and Space Vehicle Department. Trained as a theoretical physicist, he has concentrated his work in the fields of radiative heat transfer, atmospheric cooling rate and infrared transmission. Before going to General Electric in 1957, he was Section Chief in the Thermal Radiation Laboratory at the Air Force Cambridge Research Center.

DAVID G. SIMONS is Chief of the Space Biology Branch of the Aero-Medical Field Laboratory at the Air Force Missile Development Center, Holloman Air Force Base. In this capacity he is in charge of the U. S. Air Force program for evaluating the biological hazards of primary cosmic radiation at altitudes above 85,000 ft. In August 1957, he set a new manned balloon altitude record by soaring to an approximate 100,000-ft. altitude. Prior to the Korean War, he was assigned to the Army Air Corps Aero-Medical Laboratory at Wright Field, where he studied physiological responses of monkeys to weightlessness.

I. M. LEVITT, Director of the Fels Planetarium of The Franklin Institute, has been actively interested in space travel from 1936 on. Since 1952 he has been writing a syndicated space-travel column, "Wonders of the Universe," which now reaches 60 million readers in nine languages. In 1954 at the Fifth International Astronautical Federation Congress in Austria, he outlined the advantages of an unmanned, uninstrumented satellite vehicle. He is a contributor to many newspapers, magazines and encyclopedia of science. In addition to being a writer and lecturer, he has appeared on TV and radio. Earlier this year, he developed a series of clocks to show the time-dilation effect on space travelers.

JOURNAL OF THE FRANKLIN INSTITUTE MONOGRAPHS

NANCY S. GLENN, *Editor*

1. *Exhaust Turbocharging of Internal Combustion Engines,* by Alfred J. Büchi. 1953. 75 pages. $1.00.

2. *Earth Satellites as Research Vehicles,* a Symposium. 1956. 115 pages. $2.50. (Out of print.)

3. *Automatic Coding,* a Symposium. 1957. 118 pages. $3.00. (Out of print.)

4. *Particulate Emission,* a Symposium. 1958. 105 pages. $3.00.

5. *The Airways Modernization Board—Its Mission and Methods,* a Symposium. 1958. 160 pages. $4.00.

6. *Ten Steps into Space,* a Series of Lectures. 1958. 202 pages. $4.00.

THE LONG HISTORY OF SPACE TRAVEL

BY

WILLY LEY[1]

INTRODUCTION

It may sound a bit strange for somebody to talk about the history of something which has just begun or is going to begin in the near future, depending on your interpretation of the term "space travel." Just the same you cannot do anything unless you think of it first and when you think of it history so-to-speak begins.

I am reminded of the newspaper interview which I gave roughly a week ago, in which I casually said that of course the men are tired now of just orbiting around the earth and they would like to give their rocket a kick in the perigee or at least send their planetary probe into interplanetary space.

Whereupon, the newspaperman looked at me and said, "Who made up all these new words?"

I said, "These new words were made up by the astronomers quite a number of years ago and especially those words which are bandied around in the daily press these days were, for the most part, coined by Johannes Kepler, who died in 1630. So these are not precisely new words."

But this, of course, is the fulcrum of my lecture. The words are not new. The ideas are not new. But you will have to make a distinction (at least I make one) between what I call the history and the pre-history. The history of space travel in my book and to my mind began at the time when a scientist, preferably a modern scientist, sat down and said, "Now, if we wanted to fly into space, what would we have to do?" To phrase it in very modern language: What velocities would we need for which job? What means of propulsion would we need for such a job? What additional side issues crop up, for example, things such as "can you receive a radio signal," or less important things such as "will the pilot become unconscious or not?" This is what I call the history.

PRE-HISTORY

The pre-history begins with the idea itself, and the idea (and, again, this is a straight steal from the astronomers) of space travel naturally presupposed the idea of other bodies in space. As long as you don't know that Hawaii exists you cannot wish to go there. As long as you consider the lights in the sky to be just lights in the sky and nothing else, you cannot wish to go there. Whether your means for doing it would be effective or not is still an entirely different problem, but not even the wish could come

[1] Space Historian, Jackson Heights, N. Y.

up or the concept couldn't start unless you knew that there is a place at which you could conceivably arrive.

This means, of course, that space travel actually began more or less at the moment when the people who were interested in what is now called astronomy started the concept of other worlds in space.

Let me say something which does not really belong to the theme, but which is important in the sense that it is often overlooked. The oldest astronomy was astronomy of position only. The oldest astronomers were not interested in what we now take for granted; in the nature of the heavenly bodies which they observed. They were interested in their location, in their appearance, disappearance and reappearance—in short, in their positions

It was only, let us say, at about the time of Christ (this fits pretty nicely although there is no direct connection) when the intelligent people, at least of the Mediterranean world, became convinced for the first time that the moon was actually a body. That was put on paper for the first time by Plutarch who died in 120, and only a few decades later the first science fiction story was written; of course, it was a story about a flight to the moon Flight in this case is to be taken literally.

I might add that the aerodynamics were awful. Not only was the atmosphere of the earth supposed to reach to the moon, but the hero of the Greek poet took one wing of an eagle and one of a vulture and by means of this kind of equipment he made a flight to the moon. And then the gods took his wings away.

In any event, the man who wrote the story was a Greek by the name of Lukian of Samosata, and, to the best of our knowledge, this was written in 160 A.D.

I just mentioned that we first needed the concept of other worlds before we could get the concept of space travel This is a statement where you have to be awfully careful about your semantics because all of you who at one time or another studied philosophy will remember, it is hoped, that there was for a long time a controversy on the so-called plurality of the worlds. That was an entirely different thing.

This plurality of the worlds concept in philosophy had nothing at all to do with the fact whether the moon should be considered a light in the sky or a silver shield in the sky, as Pliny reported as an old belief, or whether it was a solid body. That was an entirely different thing having to do with Ptolemaic ideas about the construction of the world. It is of no other than historical interest to us now and doesn't have to concern us in this lecture. The reason I brought it up is that that is not the same.

In the later Middle Ages this old philosophical fight was revived on theological grounds, this time meaning a world like, say, the moon. These people always talked about the moon. You have to remember that this was before the invention of the telescope and only the moon is large enough, not counting the sun, to appear as a visible disk in the sky, while everything else needs a telescope to look different from a pinpoint.

But before the philosophical theological discussion could go too far, namely, in 1277, the Bishop of Paris said, with authority by the Pope, that it would be wrong to think that there can be only one inhabited world for theological reasons. So the stage now was set. The stage was set for the astronomical discoveries which hinged partly on a better mathematical understanding of what was going on, and partly on the invention of the instrument which we now call the telescope.

THE PERIOD FROM 1543 TO 1750

The mathematical understanding of what went on was given by two books: by Copernicus' original work *De Revolutionibus Orbium Coelsticum* (On the Revolutions of the Celestial Bodies) published in 1543; and Johannes Kepler's *Nova Astronomia* (The New Astronomy) which had the very interesting subtitle of *De Motibus Stellae Martis* (On the Motions of Mars) (interesting to us because that is where Kepler got his new astronomy), which was published in 1609. Then Galileo Galilei's *Siderius Nuncius* appeared in 1610, on his telescopic discoveries And here you have one of these very important psychological differences.

I have often said (a few dozen times at the very least since October 4 of last year) that Sputnik I mainly had the purpose to prove to everybody that it could be done. Of course, all astronomers, most mathematicians and a good many engineers knew that before, but the ones who were not astronomers and not mathematicians and not engineers had to be shown.

Back in 1610 in the astronomical field we had the same story. The Greek philosopher Anaxagoras had suspected that there might be mountains on the moon. Plutarch in 100 A.D. had logically proved that there should be, but Galileo in 1608 could say "I saw them." And that made the difference.

The next book dealing with a trip to the moon came after Galilei. In this whole interval from Lukian of Samosata in about 160 A.D. to the next one in 1634 there is just one minor item. You all know about Ariosto's famous poem *Orlando Furioso* which was finished in 1516 after some ten years of writing it.[2] (I may add here as an aside that this book is also very interesting for the history of technology in various aspects). The *Orlando Furioso* contains one short story about a trip to the moon by somebody else. This is the only item between Lukian of Samosata in 160 and the next one in 1634, Kepler's *Somnium*.

This was not only the time when the telescope was invented, but it was also the time when the printing press had been invented, and Lukian's book had been reprinted on the newly invented printing press. The reprints of the original Greek editions were in 1496, 1503, 1522, 1526 and 1535. That means five reprints of the original Greek in forty years. For those people who were not well educated there were Latin editions. The Latin editions

[2] First publication was in 1516, but this was not the complete work, which appeared in 1532.

appeared in 1475, 1493, 1543 and 1549. A little later Johannes Kepler added one which was published in 1634, Latin translation. There were also two bilingual editions, Greek and Latin, in 1615 and 1619. Finally, late in the game there was an English edition in 1634.

This started something. In the first place, Johannes Kepler, the man who discovered the laws of planetary motion, wrote a book which falls into this general theme. Its title is *Somnium* (Sleep), meaning dream in this case, and it is actually a kind of philosophy and geography of the moon. It contains two things which were new. One was that he gives the distance to the moon roughly as it is now in our textbooks—50,000 German miles, which works out to close to 300,000 English miles. This is a little bit too large, but it is the first printed estimate of the distance which is reasonably close to the truth.

The second thing that is new in Kepler's book is that although the Earth has an atmosphere and the moon has an atmosphere, these two atmospheres do not touch. There is empty space in between. And he makes one more remark which I find very charming, and that is the following. (Incidentally, it is factually wrong, but shows thinking in the right direction.) He says that you must all have observed how a spider when it is chased from its web drops down and pulls its legs to its body. This way, Kepler said, a traveler in space between Earth and the moon will have his arms and legs folded up against his body because the larger part attracts the smaller parts.

This was followed (and I am skimping now a little bit in my list) by a book by Dr. Francis Godwin of England first published in 1638—*The Man in the Moone*—of course, a man who goes there. This is done by the very simple—not too simple, but let us say non-technical method of imagining a race of birds called ganzas which migrate to the moon every winter and they take the traveler along.

The interesting point is that Dr. Godwin did not even have to invent the ganzas, or at least he saw something that was his model for it. The extinct dodo was the model for Godwin's ganzas and, of course, there are two small errors in the story. One is that the dodos couldn't fly at all, and the second one is that the bird that was exhibited in London as a dodo at the time, and presumably seen by Dr. Godwin, happened not to be a dodo but a solitary.

But let us not go into these complications of history. Let us just point out that Dr. Godwin's book was very successful. Between 1638 and 1768 it was published or printed twenty-five times in four different languages: English original, French edition, German edition and a Latin edition.

By that time, when it came out last, the moon had slowly begun to fall into disgrace. In about 1650, and this is the earliest date I have been able to find, an Italian by the name of Giovannia Battista Riccioli wrote a book which was called *Almagestum Novum*—straight astronomy. It is this book which contains I believe for the first time (I wouldn't swear to it in court) the statement that there are no seas on the moon, there are probably no lakes on the moon, no clouds have been seen on the moon and it probably has very little, if any, air.

I made myself a small note here which might amuse you as it amused me. Some time after Dr. Godwin, who had this traveler carried to the moon by ganzas, as I told you, a "poet" by the name of Meston, an Englishman, wrote a poem which was a call to his muse; and in this poem he expected his muse to inspire him with poetic thoughts which "are soaring in high Pindaric stanzas, above Gonzales and his ganzas."

The next man in literary history is a man who is often thought not to be a real character, but he was one, namely, the Frenchman Cyrano de Bergerac. His *Voyage dans la Lune* was printed for the first time "sans privilege," which means anybody could copy it, in 1650; then with privilege it was printed six years later. This is the first book in which rockets are used to carry the box in which the traveler sits.

This was imitated a little while later by an anonymous Frenchman who makes things a little more modern sounding. His hero goes to the moon by rocket and returns by parachute. The re-entry problem isn't quite that simple, I am afraid, but I am just reporting now on what other people wrote in the past.

Then, of course, we have one English story printed in 1728 by a man whose name we don't know, but he put the pen name of Murtagh McDermott on the cover. Well, I wouldn't say he foresaw the stage or step principle, but his ship consists of ten hulls inside each other, in case something goes wrong. And that something might go wrong is shown by the fact that he is propelled to the moon by placing that ship on top of a mine shaft, in which 7,000 kegs of gunpowder are going to be ignited.

You will see that by this time a number of ideas had already been vaguely thrown at the public, but this ended in about 1750 and then there was a long hiatus of roughly one century. This is speaking from the fiction point of view, but from the point of view of history of science there is some very early scientific thinking precisely in about this interval, all at about the year 1820.

EARLY SCIENTIFIC THINKING

First comes a man, whose nationality I do not know, who wrote in very difficult professorial German, but whose name sounds Dutch—it was Franz von Paula Gruithuisen. Professor Gruithuisen, in the first place, thought he had discovered a ruin on the moon. It is one of the cases where you can see this ruin quite clearly with a small telescope if you know what to look for, but it doesn't show up in a big one! And I mean this seriously. I have seen it myself in a 4-in. telescope.

In the second place, Gruithuisen was convinced that the planet Venus was inhabited. He didn't say, as was left to our century to claim, by little green men, but he did say that it was inhabited.

In the meantime, a man who is really one of the great ones in science, a mathematician, Karl Friedrich Gauss, and the Viennese astronomer von Littrow both were thinking about the idea of what you might do if you wanted to communicate with the inhabitants of other planets. Both agreed that they might have nothing in common except the laws of mathematics. Two plus two must be four on Mars too, no matter what you call the figures.

And so it was Karl Friedrich Gauss who evolved the idea of producing a giant mathematical symbol, the right angled triangle with the three squares attached to it. Gauss, looking for a sufficiently large blackboard to write it on for the Martians to see, thought of the Siberian tundra. What he wanted to do was to plant the triangle with wheat or rye, a grain which ripens yellow, surrounded by dark green pine forests with a thickness of the line of about fifty miles, which would make a nice contrast between pine forest and wheat or rye fields

I sometimes wonder whether this might not come up as a proposal from east of the Iron Curtain one of these days. But the original inventor was Karl Friedrich Gauss. Littrow in Vienna improved on this to some extent in saying that this would be an unchanging symbol and he wanted changing symbols which he wanted to produce by taking a desert (he said the Sahara, but that was just a name used) in which he would dig trenches forming figures fifty miles in diameter. The trenches would be filled with water, of course, and then he would float kerosene on top of the water and ignite this to send signals into space.

EARLY SCIENCE FICTION

These were, I wish to point out, scientific ideas. They did not originate in the circles of people who wrote stories. The story writers came to the fore in 1865, a year in which four novels all about trips into space were published. One of them is unimportant even though it was by Dumas, and the other three managed to exhaust all possible ideas.

One of the three, which I would like to discuss quickly, was Jules Verne's famous story *De la Terre a la Lune* (From the Earth to the Moon), in which, as you know, escape velocity is produced by a cannon shot from a gigantic cannon in Florida imbedded in a solid stone mountain. I might add here that Jules Verne moved, for the sake of the story, Stone Mountain near Atlanta to Florida. He knew it wasn't there, but he moved it there for story purposes. I don't know whether the story which is now being told in Atlanta existed then; I doubt it. (The story which they now tell in A[illegible]nta is that Stone Mountain was one day thrown by California at Florida, but missed.) So Verne did move it to Florida for story purposes.

The cannon shot was one idea. The second idea came from an anonymous Englishman whose book has the very simple title *History of a Voyage to the Moon*. This anonymous Englishman thought up the idea of finding a substance which did not have any weight. He had an ingenious treatment. This substance is weightless and carries things unless there is iron beneath it, in which case it loses its power.

His ship consists very simply of a large room, let us say, built of oakwood, lined with sheet metal, and it has two balls of this carrying substance at both ends. And it has two iron flippers, you might say, which can be lifted to a position below it to destroy the power. If you let them drop, then the power is there again.

The thing that amuses me about this old English story written in 1865 is that the man constructs his space ship as a hydroponic system He grows plants in it so that the travelers have oxygen to breathe. Again, I think that this is the first time that this has ever been mentioned.

The third idea of that year (all in the same year, strangely enough, 1865) came from a Frenchman by the name of Achille Eyraud, and his book is very simply called *Voyage a Vénus* (Voyage to Venus). He uses a reaction motor, a reaction motor which runs on solar energy and uses water as the reaction mass. He then ruins the whole story, and his fame with it, by the idea of catching the reaction mass and feeding it back into the fuel tank. But he did have the idea of the reaction motor.

So you see that the main ideas of friction which we now have in 2,222½ variations were all taken from early in the 19th century, if not earlier: the weightless substance, which then became especially famous through H. G. Wells, written in 1899; the reaction propulsion, which is the staple now; and the cannon shot, which has been discontinued for literary purposes.

I have to mention two more things here. One is that 1877 was the year when Mars came especially close. It was possible for Professor Asaph Hall in Washington to discover the two small moons of Mars, and it was possible for Giovanni Schiaparelli working in Italy to send out the astonishing news of the "canali" on Mars, the lines which still are not explained. This was in 1877, and for this reason fictional interest then shifted from the moon, that was known to be airless by then, to the planet Mars.

The masterpiece, of course, was the thing that happened around 1902, also in France, where a very rich lady with enormous amounts of money put down the necessary money for a prize of 100,000 francs in gold to be paid to the man who started communication with another heavenly body, except the planet Mars. The planet Mars seemed to be too simple, but the French astronomer Camille Flammarion, whose book about Mars (*La Planète Mars*) was responsible for the prize in the first place, withdrew in horror and said this is the "une idée bizarre" to rule out the one planet which seems to be in a position to participate. But you see the enormous optimism that was around then.

I will add one more thing (and this is not quite fair now for a reason which I will explain in a moment). The best space travel story ever written prior to, say, 1920 or thereabouts was a German one written and published in 1897. This contains the complete theory of space travel by reaction, contains theory of intersecting orbits and things like that. But when I say it isn't quite fair, it isn't quite fair because the man who wrote this novel was a scientist himself. He was a professional mathematician.

DEVELOPMENT OF SPACE TRAVEL THEORY

In 1897 this man, Professor Kurd Lasswitz, actually bridged the gap between what I call the prehistory of which I have talked so far and the history which began in about his time because now science was ripe for actually tackling the problem.

The first man who actually spoke about a space ship and meant it literally (a space ship that he would undertake to build if he were given the money and the help) was an inventor by the name of Hermann Ganswindt. I knew old Ganswindt who was in his late 70's at the time and he told me that his ideas went back to 1870. Whether this is true I have no way of judging. I can't say that he told me a story. He was an old man. Past events tend to coalesce in retrospect. But I can say this much, that I went through his documents and he could prove to me by printed programs that in the spring of 1891 he had delivered a lecture in which he declared that the new century would bring both aviation and space travel. As we now know, Ganswindt was right. However, Ganswindt himself, although he talked a lot about it, talked in public and for publication, did not produce what you might call a scientific paper.

The first scientific paper on space travel was produced by a Russian by the name of Konstantin Eduardovitch Ziolkovsky. I know from others, not from Ziolkovsky himself (although I had correspondence with him), that he wrote his paper in 1898 and he sent it to a journal, now defunct, which was quite similar to our *Scientific Monthly*, which is defunct too unfortunately. It is also a matter of record that the editor needed five years before he made up his mind whether this should be published or not.

We know this because other scientists were asked what they thought of the manuscript and they apparently couldn't find anything wrong with it and so it was published in 1903. And it influenced nobody. It made absolutely no impression. Outside Russia nobody could read it, of course. Inside Russia nobody paid any attention and the airplane still had to be invented anyway.

You can say, then, that these two men—Ganswindt and Ziolkovsky—are early forerunners. You can say that the next man in the scientific field was the American, Professor Robert H. Goddard. Goddard, in 1914, took out a patent in which the step principle is mentioned. The little I know about patent law makes me wonder how he could get the patent because the step principle had been patented about three years earlier in Belgium.

There must have been a legal technicality, but I am quite sure that Goddard did not know about the Belgian patent and Goddard wrote his first book on space travel in the latter part of the First World War. It was published by the Smithsonian Institute with 1919 on the title page, but actually released in January, 1920.

It also didn't cause too much of a stir, and I may here say something which I consider important because an issue is being made of it on occasion and by a few people. While there is no doubt about the priority of Dr. Goddard in the Western world, I consider that Dr. Goddard's importance lies in the field of rocket research and rocket engineering, not in the field of space travel.

Dr. Goddard mentioned space travel in only one aspect, an unmanned shot to the moon. But all his life, and in all of his publications, he was concerned with rocket research which, of course, is as closely related to space travel as is the right hand to the left, but still it is not quite the same thing. And personal interests do count and should be evaluated in retrospect.

At any rate, Professor Goddard had his first rocket motor for liquid fuels running on November 1, 1923 and there was no earlier one. The first flight of a liquid fuel rocket of Professor Goddard's design was on March 16, 1926, and, again, there was no earlier one. So his position is absolutely secure.

Things then shifted over in the direction of Europe for a while. The book which, if you discount Ziolkovsky, is the foundation of space travel theory as distinct from rocket theory is Hermann Oberth's *Rocket into Interplanetary Space*. This, of course, is a translation of the original German title. The book was published early in 1923.

This book covers, as you can see in retrospect much better than you could see then (although I read it then) an amazing variety of subjects. It is not only rocket motor computations. It is not only ascent computations. It contains questions of pilot resistance to acceleration. It contains a discussion of a shot to the moon. It contains the first published plan for a station in space. All this in 1923!

As a matter of fact, in this book you can find a forecast of what happened to the second Vanguard that failed. He has a discussion that lasts for one and a half pages (and mostly mathematical) on the fact that if a rocket is very tall and thin all the aerodynamic forces tend to break it during the ascent.

It is highly interesting, however, that in 1923, when Professor Oberth managed to speak about a station in space, that he did not forecast the unmanned artificial satellite. The reason is a strange one. He wrote the book, or the actual writing, I would say, was from late 1917 to late 1921. At this time, radio was not exactly under military secrecy, but only the people working in radio knew anything about the subject.

For example, Oberth, in all seriousness, in his book has a lost expedition ask the space station by means of signals by hand mirrors reflecting sunlight. Well, maybe even radio men could not have figured out at that time that on a three centimeter wave you could call the space station and would get your call through with the power contained in four flashlight batteries But it is interesting that he doesn't mention radio at all, because what radio there was was long wave stuff, needing enormous stations, and whatever research may have been going on was not known to him; and telemetering was not invented until three years later.

There was one more important work, this one by a German (Professor Oberth has to be considered an Austrian by birth, although he lived in Germany later), a Dr. Walter Hohmann, who in 1925 published a work which goes into great detail of leaving the Earth, going to the moon, around the moon to another planet, returning and re-entry. He has a long re-entry discussion in this book published in 1925.

The interesting point here is that it is all pure theory, pure in the purest sense of the word. I asked Dr. Hohmann, who is no longer alive (he died during the war in a bombing raid), by letter before the war started whether he had had any ideas on construction. He wrote that he had tried to think of construction, but he could not find anything that as an engineer he would undertake to develop. But by just assuming thrust of the necessary order he could as early as 1925 work out most of the things that we are now on the verge of doing.

This completes what you might call the history of the idea. By the end of 1925 the idea was complete. What has been contributed to the theory since was either in engineering on how to calculate the exhaust nozzle of a rocket motor, or it was refinement of certain things because obviously the men who calculate the re-entry problem with a slide rule and somebody else who has an IBM machine—well, the man with the IBM machine is likely to be better off and to be able to offer a greater variety of possibilities. But still I feel that by 1925 the story of the idea was complete. What comes after that is the story of building.

CONSTRUCTION OF THE ROCKET

The story of building, which most of you have read somewhere, I have no doubt, is something which can be shortly described in the following way. I told you that Dr. Goddard made his first liquid fuel rocket take off the ground in March, 1926. The first liquid fuel rockets in Europe followed in 1931. Six years later, there was a neck-to-neck race, which nobody knew existed.

The German Army, in 1936, got two liquid-fuel rockets to a height of about 6500 ft. each. Professor Goddard in this country had accomplished 7500 ft. in 1935. And in 1936, presumably somewhat ahead of the Germans since they fired in the middle of December, the Russians got a liquid-fuel rocket to a height of twelve miles.

What then happened was that Dr. Goddard ran out of money, the Russian group that had got a rocket to twelve miles became politically suspect, while the German Army, being an army, just kept working. They didn't get much of a budget. One night Hitler dreamed that long-range missiles couldn't work, so the next day he withdrew all priorities from the long-range missile program. But armies can sometimes, if there is a good general sitting in the right place, shuffle internal funds around in such a way that they do things for which they are not precisely authorized. At least it may take some explaining afterwards why they thought they were authorized, and some work has been done that way once in a while. In this case the German Army ended up with a V-2 rocket which was in one way or another the prototype for all the rockets to come, the liquid fuel rockets, that is.

The next great event in my mind after the V-2 capable of going to a hundred miles vertically was Project Bumper Shot No. 5, which was the first two-stage shot of a liquid fuel rocket. It went to 250 miles, and this height was not surpassed for quite a number of years to come because the next job to be done, all scientists were agreed upon, was not to shoot higher into space, but to put something into space to stay there, an artificial satellite.

As a matter of fact, aside from a Jupiter C shot which went higher than 250 miles, but in the process of traveling 3,000 miles horizontally, the first shot higher than Project Bumper was one that happened by mistake. I am speaking about the famous case of the runaway X-17. The X-17 is a three-stage rocket of which the first stage throws stages 2 and 3 into space so that they can turn around and come down to simulate a re entry, but they wouldn't come down fast enough to simulate really so the second and third stage fire upside down to push the measurement instruments in the nose of the third stage into the dense atmosphere at the highest possible velocity.

Well, one of the X-17's misbehaved and fired all three stages on the way up. The trackers promptly lost it. That was something they didn't expect, but calculations indicate that it must have reached a height of 1,000 miles. This was early last year.

In the middle of last year Project Farside got a rocket up with a balloon launch to a height of 3000 miles. The transmitter cut out at 2700, but it is believed that it rose for another 300 miles after the transmitter stopped.

Then, of course, came the age of the satellites. The next events are easy to prophesy since we know what technological capability there is: a shot to the moon within a year, a flight to a hundred miles or thereabouts in the X-15, and several years from now the first orbital flight by man.

All this is easy to prophesy. I have used this phrase before, but I think it is most apt if you remember everything I have said tonight. It is easy to prophesy the future because it is a future which began quite some time ago.

THE ROCKET AND THE REACTION PRINCIPLE

BY

KURT STEHLING[1]

INTRODUCTION

Everyone is talking about space flight these days; the papers are full of it. This evening, then, why don't we go a little bit into the thing that makes space flight possible—if it is ever going to be possible—namely, the rocket.

The two terms "rockets" and "missiles" are used interchangeably and widely, but they don't always mean the same thing. The traditional rocket is perhaps nothing more than the old Fourth of July firecracker, with which we have more fun than the big rockets at times; and a missile is a vehicle that uses a rocket.

I want to talk about the rocket and how it works, and how a rocket engine behaves. In the short time we have available, I will deal with the rocket engine. That, after all, makes the vehicle go—sometimes, and not always up, either!

Unfortunately, a jet propulsion course of some six months or a year cannot be compressed into an hour, so I will have to gloss over the subject very skimpily and touch the highlights of propulsion developments as we have them today, including some of the subtleties of rocket engines.

THE ROCKET ENGINE

The rocket is the thing that makes the vehicle go. It is the power plant, and when the power plant doesn't work well, the rocket doesn't work well, either. The same is true in an airplane or a motor car. If the power plant stops, drastic things happen; so it is necessary to have a power plant which performs well.

The thing that distinguishes the space vehicle from other vehicles is its unique power plant, the rocket engine, which is an application of jet propulsion. All jet propulsion can be lumped together in a very large body of jet propulsion engines. The turbo jet, the ram jet, the ram rocket, the hybrid engines and the rocket itself and devices that use the momentum principle of the jet which is exhausted through an orifice in the rear of some combuster or other device which generates high velocity particles of gases.

Figure 1 is a drawing of a very elementary propulsion system which would apply, I dare say, to any space vehicle or any rocket vehicle. I shall pick out little points in this figure and try to explain what the functions are and what they mean.

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A rocket missile these days is, of course, to be written with a capital "R" for Rocket and a small "m" for missile, because the rocket missile is largely tanks and engines. Only a very small fraction of a rocket missile today (and of a space ship in the future) is devoted to payload or guidance and control and any ancillary devices that have to do with the rocket function.

The engine, which I call the rocket thrust chamber, the pumps, tanks, etc., makes up perhaps 60 or 70 per cent of the total vehicle weight. Any power plant that I know of requires some fuel, so we must have a place to store the fuel to run the power plant. It doesn't matter whether the power plant is a nuclear power plant or a chemical power plant or an ion power plant—something must be done to a working fluid so that it in turn may produce work. Then, on one end of the fuel (propellant) tank, is the rocket thrust chamber.

People have recognized for a long time that the rocket power plant is the first step towards space flight. The early rocket societies in the 1920's and 1930's realized that a rocket engine has the unique ability to operate in a vacuum—although some people still don't realize it!

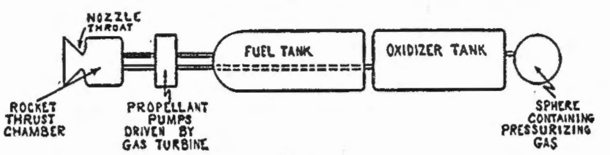

FIG. 1.

The early rocket societies also recognized that space flight can only be achieved with a device such as the rocket engine which is independent of atmosphere, because there is a point beyond which there is no atmosphere. Therefore, if you want to get beyond the atmosphere, you can't use an ordinary gas engine, which needs atmosphere to function; you must use something that carries its own oxygen for its combustion process.

Various scientists long ago recognized this. Dr. Robert H. Goddard in this country wrote some very good treatises on rocket engines and for many years, practically unsupported, he developed a lot of the elements of the rocket propulsion system and worked out the theory of some of the rocket combustion processes. The Europeans, notably the Germans, did a lot of work, too. Professor Hermann Oberth worked out a scheme of propulsion even before Goddard, although Goddard did a lot of practical work.

The rocket propulsion system (shown in Fig. 1) is the critical device that separates the space ship men from the ordinary Earth-breathing boys. The rocket chamber with its propellant tanks permits space flight to occur, because the rocket engine can transcend beyond the atmosphere and, in fact, it works better out of the atmosphere than in the atmosphere. (In our own Vanguard project we found that the performance of the rocket engine in vacuum was better than we had anticipated.)

LIQUID ROCKET FUELS

It is necessary to get the propellants or fuels into the rocket chamber. This is done by means of two tanks connected to the rocket chamber.

Before I go on to more elements, I want to stop a moment to analyze some of the components. First, we have a combustor, a device which permits the rapid burning of some combustible. The combustibles are classified as fuels and oxidizers. Most people just call the whole thing "fuel." The fuel, however, is only one half of the propellant, or a portion of it; the other portion is the oxidizer, or oxygen. The two tanks connected to the rocket chamber carry fuel and an oxidizer, respectively, and there are many fuels and many oxidizers.

Hydrocarbons

One type of fuel is the hydrocarbon. Kerosene and gasoline are the two major ones, and there are also jet fuels which are related to the ones above. These are ordinary common garden variety of hydrocarbons which the early rocket societies used, which Dr. Goddard used in his work, and which perform very well.

The reason for using them is largely logistic. The Military in this country are still the largest buyers of missiles and they don't want to be bothered by carrying exotic propellant combinations all over the world if they have to fire missiles from some God-forsaken spot such as the South Pacific or Siberia. They want propellants which are easily available and reasonably cheap, for firing a missile is an expensive business. These large rocket engines use propellants at the tremendous rate of several thousand dollars a second.

Alcohol

Methyl alcohol is another fuel, but it is reasonably passé at this time, for very few rockets these days use alcohol.

Amines

The amines are interesting propellants because they have little or no carbon in their molecules. When a fuel has little or no carbon in it, it becomes a more efficient fuel because the exhaust which comes out of the rocket engine has a lighter molecular weight and also the carbon does not eat up useless energy in the rocket engine.

One of these interesting amines is ordinary ammonia. It is fairly cheap, a little hard to store, but readily available. Another one is hydrazine, which is a derivative of ammonia. Then there are so-called substitutes of hydrazines, one of which, called dimazine, is used in our second stage in the Vanguard. It is a very energetic liquid, very nice material to use, and it gives more energy than the hydrocarbons. We are using amines because they give more energy per pound of propellant weight.

OXIDIZERS

In the oxidizer class, the normal oxidizer is air, but chlorine or fluorine can be used. The favorite these days for rocket systems is oxygen in liquid form. One reason for its wide use is that it is easy to liquefy oxygen—perhaps I shouldn't say easy to liquefy oxygen, but it is easy to obtain liquid oxygen by building a big liquefaction apparatus consisting of huge machines that pump day and night producing liquid oxygen, and break down just before you want to use them!

Liquid gases have their limitations, of course. Liquid gases are difficult things to use. They are hard to pump and hard to store because they are so cold the valves freeze. They tend to react with materials—substances brought into contact with liquid oxygen will be set on fire.

Liquid fluorine is a marvelously energetic material, but difficult to handle.

Of the three main classes of oxidizers, chlorine is out for various reasons; fluorine is an exciting possibility for the oxidizer in the rocket engine; but oxygen is much more widely used.

Another interesting oxidizer that the Germans first tried is 90 per cent hydrogen peroxide—not the bleaching variety! Another nice one is concentrated nitric acid. These two substances alone are rather nasty liquids. Anything in the rocket business is nasty, but particularly the substances one must handle, such as the liquid propellants. Nitric acid has one advantage that the liquid gases do not. It can be stored for a long time (in stainless steel drums for months and years).

THE ROCKET CHAMBER

The rocket chamber permits propulsion to occur in vacuum, which in turn makes space flight possible. I know this is the end product we all look for, but without such a device as the rocket chamber, we cannot achieve space flight in any reasonable way. It is the only practical way which will propel you in a vacuum.

Combustion

Now, how does it do this? Let us assume that we have sprayed the two liquids into the chamber and they have begun to burn. There are two ways in which they begin to burn: they can be ignited with a sparkplug (as was done in the X-1 aircraft which used alcohol and oxygen); or, fluorine and ammonia, which ignite on contact, may be used There is no problem of ignition—as soon as the liquids are sprayed into the chamber, they will burn.

Hydrogen peroxide, which is hypergolic (self-igniting), may be used; and peroxide may be used with alcohol or gasoline.

Combustion, then, has been produced in the rocket chamber. Let me review the cycle once more. There is a little slug of propellant (fuel) in one of the pipes shown in Fig. 1 and another slug of propellant (oxidizer) in the other pipe, both travelling into the chamber by devious means which I will describe later. The two propellants become mixed, just as the carburetor in your car mixes air and gasoline, and upon mixing, they ignite, either from an external stimulus or in a rather crude way (as was done in our first stage engine) from an igniter placed in the chamber and set off. This creates a great "spritzel" of fire in the chamber and away goes the rocket.

For most rocket engines, such as the Vanguard, the combustion process must be ignited only once. Once it begins to burn, it continues to burn—you hope! The combustion in a rocket chamber is really ordinary burning, similar to a Bunsen burner. In fact, if a microphone is placed next to a rough burning Bunsen burner and amplified a few thousand times or a few million times, it sounds just like a rocket engine—worse than many!

Thrust

There is a regular chemical reaction with the particles that are burned—they travel down the chamber because they have no place else to go. This burning immediately increases the chamber pressure from atmospheric to 500 psi., which is an average chamber pressure. Most rockets today work at 500 psi., although some can work at 300, 200 and even less, and some at higher.

The burning particles come to the end of the chamber which may, in the case of a 50,000-lb. thrust chamber, be any length, depending on the pressure of operation. If the pressure is very high, the chamber size shrinks. If it is very low, the chamber size expands.

The gases move along at high temperature in the thrust chamber (a 50,000-lb. thrust chamber is 2 ft. in diameter), and they begin to accelerate because all the propellants are coming in but they are all burning away at a given rate. The material, then, in order to get out of the chamber, must accelerate. It can't go at a steady speed. Indeed, it begins to accelerate more rapidly when it reaches the constriction of the throat.

The gases now expand to the outside into what we call the nozzle. The nozzle has an interesting function. The gases are generated in the chamber and try to get out of it; on getting out, they impart momentum to the chamber. We have, then, a variation of momentum with time, or a variation of mass flow with time, which produces a rate of change of momentum which in turn produces thrust.

The thrust that is produced in the rocket is nothing but the product of mass times acceleration. The mass is in the chamber and the acceleration is produced by the change of velocity and, later, the change in acceleration of particles going through the nozzle, which results in a net force or thrust. This is the thrust of the rocket engine.

On exhaustion of the chamber, thrust is produced at the throat, or entrance to the nozzle, where there is a very violent change in velocity, with a later change in acceleration in the nozzle, which is also called a supersonic diffuser.

The gases which travel at sonic speed, Mach 1, are accelerated to a higher Mach number depending on the initial pressure ratio in the chamber which may be as high as Mach 5 or 6, or even higher. This acceleration to supersonic speed produces the thrust. A portion of the thrust is produced in the chamber; a good deal more is produced in the throat; and there is a certain amplification factor produced by the nozzle. The nozzle may produce roughly 40 per cent of the thrust, with the other 60 per cent produced before the gases reach the nozzle. This is entirely a function of the size of the nozzle and the pressure ratio across the throat.

The Nozzle

The gases expand until they reach a pressure which is determined by the length of the nozzle and the initial pressure ratio in the chamber. In vacuum, these gases want to expand to infinity and reach zero pressure. So, to get all possible work out of these gases, which impinge on the walls as they accelerate, in total vacuum the nozzle would have to be of infinite length—quite an engineering feat! But, because of the rate of change of entropy of the gases in coming out and the fact that the particles are discrete sizes and have certain energies, for all practical purposes all the energy of the expanding gases at the nozzle would be used if the nozzle were 20 or 30 times the length of the rocket chamber. This, of course, is not feasible.

There are limitations to the nozzle. The difficulty in designing a very efficient nozzle is that as the nozzle becomes too long its weight increases and the rocket is far more sensitive to weight than it is to the efficiency of chemical conversion, also known as specific impulse. A 10 to 20 per cent change in the chemical efficiency of the rocket engine produces a relatively insignificant change in velocity or altitude, but a tremendous change can be made by simply hacking half the nozzle off. The designer, therefore, has to compromise. He says to himself: "I am converting chemical energy into a thrust in the nozzle. I am doing it at a certain rate. Is it more advantageous to use a lightweight chamber and only convert a portion of it and save that weight to put in something else, or not use it at all?"

Usually the answer is in favor of a shorter nozzle. Another consideration in nozzle design is that rocket engines have a steering function on the large rocket vehicles. The engine may be gimballed, as it is in Vanguard or the other big rockets, and if it has to steer back and forth, up or down, a large nozzle flopping up and down puts a great strain on the rocket chamber. Big nozzles, then, are to be avoided if at all possible, where weight is important.

The propellant has come down into the rocket chamber to produce this effect of high pressure combustion and exhaustion through a filter where the gas moves at sonic speed, provided this internal pressure is some definite fraction above the external pressure.

THE FUELING PROCESS

Pressure Method

Let us go back and see how the propellant reached the rocket chamber. The simplest thing is to force the propellant by pressure. This can be done by including a pressure tank filled with helium, which seems expensive and wasteful, but it is done. Believe it or not, in a large rocket it is often useful to substitute helium for nitrogen. You wouldn't think gases weigh much, but when you have a large sphere 6 feet in diameter at 5000 psi. the difference between nitrogen and helium may mean quite a few pounds. Also, helium is not easily absorbed into the liquids under pressure.

By releasing helium gas, the fuel tank in the vertical rocket can be pressurized. The pressurized helium acts like a piston on one end. (In fact, tanks have been designed with pistons in them, to take the place of helium or gases.) The liquid is thus forced into the rocket chamber, but the force of the push has to be higher than the internal pressure, otherwise the propellants won't enter the chamber.

The pressure required for forcing the liquids into the rocket chamber is so high that the tanks begin to look like battleships, with big heavy walls. So, it is necessary to find some other stratagem for putting the liquids into the rocket chamber. Pressure can be used to feed the propellants into very small rockets such as the Aerobee Hi, a high altitude rocket used by the Naval Research Laboratory.

If there is not too much pressure drop in the lines, there will be a flow of liquid into the rocket chamber. This flow is self-compensating, self-controlled, because the throat acts as a little orifice. The sonic flow in the chamber will allow only so much gas to flow at a given time. If you try to put too much propellant in the chamber, the chamber pressure builds up too rapidly and the rocket chamber tries to go back to an equilibrium. The rocket chamber has to be designed for a certain given amount of propellant flow per time, with a certain area relationship in the throat.

It is not possible to open up the throat or do much with the nozzle without changing the critical balance in the chamber, that is, the balance of the internal pressure to the outside pressure.

The throat is just like a little valve which can be opened by hand. If you set it at one point you must leave it that way, since if you tinker with the throat, the whole thermodynamic relationship changes. When that happens, the rate of injection of the propellants has to be changed, also.

With the pressure method of injection, in big rockets, the tank walls become so heavy that the rocket will never get off the ground. Tank walls are so thin in some of these big rockets that they will collapse in on themselves. Some of these giant intercontinental rockets, for instance, are like pressurized blimps.

Pumping Method

How, then, do we get the propellant from the tanks into the chamber? We use a very, very old device, namely, a pump. There are all kinds of pumps which can be used, keeping in mind that a driving device must be used which is independent of the atmosphere. This is done in some rockets by setting up a system with hydrogen peroxide inside of it.

Hydrogen peroxide has the nice property of not having to "burn" with anything. It is able to decompose without having the benefit of reaction with one of the hydrocarbons or the amines. This is done—in all the big rockets—by feeding the hydrogen peroxide out of a little tank into a decomposer, with a tiny rocket chamber having a small screen of silver coated with some chemical (or a mesh filled with calcium permanganate).

The peroxide tank is pressurized from the helium sphere mentioned previously. This introduces a secondary combustion process, for the peroxide will decompose when it hits the screen and form super-heated steam with a little free oxygen in it, at very high pressure. The gases are choked down again to give a high velocity jet of gas, which impinges on the blades of a gas turbine on a shaft and turns them like a little windmill. It is all enclosed. The gas turbine in turn is hooked up to two centrifugal pumps, for each propellant has to be pumped, of course.

In short, a separate external process pressurizes or drives a gas turbine (just as is used in a jet engine) at very high speed up to 30,000 or 40,000 rpm. The gas turbine then turns the pumps to pump the propellants.

That is the major scheme as used in rockets today. It is a source of much pain and agony, since it is very hard to design good propellant pumps and good gas turbines. In fact, it is harder to do than to design the chamber. Using basic principles, one can design a rocket chamber in half an hour. The fun begins when you try to design the injector at the end of the rocket chamber and the pump. The pump design is difficult and the injector design is empirical, which is disconcerting in a scientific world.

Now, we've gotten the fuel into the chamber through the pump, instead of using pressure. The pump has introduced an artificial driving force which will raise the energy level of the liquids to the requisite level to pump it into the chamber so it can burn and be all atomized into a fine mist. The oxidizer has been pumped in, also. The two pumps have probably been geared together and driven by a common gas turbine.

COOLING THE ENGINE

The oxidizer or the other liquid can be used in an interesting way. Before it is pumped into the rocket chamber, it can be pushed through the nozzle of the chamber, fed back along the walls of the chamber and back into the rocket engine. This cools the engine.

All ordinary rocket engines with liquid propellants need to be cooled, for an uncooled rocket engine is useless—with one exception which I will describe later. I want to give you some idea of why cooling is necessary.

The usual combustion temperature at 500 psi. is between 5000 and 6000° F. for an ordinary propellant combination such as kerosene and liquid oxygen. For fluorine and ammonia it is not much higher, oddly enough, but we obtain higher efficiencies by having less to accelerate. In the peroxide decomposer the temperature may be about 1200° F., if it is that high, so it is much less efficient.

A chamber with a 5000° gas environment inside and operating at a high pressure poses severe problems of heat transfer. The heat must be liberated. A lot goes out through the nozzle, but there is still heat flow into the walls.

Most rocket chambers today are made of some material such as stainless steel or mild steel. Stainless steel is the favorite because it is so resistant to chemical erosion, which occurs in a high degree because oxygen at high temperature, for instance, acts as a very fine cutting torch.

The chamber wall is fairly strong, say an eighth of an inch thick. On one side there is a 5000° F. temperature; on the other side, there is the ordinary atmospheric temperature, say, 70° F. In fact, in vacuum, there is no temperature. Heat can't even get away, for the only way to liberate heat in vacuum is to radiate it, and that is a very poor way to transfer heat.

The wall, then, must be kept cool. One of the big steps forward in the last twenty years in rocket design was a method of regeneratively cooling the chamber—regeneratively meaning here that one of the working liquids is used as its own coolant, so to speak. It goes along the walls and is re-injected.

What limits will the heat flow cause here besides any coolant we may introduce? Suppose we didn't have this coolant, what would happen? Three or four seconds after the rocket engine started, several things would occur, the major one being disintegration of the chamber. But before that, the walls would begin to erode away.

One reason for this is that in a liquid rocket engine, the atmosphere usually has a surplus of oxygen which erodes the walls. When the walls reach a temperature near their melting points, the metal actually burns off. The metal can be protected with a ceramic layer on the inside. This is one exception to the need for cooling—a rocket chamber with a very thick ceramic coating. The ceramic gradually erodes, but if the firing is for thirty seconds or so, the chamber can survive.

Without a ceramic lining, we must do something to protect the wall.

The inside of the wall already has a certain self-protection in the form of a stagnant film of gas, a boundary layer, which is a very severe attenuator of the heat flow. It is a quiescent layer of gas which does not have much turbulence in it and which severely limits the convective heat transfer that occurs inside the chamber.

This gas film is a natural protector and even if we put a cooling jacket around the outside, just as a car engine has, this gas film must still be there. If in a rocket chamber the gas film is destroyed by oscillation or turbulences, in many cases the chamber will burn out whether it is cooled or not. This, unfortunately, occurs fairly frequently.

The wall, then, must be designed so that it is always cool enough to maintain a boundary layer of cool quiescent gas along the wall.

Rocket walls must not be too thick, because the thinner the wall, the better it is from the heat transfer picture. The designer always makes the rocket wall as thin as possible, consistent with the internal pressure and the structural strength of the rocket chamber.

Regenerative cooling allows the wall to be kept cool enough so it doesn't burn or doesn't evaporate. This is the thing that allows the rocket engine to be a practical reality. Without it we wouldn't have anything unless we used enormously thick walls, tremendous ceramic shells or, as I will tell you later, solid rockets.

One of the most commonly used oxidizers, nitric acid, is an excellent coolant. In Vanguard in the first stage we used the fuel (kerosene) as a coolant, because liquid oxygen can't be used; in the second stage we used the nitric acid (the oxidizer) for the coolant, because the second-stage fuel is not too good a coolant, and, more important than that, we don't have enough of it. It is not only necessary that the coolant flow through at a certain velocity (it must have a high velocity), but it is also necessary to have a good quantity of it in order to carry away the heat. So regenerative cooling of a rocket chamber is a prime and necessary element of the rocket engine complex.

INJECTION PROCESS

The next step is injection. The injector is a small element of the rocket engine with little holes in a face plate (see Fig. 2). The liquid comes up behind it and it looks just like a shower spray. In fact, some injectors are called shower-heads. The liquid is sprayed out and is mixed in fine droplets. There is a vast body of theory on how we should mix it and what the atomization rates should be, but no one has yet worked out a good theory for the rocket designer. The fact is that there is nothing really scientific about injector design. We inject tiny droplets of liquid; they all mix, making a tremendous turbulence of gas vapor which creates a great fire. This represents a low order combustion process. All these tiny vaporized droplets of propellant try to meet, mix together and burn. They must be atomized to some extent, otherwise it takes too long for the combustion process to be completed. The more combustion occurring near the injector head, the more efficiently we use the volume of the chamber.

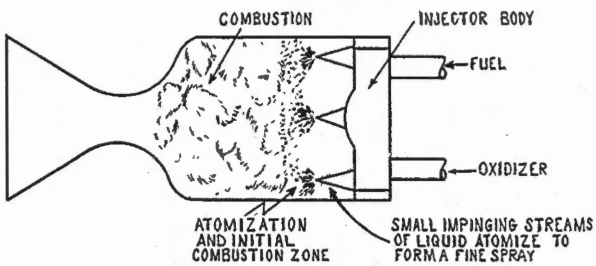

FIG. 2.

There are several types of injectors. One type has two holes with a liquid spraying out each one. The two sprays impinge forming a little flame; this process can be repeated all over the face of the injector. Another type is a swirl nozzle which makes a great big vaporized spray, very much like a home oil burner nozzle.

Cooling of a rocket chamber is most important and without it we could not exist with rocket engines; however, the injector is the heart of the system. The rest of the components can vary considerably—for example, the nozzle can be cut off or twisted around. In fact, any variation beyond the injector doesn't bother the chamber at all. Once the gas flows sonically through the throat, what happens beyond the injector has no effect on the inside. The sonic flow is a complete and definite cutoff from the outside world for the inside chamber.

SOLID ROCKETS

Although I have said a good deal about liquid rockets, don't let me give you the impression that liquid rockets are the *raison d'etre* of space flight. There are other things. I mentioned liquids because they are the more complicated, and because they still have considerably greater application than the others. Most of the big rockets which are fired from the various missile ranges are liquid rockets. We have yet to learn how to cast and mold and squeeze out the propellants for large solid rockets.

Generally, the rocket engine in the liquid rocket comprises the chamber and the pumps and the valves that go with it, and there may be hundreds and hundreds of valves. On the other hand, the solid rocket is disgusting in its simplicity. It is nothing but a solid charge. To make it, an ordinary metal or Fiberglas case is filled through a spigot at one end with some solid propellant material which could be several things. Solids are not usually as energetic as liquids, but they can be.

Types of Solid Propellants

There are two main divisions of solid propellants. One of them, fairly widely used today, is the so-called double-base propellant which is the traditional propellant made of explosive nitrocellulose or nitroglycerine tempered with additives to decrease the burning rate. Nitroglycerine would make a nice mono-propellant (a propellant which burns all by itself without adding oxygen), but it burns so fast that it makes a nice bomb, too. You could not spray nitroglycerine into a rocket chamber and expect the thing to stay put, because the explosion wave would travel up the pipes and into the tank

Another type of solid propellant are the synthetic rubber propellants, usually called composite propellants. They are made of synthetic rubbers with some oxidizing material such as ammonium nitrate or perchlorate mixed with them. These two combinations are widely used. I think the emphasis presently is on the composites.

Specific Impulse

The specific impulse of these propellants, that is, the rate of usage, is not as high as that of liquids. "Specific impulse" is a very treacherous term to use, by the way, even though it is used widely. We define specific impulse as the thrust of the rocket divided by the propellant weight that is burned times the gravitational constant.

$$\text{Specific impulse} = \frac{\text{Thrust}}{\text{lb/sec propellant}} \times g$$

This is an important factor and does tell you that for a given amount of thrust and for a given propellant burning per second, you get a certain amount of thrust, or vice versa.

Another term that has been used widely in rocket practice is "exhaust velocity." Exhaust velocity is the velocity of the gas coming out of the rocket; the higher that is, the more efficient the rocket vehicle is and the higher the ultimate velocity of the rocket vehicle will be.

If the exhaust velocity is divided by the gravitational constant, the answer is the same thing as impulse. The specific impulse definition is somewhat anomalous because it depends on thrust, and thrust is a hard thing to measure in a rocket engine. Of all measurements to make, thrust is the most difficult because the whole large rocket engine and all its parts must be made to push against a spring scale or hydraulic or electric indicator of some kind. We can set the unit on a big pivoting test stand; but all the pipes that go with it and all the valves have to move with it, also. If they don't move with it, you have to account for the flexure in the lines. The whole complex thing has to be calibrated, and so it is very difficult to measure thrust.

Thrust is also a function of the size of the nozzle. The bigger the nozzle and the higher the altitude of the rocket in vacuum, the more thrust is developed for a given pound of propellants burned. So thrust is a variable thing in a rocket engine.

Our Vanguard rocket engine starts at 28,000 pounds at sea level and probably goes to about 29,000, although our nozzle isn't really long enough to make full use of that thrust capability. Our second stage engine has an 8000-lb. thrust in vacuum, but only about 5000-lb. thrust at sea level because the nozzle is not being used fully. When the gases of our second stage engine are burned at sea level, the gases begin to separate and break away from the wall. They expand to atmospheric pressure, 15 psi., so the big nozzle does no good at all at sea level. We call this an over-expanded nozzle when the nozzle is a bigger size than can be used fully by the gases.

Combustion Efficiency

A more interesting figure is the so-called combustion efficiency or characteristic exhaust velocity. It is a function of the throat area of the rocket engine and the weight of propellant. It is a more complicated relationship. Combustion efficiency is a more fundamental parameter which describes how we burn the propellants. This is more of an application factor, such as horsepower on an ordinary motor car instead of giving the pressure-volume relationships of the piston or the displacement of the engine or, better yet, the entropy of the gasoline or even the combustion temperature in the cylinders.

Design of Solid Propellant Rockets

The solid rockets are masterpieces of simplicity by their very nature. Suppose we take a large shell and we pour in a liquid which becomes a solid. What the designer usually does is to cast the solid propellant so that it burns evenly on the inside. He does this by using a trick the Chinese used two thousand years ago—he puts a star pattern on the inside of some kind. Then he puts an igniter down the center of the rocket, a long igniter which ignites and sprays out flame at the "start." The propellant then burns outwards, with a pressure rise up to some terminal value which may be 1000 psi. The pressure then stays at this level until decay at the end, when all the solid propellants burn out. Without this leveling off of the pressure curve, the rocket engine would pulse at a tremendous rate and tear itself to pieces. It is not efficient to have a high thrust. It is better to taper out the thrust.

You can design a solid charge with special layers to actually taper the thrust outwards. The pressure and thrust curves would be similar. The thrust follows the pressure very closely. The thrust, in fact, is very dependent on the chamber pressure. So, you can program this curve by the proper design of the charges. It is not always easy to succeed in this. The solid charge, once it is designed, is a very reliable device. It will usually ignite within an ignition period of only a few milliseconds. The pressure then rises, stays up and then comes down.

The solid propellant reaction is a complicated reaction. It is an oxidation between complicated molecules, with the burning adjusted so that the exhaust gases are not rich in oxygen at all—they are a reducing flame which is a lot easier on the throat.

LIQUID *VERSUS* SOLID PROPELLANTS

Because the solid propellant eliminates all the valves and pumps required for a liquid rocket engine, this weight reduction is useful in medium-size rocket vehicles. Very small rockets can use either liquids or solids, although when they are only a few feet high they are usually best made of solids. Then comes an intermediate range where liquids are good. Then comes another intermediate range where solids are better. The bigger rockets, such as IRBM and ICBM, use the liquid propellants.

The difficulty with the solid charge is that for a very large rocket the grain begins to flex and bend. It is very hard to pour and, once it has been poured, it has to be put in an oven and baked. There is nothing quite so precise as baking a solid rocket! At any rate, the solids have to be baked or cooked or cured under very precise conditions of temperature, humidity, etc. This is very difficult to do for very large rockets because as the solid mass gradually cools down and releases a lot of heat, it may develop cracks. At present, therefore, it is very difficult to make a big solid, but the techniques are being learned.

I mentioned that the chamber pressure in a solid may be 1000 psi., which is higher than most liquids have. It doesn't have to be that; indeed, in vacuum the chamber pressure can be anything you want as long as you have sonic flow through the throat and the proper pressure ratio relationship. But at sea level, the chamber pressure should be high for a given thrust. Also, a solid needs a certain high chamber pressure for maintaining its burning rate. If the chamber pressure is too low, the fire may be unstable and may die out. There is a critical pressure region which is very important.

Presently, efforts are being made to reduce the chamber pressure in solid rockets and also reduce the case weight, because all the pressure and the heat of the fire must be contained in the case. Although the solid propellant is self-healing (as the fire burns, the solid propellant material insulates itself), there is a limit to this. It gradually breaks down and fissures may be established. When fire reaches the walls it burns through, in which case it is necessary to insulate; this raises the weight of the rocket.

If the chamber pressure is a thousand pounds, it is seen that even though the propellant contributes something to the strength (not much), the case must take that whole 1000-psi. pressure. But with modern techniques, it is now possible to improve the mass ratio to the point where one can get mass ratios of 0.9 for solid rockets, which is the weight of the propellant divided by the weight of the whole vehicle. This mass ratio is a very important factor.

The best mass ratio that I know of for a large liquid rocket gives about 0.85. This factor is of extreme importance. It tells you how much of the rocket is available for energy usage. The ideal mass ratio is 1. A little reflection will show you that if you have that in the case of a liquid, you have just a big "popsicle" of propellant because there would be no skin—nothing—no structure weight. It would all be propellant. This is the ideal rocket, but it still is not possible. If we had a rocket of 0.97 mass ratio (97 per cent of the propellant weight available for energy), you could reach the moon in one stage with any given payload. This, then, is a very powerful factor in the ballistic equation.

I don't want to go into ballistics here, but I just want to tell you that a solid rocket will have about 0.97 mass ratio (or it can be produced), while with liquid propellants it is almost impossible to better the mass ratio of 0.85 with the propellants we now have.

Since burning in the solid rockets progresses outwards, it is self-healing and no cooling is required. The throat is the hottest part of the rocket and is very critical in either a liquid or solid rocket, because the gases impinge on it, exchange momentum with the walls and give up energy, so we have total temperatures of rather high levels.

Thus, the solid rocket now has moved in on the liquid field because of the simplicity. We can also change the exhaust gases so they don't oxidize the walls. We can have a reducing gas. We can use molybdenum metal throats which are easily oxidized, but not at all reduced. We can use carbon and we could build solid rockets which will burn from a lower level of nothing at all to up to a burning time of 60 or 70 seconds, which was unheard of before; this means that now we have an uncooled rocket engine or rocket chamber which can last for 70 seconds.

That contradicts a bit what I said before—which was that one couldn't have uncooled chambers. In a solid, because the solid material helps to insulate, this can be done. The solid propellant now acts as an insulator while the liquid does not have that advantage, although in some cases liquids can be sprayed along the inside of the chamber thus forming a liquid film along the wall; but that is a poorly understood technique and not widely used.

The question now is: Where do we go with the solids? Do they have any meaning in this picture? What is the point in discussing them? I say they do have meaning, indeed. As you know, our third stage Vanguard is a solid rocket. The reason it is used is that it is easy to ignite in vacuum; it has a closure in the front so that air stays trapped on the inside; it is ignited, the fire comes out and once the fire is started the solid will keep on burning. A liquid can be ignited this way, too, but a solid is a more reliable device.

The Explorer II, the Army satellite, did not ignite its fourth stage, which was a solid rocket. Usually when the electrical signal gets into a solid rocket and starts igniting, the solids will burn regardless of what is outside—vacuum or no. It is a reliable device, since there are no valves to start and stop it.

Another interesting problem is what can be done to program the thrust of a rocket engine. In a solid, one can do very little, for once it burns, it stays at a constant thrust.

In a liquid rocket engine, if one wants to vary the thrust, several things must be done. It is possible to change the throat and adjust the thrust that way, by adjusting the chamber pressure. One way to do that is to put a little egg-like body into the throat and move it in and out. In this way the throat area changes without physically changing the throat itself. This can and has been done in a liquid rocket. It is an easy thing to talk about, but almost impossible to do because the "egg" burns up quickly.

One can change the flow rate in a liquid rocket. The trouble is that as one changes the chamber pressure in a liquid rocket, the stability of the chamber changes and becomes violently unstable and does not want to burn any more. The fire goes out, or at least one wishes it had gone out!

So, little can be done to change the thrust. However, a liquid engine can at least be shut off by shutting any one of the valves; this is impossible in a solid rocket, for once it is lit, it burns to the end.

It is possible to stick a pipe with a valve in the side of the chamber of a solid rocket and as the solid burns, one can suddenly open this valve or blow a hole in the side of the chamber. All the gas will vent out the side and the chamber pressure will drop below a critical value and the fire will go out that way. This is possible. But to program the thrust level of a liquid or solid rocket engine, or even to stop a solid when one *wants* to stop it is very difficult indeed and, again, very little used. (The British have some throttling liquid rocket engines used in rocket aircraft.)

Today we can say that liquid rockets are mostly used for very large intercontinental missiles, certain high altitude rockets such as Vanguard, and rocket aircraft. The solid, despite its simplicity, lacks controllability and you would not want to use a solid rocket in a rocket aircraft such as the X-15 or the former X-2, because when the pilot pushes his button, he is away whether or not he wants to be—a very uncertain situation for him. Therefore, liquid propellants are used; they may be a little more dangerous and harder to build, but at any rate, the pilot can control this liquid rocket —he can shut it off or vary the thrust to some extent by having a number of chambers or adjusting the injector flow within small limits.

The solid rocket has come along quite rapidly since the war. It is occupying presently about 30 per cent of the propulsion spectrum in terms of total energy used, and in the near future may occupy as much as 50 per cent. There has been a lot of talk in the newspapers about anti-ballistic missiles: these could be built using solid propellant engines because a solid rocket can be stored any place and it is only necessary to push a button to fire it. There is some truth in this talk, although it would have to be stored at certain temperatures and certain environmental conditions because the solids are temperamental this way.

SUMMARY

I want to reiterate what have been the major accomplishments in the propulsion principle in the last few years—the main things that have changed the propulsion picture from the exciting times of the early rocketeers such as Goddard.

The first big step forward in rocket engines was the controlled burning of liquids through the use of valves. The second step was the discovery of new liquid propellants such as liquid oxygen and amines. The third great break-through was lightweight chambers, manifest in several ways: one is the ceramic chamber; another is the self-cooled or regeneratively cooled chamber; another is the use of light alloys of aluminum and even magnesium.

Other advances include: the use of pumps (centrifugal pumps have made an enormous difference); a better understanding of nozzle theory to give us altitude performance as we want it; new injector design; the use of throttling in rocket engines (still little used, but coming along); the use of solid propellants; and the use of thrust vector control which allows one to use the rocket engine itself by gimballing the chamber and using it for a steering device (this cannot be done easily in a solid rocket).

There is also a new understanding of the new fundamentals of rocket engines. Some work has been done; hundreds of theses have been written; and millions of dollars have been spent in an attempt to understand such things as combustion process, temperature reactions, etc.

Mainly, the use and understanding of the nozzle theory, the use of pumps and good cooling processes and an understanding of the heat transfer process have been big steps forward, as has been the evaluation of the propellants. These have been the major advances which will probably lead us to space flight.

ROCKET FUELS—LIQUID AND SOLID

BY

H. W. RITCHEY[1]

It is interesting and stimulating to talk about man in space—what happens to him in the absence of gravity and how he dodges the meteors; but behind it all we still have that old problem of how to get him up there.

I happen to be one of the fortunate or unfortunate people faced with the problem of making rockets that work, and my lecture tonight is probably the most unglamorous one of the series; the subect is rocket fuels. It represents the real spade and shovel work of the space travel field.

Making rockets to get a man into space by conventional methods is not an easy task, but I hope to point out, however, how easy it might be by the exercise of human ingenuity, involving new and unconventional methods for accomplishing the task.

In talking of rocket fuels, I have some spadework to do first to set the record straight. When you drive up to a filling station and say "fill up my tank with gasoline" you probably think you are getting fuel; however, you are only getting about a tenth of the total fuel you burn—the free portion is air. A rocket is not fortunate enough to have free air available.

WHAT FUEL IS

I would like to start off by defining what we mean by fuel in the field of rocketry. When you drive up to a gasoline station, you fill your tank with hydrocarbons. The first line of Table I shows that the heat of combustion of an ordinary paraffin hydrocarbon is something over 18,000 Btu's per pound of fuel.

TABLE I.—*Heat of Combustion.*

Paraffin Hydrocarbon (Gasoline)	18,600 Btu/lb
Paraffin Hydrocarbon + Oxygen	4,200 Btu/lb
1 part Hydrocarbon + 3.43 parts Oxygen → 4.43 parts Combustion products	

This is the way we measure the energy content of the fuel that you put in your gasoline tank. What you buy when you buy a super-grade fuel (and this is very important and we will draw on it even more in talking about rocketry) is not extra Btu's per pound; instead, you are buying something that can be burned in your engine to give you more energy output—in other words, the realization of more of those Btu's as you drive down the highway.

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How about the other component of this fuel? In a rocket engine we are not talking only about the hydrocarbon component, which is called the reducing component of the fuel or the reductant. In rocketry, an oxidizing component must be added to the reductant. If this had to be done in ordinary automobile fuel, the heating value per pound of total mixture would be reduced to about 4200 Btu's per pound, or thereabouts In other words, the energy content per pound goes down terrifically when the mixture includes both the reducing component and the oxidizing component. This is an extremely important factor in the field of rocketry because, after all, we are flying in outer space where there is no oxygen and we must carry both of these components along with us.

How does the ratio of these two components compare weight-wise for the paraffin hydrocarbon? One part by weight of paraffin hydrocarbon and 3.43 parts of weight of oxygen are necessary to give the most efficient combustion mixture, and, of course, this results in about 4.5 parts by weight of combustion products. In other words, you would have to have a gasoline tank four or five times larger than the one you have now to get an equivalent amount of mileage if you had to buy both components.

In talking of a rocket fuel, then, I must cover both of these components and the term "fuel" covers both the reducing component or reductant, and the oxidizing component or oxidizer.

I want you to remember this figure of 4200 Btu's per pound while we consider the problem of providing energy to get us into space. What kind of energy conversions are we talking about? What do we need to get outside the Earth's gravitational field?

For the sake of aiding your imagination, Fig. 1 is a drawing of what I call the Earth's gravitational hill. The central circle is supposed to represent the Earth, and starting up from this Earth there is a hill that looks like the bell of a trumpet. For every radius of the Earth that we travel up the hill, the hill becomes half as steep. The hill extends on out for an indefinite distance, for all practical purposes, getting less and less steep as it goes, but it never stops completely, at least not until we run into a much steeper hill belonging to the sun or some other astral body.

Although this hill never stops, we can calculate what would happen if somebody started a marble rolling at the top of the hill and that marble was allowed to roll down the hill until it struck the Earth's surface. That marble would have a certain amount of energy. That amount of energy is represented by its velocity and that velocity, without the effect of friction or any other adverse or confusing effects, is about 37,000 feet per second. This represents the energy that a body would have rolling from an infinite distance or from far, far away, down this hill and striking the surface of the Earth.

We can imagine sending that marble back up to where it originally was. Surrounding the Earth's surface and surrounding us, of course, there is a relatively thin layer of very, very thin material that we know as the atmosphere. It acts like sorghum molasses to the type of vehicle we are speaking of now, when it hits at that speed. And, except for that sorghum molasses, if we started this body rolling back up the hill at 37,000 feet per second, it would roll completely back up the hill and end up eventually where it originally started.

This is what we mean by escape velocity: it is a measure of the energy content of a body that can leave the Earth's surface and go on out against the Earth's gravitational hill coasting up and up and up forever and ever until it is eventually lost in outer space.

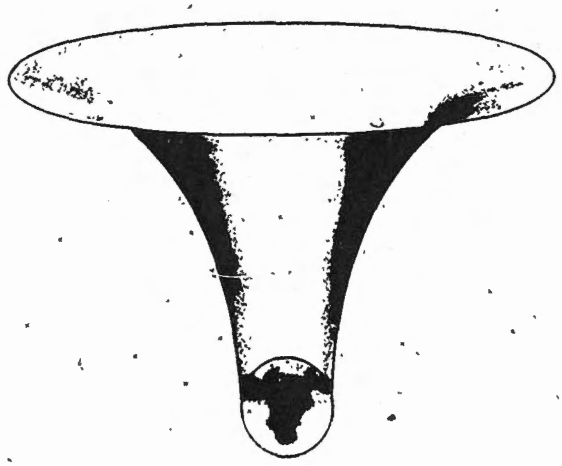


FIG. 1. The Earth's gravitational hill.

It is true that if we crawl up this hill a little at a time we never have to achieve quite this 37,000 feet per second, but that doesn't help any in reducing the total energy needed for escape. Therefore, to get this body back up into space, we have to put into it the same amount of energy it had when it came down, which is represented by the velocity of 37,000 feet per second. Expressing this kinetic energy in terms of heat, this comes out to about 27,500 Btu's per pound of object, which is about six times the energy contained in the mixture of paraffin hydrocarbon or gasoline and oxygen listed in Table I.

In other words, if we could convert all the heat energy in a pound of gasoline into mechanical energy (which, of course, can't be done because according to the second law of thermodynamics only about a third of the heat energy can be converted) and if we had a sky hook up there in space with a cable over it and a winch, that one pound of gasoline we burn on the Earth's surface has the energy content to lift about a sixth of a pound of mass out into outer space outside the Earth's gravitational field.

This is the problem we have facing us.

PROPULSION PRINCIPLE

The hard core of this situation is the question of what is involved in burning this fuel in order to get an object up into outer space. The hill in Fig. 1 is strictly a fictitious one because there is no surface, slippery or otherwise, that we can use to get traction. Whatever object we use must be able to propel itself by something other than frictional contact with the surface; and this, of course, is where the rocket engine comes in. It must carry its own fuel, both the oxidizing and reducing components, and it must not be driven by wheels, by cables or chains or pulleys. It must be driven by some special force which is different from those we normally encounter in everyday life.

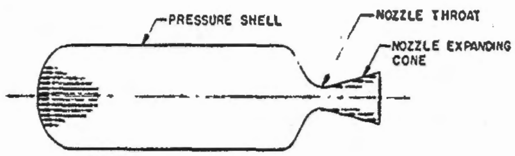


Fig. 2. Diagram of a rocket engine combustion chamber.

We have to rely on the third law of Newton in order to propel an object into outer space. If it weren't for this law, we could count on six pounds of fuel to carry a pound of payload to escape velocity or outside the Earth's gravitational field; but things are not quite that simple!

Figure 2 is an outline of a rocket engine combustion chamber. Inside, gas is acting under pressure and pressing out against all the walls. You can imagine for just a moment a plug in the nozzle throat, separating the combustion chamber from the nozzle expansion cone. With such a plug, all the pressure forces in the chamber would be equal in every direction against the walls, and this object would go nowhere at all.

The moment we take the plug out of that hole, the situation changes. Most of the remaining unbalanced forces of pressure are acting on the forward end of the combustion chamber. The gas on the inside, which is pressing against the forward end of the combustion chamber of the rocket, is not quite all expanded when the plug is taken out. Also, some additional forces are reacting on the nozzle expansion cone.

Since the gas in the rocket presses on the rocket itself, this is the means by which it propels itself in outer space. There is only one catch—there has to be a hole or there are no unbalanced pressure forces; and, because we have that hole, the gas gets out of the rocket and keeps on escaping until the pressure goes down to zero, when it quits pushing against itself. That is where the trouble comes in.

We can calculate how much stress acts on the rocket as follows:

$$F = \int_s P \sin\theta \, dA = C\dot{m} \quad (1)$$

which says that the thrust force, F, is equal to the integral of all the pressure forces, P, around all the surfaces with appropriate correction for direction. (θ is the angle between surface increment and horizontal axis.) F is also equal to the velocity at which the jet comes out of the nozzle times the mass rate of flow in slugs per second or pounds per second divided by 32.2. In other words, the thrust force is equal to effective exhaust velocity, C, times the mass rate of flow of gas through the nozzle opening, $\dot{m}$.

Our problem, then, is to keep the rocket chamber full of gas just as long as we possibly can.

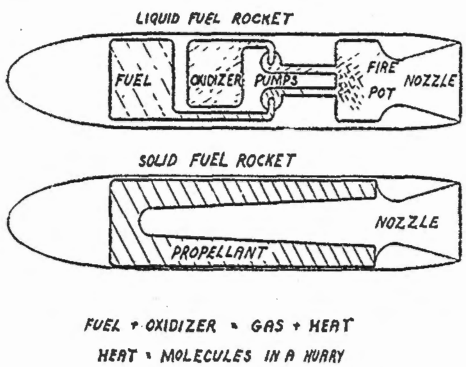


FIG. 3.

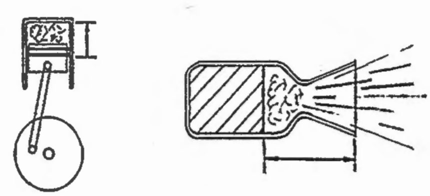
LIQUID AND SOLID PROPELLANTS

There are two ways of attacking the problem of keeping fuel in the chamber: (1) using solid propellants and (2) using liquid propellants. The top diagram of Fig. 3 shows a liquid propellant rocket, with its tanks of reductant and oxidant. The fuel is pumped into the combustion chamber where it burns under pressure and the pressure forces act against the combustion chamber and the gas escapes out through the nozzle. As long as we don't run out of fuel and as long as the pumps and other apparatus work according to design and calculation, we will have a thrust force and the rocket will continue to accelerate out into outer space.

One advantage of this type of rocket is that the fuel can be packaged in low pressure tanks, since the tanks only have to stand the hydrostatic pressure of the fuel itself. The pumps develop the pressure to get the fuel into the combustion chamber. The system, however, is complicated because of the pumps and valves and mechanical drives used to drive them. It is a hard system to start and, if you have read the newspapers to date, you might conclude that it is notably unreliable.

The solid propellant system (shown in the lower diagram of Fig. 3) involves a charge, which ordinarily burns from the inside towards the out-

ROCKET IS A HEAT ENGINE



GAS EXPANDS PUSHES PISTON

GAS EXPANDS PUSHES ITSELF

WORK = MASS x ACCELERATION x DISTANCE

FIG. 4. The rocket as a heat engine.

side, generating hot gas which escapes out through the nozzle. The solid propellant is already in the combustion chamber, prepackaged, so there is no trouble with pumps and injecting mechanisms. It is a very simple device ready to go on an instant's notice, but it has one disadvantage—the entire fuel charge must be packaged within the combustion chamber which involves the addition of certain inert weight components. That is a very great disadvantage for certain space applications, or at least theoretically it is a very great disadvantage.

Figure 4 points out that a rocket is a heat engine, and, like all other heat engines, it converts the energy of a chemical combustion process by the expansion of hot gas into mechanical energy in one form or another. In the piston engine (shown on the left) this pressure force presses down against the piston producing rotation of the shaft and power is taken out as shaft energy. The artist has put this in very simple terms—he says that in a rocket the gas expands and pushes itself, and the work equals mass times acceleration times distance. Mass times acceleration is equal to force, and it can be force on the piston or it can be force on the gas element itself coming out of the gas jet.

To compare a rocket engine with an ordinary internal combustion engine, we find that power is taken out as shaft power in the ordinary engine, while in the rocket, the power comes out as kinetic energy of flow of the exhaust jet. Like all other engines, the fuel efficiency of the rocket engine depends upon its expansion ratio. The same equations are used to determine the expansion ratio in the internal combustion engine and in the rocket engine.

ESCAPE VELOCITY

If we look at what is required to make a good rocket engine for space travel, we get back once more to the Earth's gravitational hill. To send a missile into outer space, enough stages of rocket must be stacked one on top of another to add up to something over 37,000 feet per second total velocity departing from the Earth's surface.

How much velocity do we get from a stage of rocket propulsion? Without the effect of a gravitational field and without the effect of the Earth's atmosphere, the atmospheric drag, this is a very easy thing to calculate, using the following equation:

$$V_b = g\, I_{sp} \log_e \left(\frac{M_1}{M_2}\right), \qquad (2)$$

where

g = gravitational constant 32.2 ft./sec.^{2}
I_{sp} = propellant specific impulse, lb.-sec./lb.
M_1 = initial mass of missile
M_2 = mass of missile at burnout.

Equation 2 says that the velocity obtained from burning of a stage of rocketry, in feet per second, is equal to acceleration of gravity times the propellant's specific impulse, which is related to the Btu content in a rather secondary way, times the log to the base e of the ratio of the initial mass of the object divided by the mass after the fuel is burned out. This equation applies to all moving objects developing their force by the reaction type of propulsion force, where it is necessary to squirt matter in a backwards direction.

There are only two ways of increasing the velocity, V. One of them is to increase the specific impulse of the propellant, and the other is to increase the ratio M_1/M_2, which is the initial mass of the object divided by the mass of the object after the fuel is burned. I shall discuss increasing the specific impulse later, because it is related primarily to fuel energy.

Increasing the Ratio M_1/M_2

In regard to the second method of increasing the velocity, there is only one way to increase the ratio M_1/M_2, and that is to make more of the object fuel; and the more of the rocket you make fuel, the more increment of velocity you get by the burning of that particular rocket stage. It is just too bad we can't make a rocket all fuel (there wouldn't be any place for anyone to ride), for this would make the problem of escape a lot easier.

The problem of increasing the ratio M_1/M_2 is related to a lot of other things including how much inert components you have to use in the engine itself and how much payload you are going to carry in the rocket missile system For instance: How light can you make those tanks that have to contain the fuel? How light can you make the pumps? How light can you make the nozzle? How high a ratio of fuel to inert components mass can you get in this rocket system?

This becomes a very important point because in order to get high energy, we must not only handle materials that are extremely corrosive, with flame temperatures that are above the melting point of any known materials, but we must contain those materials (which have a lot of very nasty properties in addition to being extremely corrosive) under pressure in some kind of physical vehicle to develop proper flow in the rocket system.

In Eq. 2, the velocity of a rocket was seen to be linear with specific impulse; in other words, a 1 per cent improvement in specific impulse gives 1 per cent improvement in velocity.

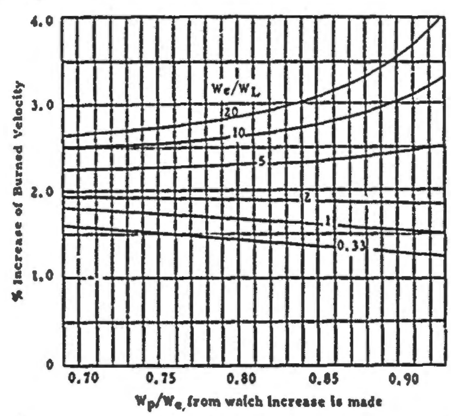


FIG. 5. Percentage increase of burned velocity for 1 per cent increase of W_p/W_e by reduction of inert component weight and addition of propellant to maintain a fixed total engine weight and engine-to-load ratio.

The set of curves in Fig. 5 shows the percentage increase in velocity obtained by increasing the ratio of fuel mass to engine mass, for varying engine-to-load ratios. Referring to the top curve in Fig. 5, you notice that a 1 per cent improvement in the mass ratio between fuel and inert components will give a 3 per cent improvement in velocity, where the 1 per cent improvement in specific impulse only gives 1 per cent improvement in velocity. So really we have almost as much to gain by improving the inert components of this rocket as we do by improving the fuel energy. On the other hand, the fuel energy can be improved at least to some extent beyond limits, but it is obvious that it would be quite difficult to make a rocket that is 100 per cent fuel because tanks and valves, etc., must be included in order to make the engine work.

Just to give you some idea of the meaning of this ratio between useful contents and inert components (comprising the packaging), I should like to mention some things that you encounter in everyday life and give you their M_1/M_2 ratios.

A can of beer is 81.4 per cent payload and the rest of it is inert components, the can itself. If we made rockets that poor we would never get any place; even when that rocket may have to carry pumps and valves and sustain pressures of 1000 psi. or better, we wouldn't think of making a rocket that was only 81 per cent useful contents and the remainder a packaging device.

The egg is a little better, with 89.6 per cent by weight of useful contents and the rest shell. We are getting up in the rocket range, but I think we still ought to do better than that if we are going to make a good rocket.

A candy bar's ratio is 95.6, which is really good. I am not going to tell you how well we are doing because that is classified information, but I will say that there are a lot of rockets flying today that aren't anywhere near as good as the candy bar.

A loaf of bread is better yet—it is 97.7. But don't forget that we have to sustain pressure and to sustain heat above the melting point of any known structural materials in order to make this rocket work, and being able to achieve something close to the packaging efficiency of a loaf of bread looks like a very difficult proposition.

Increasing Specific Impulse

Let us get back to the specific impulse idea now. Remember, a rocket is a heat engine and those of you who have studied thermodynamics know that you can convert heat or combustion into energy in the proper kind of engine. The energy obtained is expressed by

$$I_{sp} = \frac{c}{g} = \sqrt{\frac{2(H_c - H_e)}{g}}. \qquad (3)$$

This says that the specific impulse is equal to the effective exhaust velocity divided by the acceleration of gravity In terms of the energy contained in the fuel's components themselves, this is equal to the square root of 2 times the heat (enthalpy) in the combustion chamber, H_c, minus the heat at the end of the nozzle expansion cone, H_e

The heat in the combustion chamber minus the heat contained in the materials at the end of the expansion cone represents the amount of combustion enthalpy that is converted into flow work in the rocket. I want to make this point especially emphatic at this time, because later we are going to talk about atomic-powered rockets. If a working fluid that expands out through the nozzle is used, it doesn't make any difference where this enthalpy comes from, whether it is coming from the heat of an atomic pile or from the heat of a match, or whether it is formed by a chemical combustion process.

Equation 3 is based on the first and second laws of thermodynamics. The calculations are not extremely difficult, and if you are interested in how the calculations are made, refer to any standard book on thermodynamics. The same equation holds for steam turbines. In fact, the same principle holds for the internal combustion engine in your car.

For the rocket engine, we use the following equation:

$$I_{sp} = 9.8\sqrt{1 - \left(\frac{P_e}{P_c}\right)^{\gamma-1/\gamma}}\sqrt{\frac{\gamma}{\gamma - 1}}\sqrt{\frac{T_c}{M_c}}, \qquad (4)$$

where P_e is pressure at the end of the expansion cone. P_c is the pressure in the combustion chamber, γ is the specific heat ratio, T_c is combustion chamber temperature, and M_c is the molecular weight of the combustion products.

The expansion ratio (P_e/P_c) in Eq. 4 depends upon how the fuel is used —how far the combustion products are expanded, what the chamber pressure is, what the outside pressure is, what the specific heat ratio is, and whether the nozzle is optimum. These latter factors are controlled by the rocket designer and are not inherent in the fuel itself. The specific heat ratio, on the other hand, is inherent in the fuel, but the performance of the fuel is not too sensitive to it.

The really important factor in Eq. 4 is the ratio $\sqrt{T_c/M_c}$ which does not depend upon how the fuel obtains its heat or where the enthalpy comes from. These facts are as immutable as Newton's laws and the laws of thermodynamics.

For convenience in this rocket field the factors that are pertinent to the fuel characteristics themselves (for example, the combustion chamber temperature and the molecular weight of the combustion products) are designated C^* in Eq. 5, which is a slightly different equation for specific impulse.

$$I_{sp} = \frac{C^* C_F C_D}{g}. \qquad (5)$$

C^*, fuel energy, is usually measured in feet per second; C_F represents the nozzle thrust coefficient which varies with the expansion ratio; C_D is the discharge coefficient which is a correction factor; and g is the gravitational constant.

Let us examine Eq 5. Suppose we want to calculate the performance of a rocket fuel. C_F can be estimated from P_c/P_e in Eq. 4; C^* can be estimated if we know T_c/M_c. The gamma doesn't make too much difference because we can probably guess that. Thus if we know all of the products that are formed in combustion (which change with temperature) and if we know the temperature, then we can calculate the two terms C^* and C_F. This involves using a very, very complex set of equations that depend upon the knowledge of the high temperature thermochemistry of some extremely hot gases; obtaining experimental information on these gases is an extremely difficult matter. The constants themselves are known only with limited accuracy and in order to unravel what goes on in a complex mixture of these combustion products, it is sometimes necessary to solve fourteen or more simultaneous equations, many of which are of high order and require a tremendous amount of effort to decipher. Because of this, there is only a limited amount of accuracy in calculating these two terms.

The accuracy of Eq. 5 depends also upon knowing how these things behave in the expansion process, in which there is a temperature drop in the nozzle, the velocity of the flow is 7000 to 10,000 feet per second, and the nozzle is only a few feet long. In this temperature drop we have to convert some of these chemical components to achieve equilibrium—at least we hope to achieve it to some degree as the material goes down the nozzle and cools.

There are three ways in which molecules impart energy: (1) in the translational state, the molecules bounce back and forth; (2) in the rotational state, the molecules spin; and (3) in the vibrational state, the molecules vibrate in and out between their various components. If these equations are to hold in absolute degree, then all of these three states must achieve equilibrium to a known degree during the fraction of a millisecond that the gas is traveling down the nozzle of the rocket. For this reason, the correction C_D is included in the equation.

Effect of Nozzle Area Ratio

Figure 6 shows how C_F, the thrust coefficient of the nozzle, varies with the expansion ratio. Values of C_F (when $\gamma = 1.25$) are plotted against nozzle area ratio, which is the ratio of the area of the expansion cone to that of the nozzle throat. For example, with a nozzle area ratio of 4, an optimum expansion ratio of about 25 to 1 is obtained; in other words, the chamber pressure is 25 times the outside pressure. And at that particular point, C_F in Eq. 5 is about 1.45.

What happens if we take that same rocket and fly it to high altitude? The pressure on the inside of the rocket is kept the same, the same nozzle is used, but, as we go to high altitude, the outside pressure diminishes. By following the line for the nozzle area of 4 up to $P_c/P_e = \infty$, we find that

C_F is about 1.63. So, as we follow this nozzle up out of the Earth's atmosphere and eventually get into hard vacuum where the outside pressure is zero and the expansion ratio is infinite, there is a more efficient use of the fuel as we fly higher, evidenced by the increase in C_F.

The graph is not carried above a nozzle area ratio of about 25 because the curve begins to flatten off as the altitude increases. Since the nozzle gets bigger with higher altitude, eventually the increase in fuel impulse will not be enough to carry the extra weight of the nozzle. That is the reason it is not practical to make a nozzle with an expansion area ratio above about 25 to 1.

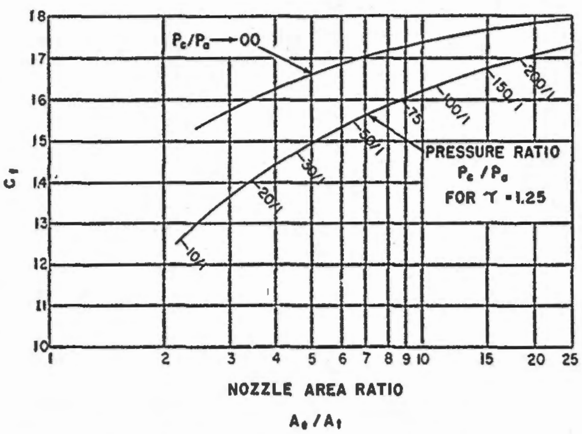


FIG. 6. Variation of the thrust coefficient of the nozzle with the expansion ratio.

Fuel Energy

Let us consider the factor C^*, fuel energy, to see what it is and how far we can go to improve specific impulse through changing fuel energy. Remember that the combustion process develops the enthalpy which, by use in an engine, is converted into flow energy to derive thrust force, which in turn determines the efficiency of a fuel in terms of specific impulse.

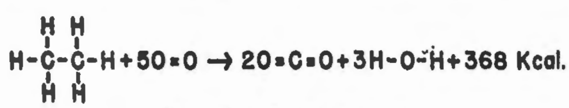


FIG. 7. Combustion and bond energy.

Figure 7 shows a simple combustion reaction involving hydrocarbons where ethane is mixed with oxygen, giving the standard combustion products, carbon dioxide and water. When burned, each mole of this mixture of the reductant and oxidizing parts of the fuel gives 368 kilo-calories of energy.

Where does this energy come from? The valence bonds (Fig. 7) are really formed by forces involving electrons in the outer shell of the atoms. These electrons arrange themselves between the carbon and the hydrogen, between the carbon and the carbon, and between the oxygen and the oxygen. In carrying out this reaction, energy must be furnished from some source to split up these bonds into their components, so they can form new bonds—oxygen to carbon, and hydrogen to oxygen. The energy resulting from the formation of these bonds furnishes enough energy to break the original bonds, with 368 kilo-calories per mole left over.

This is an extremely important point, for it would be much better if these original bonds did not have to be split, for then we would have free-radical propellants These, of course, would be quite unstable because they would be always ready to react with something. It would be extremely difficult to keep them in that form—as difficult perhaps as the problem facing the old philosophers who were searching for the universal solvent that would dissolve anything, including the vessel to put it in! This is almost the problem we are faced with in the free-radical chemistry of fuels.

TABLE II.

C—H	80	C═O	256
C—C	137	H—O	103
O═O	117	N═N	225
C—N	129	H—F	134
N—H	85	C—F	106
F—F	36		
N═O	150		
H—H	103		

In Table II, Column 2 gives the number of kilo-calories needed to break the bonds of the common fuel materials shown in Column 1. Column 3 lists the common combustion products and Column 4 gives the kilo-calories per mole obtained from the formation of these bonds.

By breaking up the bonds of the original fuel to form new bonds, energy is released in a combustion mechanism. In other words, one configuration of atoms locked in certain molecules is changed to a more stable, lower energy configuration of atoms locked in molecules; the difference in these two energy levels is the energy that we have to work with in our rocket fuel mixture.

I have called the items in Column 1 standard fuel components, but this is not quite true because hydrogen is an excellent candidate for a free-radical fuel. In other words, if hydrogen, in a monatomic form, were allowed to react to form diatomic hydrogen, 103 kilo-calories of energy would be obtained for each mole of hydrogen formed. It would have a molecular weight of 2 and would form a very high flame temperature. From the standpoint of specific impulse, hydrogen is the highest specific impulse candidate that we have today as far as all the chemical fuels are concerned. By chemical fuels I mean those things that involve certain molecular rearrangements to a lower energy form as has been represented by Fig. 7 and Table II.

As far as these rearrangements are concerned, carbon has an atomic weight of 12 and it has four electron bonds that can be rearranged to form carbon dioxide. Hydrogen, with an atomic weight of 1, has only one such bond. Sulfur, on the other hand, has an atomic weight of 32 and has only two or four bonds that can be rearranged to form H_2S or SO_2. Since the number of bonds per unit weight is not as large with sulfur as it is with these other materials, it is presumably a poor fuel candidate.

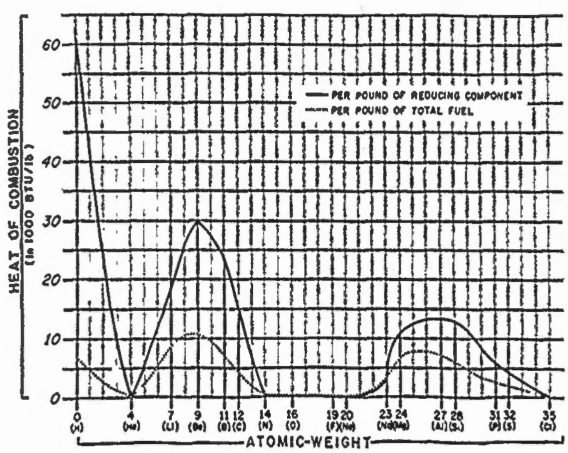


FIG. 8. Heat of combustion *versus* atomic weight.

Fuel Reduction

In Fig. 8 the heat of combustion in Btu's per pound for various elements is plotted against atomic weight. This graph illustrates the fact that the number of bonds or electron configurations rearranged does not increase as the weight goes up and, therefore, we find that for a rocket fuel, the light elements are the materials which will have high energy content per pound.

The dotted line in Fig. 8 represents Btu's per pound of total fuel mixture, counting the oxidant and the reductant together and using oxygen as the oxidant. Notice that on this curve hydrogen isn't as good as some of the other light metallic elements, such as beryllium and boron. A glance at the peaks of both curves in Fig 8 will explain why most of the jet fuels are taken from the lighter elements, in the range of atomic weight from 0 to 12. These lighter elements, including their hydrides and other compounds in their low molecular weight form, are the backbone of our high energy rocket fuel systems.

TABLE III.—*Rationalized Specific Impulse of Rocket Fuels.*

Fuel Component		Specific Impulse at 1000 psi and Sea Level
Reducing	Oxidizing	
Polysulfide rubber	NH_4ClO_4	235
Paraffin	NH_4ClO_4	250
Paraffin	NH_4NO_3	210
Gasoline	H_2O_2	240
Gasoline	HNO_3	250
Gasoline	O_2	270
Hydrazine	O_2	285
H_2	O_2	350
Hydrazine	F_2	330
H_2	F_2	370
$B_{10}H_{14}$	NH_4ClO_4	?
"HEF"	F_2	?

Specific Impulse of Rocket Fuels

Table III contains guesses as to the impulse that might be obtained from various fuel mixtures. The original calculations were based on "frozen flow" equilibrium—that is, no rearrangement of molecules in the nozzle. The calculated values were then corrected by experience and intuition. Some are quite accurate and some of the exotic mixtures may be 10 per cent in error, perhaps on the optimistic side.

Although the table shows that the use of polysulfide rubbers as a reducing and binder component with ammonium perchlorate as the oxidant gives a peak specific impulse of only about 235 pound seconds per pound (which is low), my company has used this combination in solid propellants for years, with reliable results. The reason why the polysulfide rubbers are not particularly good as the reductant components of fuels is that they have a sulfur content of about 37 per cent, which, as I mentioned previously, is not good. However, other properties of these materials permit making rockets that operate fairly reliably—a point that usually is of more practical importance than impulse.

The paraffin hydrocarbons with ammonium perchlorate should develop about 250 pound seconds per pound. The paraffin hydrocarbons with ammonium nitrate (another common solid oxidizer) do not make a very powerful combination because it takes a lot of its own oxygen to burn itself.

The rest of the table gives figures on gasoline and hydrogen peroxide, 240; gasoline and nitric acid, 250; gasoline and oxygen, 270; hydrazine and oxygen, 285; hydrogen and oxygen, 350. (This 350 figure looks good compared to the others, but think of the trouble involved in keeping hydrogen in a condensed form! Besides, the density of hydrogen is quite low, which adversely affects the mass ratio of the rocket engine system.)

Note that oxygen does not have to be used as the oxidant—there are certain materials advantageously burned by fluorine Table III gives some figures for fluorine as the oxidizer: with hydrazine, 330; and with hydrogen, 370. Of the so-called stable molecular fuels, this probably represents just about the peak I did not include ozone, although it is a possible oxidizer if it could be stabilized. Hydrogen and ozone should give a higher specific impulse than hydrogen and fluorine.

As an example of fuels based on materials other than the carbon or nitrogen hydride fuels, I have included the so-called high energy zip fuel which is essentially a boron hydride compound. These materials do have the potential of getting well above the specific impulses listed in Table III. The question mark indicates that I don't know what the specific impulse of these fuels is. The calculated high temperature equilibrium constant of the combustion products, free energy of formation and other important values are so uncertain that it is difficult to calculate the specific impulse accurately.

Considering the several choices of stable chemical fuels listed in Table III, it looks as though, based on the specifications of a 1000-psi. chamber pressure, sea level external pressure, and an optimum nozzle, there is an outside absolute upper limit somewhere on the order of 400 pound seconds per pound, while somewhere around 300 is probably the upper practical limit for the materials now used in existing rockets.

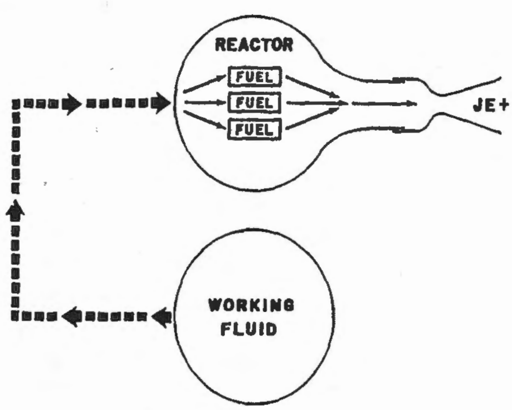


Fig. 9. Schematic of nuclear rocket.

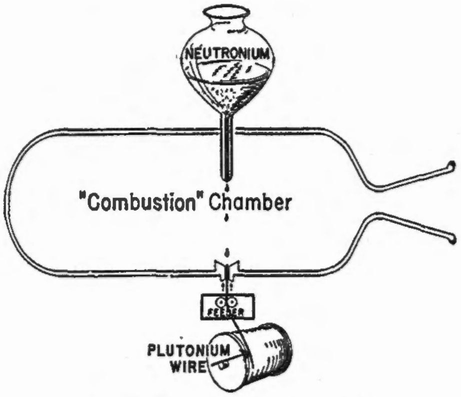


Fig. 10. Schematic of true nuclear rocket.

NUCLEAR ROCKETS

There is much talk about getting into space with an atomic-powered rocket. Figure 9 is a schematic drawing of such a rocket. Since we have to push against something, we have to have a working fluid; therefore, we carry along a tank of working fluid, which could be water or liquid hydrogen or almost anything you want to carry. This fluid is piped around to an atomic pile containing the fuel elements to heat up the working fluid which squirts out through the exhaust.

This looks good on paper, but if we do get an atomic rocket that works like this, will we get out through space? Remember that the specific impulse equations (Eqs. 3, 4 and 5) are dependent upon the way the working fluid is used in the expansion process, and what we get in terms of specific impulse is still a function—a square root, actually—of the combustion chamber temperature divided by the molecular weight.

How are we going to make this material work better, let us say, than a hydrogen-oxygen mixture? Are we going to get more out of it than we do out of the hydrogen-oxygen mixture in the Atlas or the Thor? May we assume that when the working fluid has passed through the fuel elements in the atomic pile it will be hotter than the exhaust from a molecular-powered rocket? We must remember that our present chemically fueled rockets are already working at flame temperatures above the melting point of known structural materials. What, then, are we going to use as structural materials for the nuclear fuel elements? These elements are hotter on the inside than the working fluid is on the outside because of the terrific heat transfer rate (on the order of 20 Btu's per square inch per second) needed to develop enough thrust to lift the rocket's own weight.

Until someone solves this problem, the atomic-powered rocket will not utilize working fluid any more efficiently than the chemical rocket.

Frankly, this type of atomic-powered rocket is not appreciably better (in fact, it might even be worse) than the best we can do with chemical fuels. Granted that it would have the energy, the working fluid has to be carried along and what we get out of this system is strictly a function of the ratio of the working fluid mass to total rocket weight. That means we have to get enough extra energy to pay for the extra weight of the atomic pile in order to make our rocket more efficient than a chemical rocket.

There is another way of utilizing nuclear energy in a rocket. Figure 10 is a drawing representing an idea of mine, which of course is impractical in light of today's knowledge. In this system, "neutronium" is a condensed

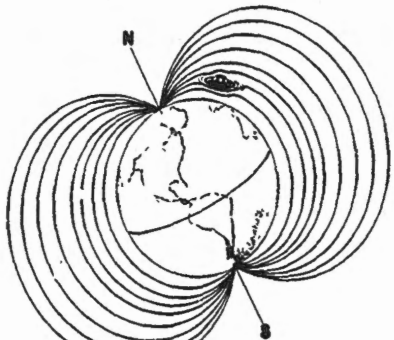


FIG. 11. Field propulsion.

form of neutrons. A plutonium wire feeds up through some rolls into the combustion chamber. Each time liquid neutrons drip onto the plutonium wire, we have fission, which generates an extremely high temperature. A very high velocity working fluid expands out through the exhaust jet. This is a true nuclear rocket in which we can really get something out of our fuel and we can travel out through space, as I will show you in a minute.

There are only a few little problems here. One of them is getting the condensed neutrons and the other, of course, is the materials in the construction of the combustion chamber. The temperatures generated by this kind of reaction are about ten million degrees Centigrade, and we have to have a combustion chamber that won't melt down. But let us not worry about that, because science can do anything, according to popular belief.

FIELD PROPULSION

There might be another way of doing it if we can get around this requirement for an exhaust jet by the use of fields. We might be able to do this in a much more simple and direct way by breaking the laws of Newton, or at least bending them to our will, to the point where we don't have to squirt a working fluid backwards to make a force.

To show you that it could be done, in Fig. 11 I have drawn a picture of the Earth and its magnetic field, with a flying saucer. This flying saucer has a coil of wire around the outside of it and it sends a terrifically high current through that coil of wire and generates a current sheet. Some of the Earth's vertical components in the magnetic field are trapped by that current sheet, creating a force that tends to lift the saucer away from the Earth.

You can imagine such a thing, but the execution of it becomes extremely difficult. I mention this merely to illustrate that it is possible to generate a force without squirting an exhaust jet backwards.

GRAVITY NULLIFIER

The idea of a gravity nullifier is an intriguing one, and in my own opinion this is the thing we are really waiting for in order to take advantage of the tremendous energy available from nuclear sources. If you look at what we already know today, we see mass—in other words, gravitational fields in the form of electrons, protons and neutrons which annihilate each other to give off particles that are essentially bursts of electromagnetic radiation. It looks as though we are on the verge of understanding electricity, electric fields, magnetic fields and electromagnetic fields well enough to begin to connect these with what little we know about the other type of field, and this is the gravitational field surrounding mass or matter. There are indications that, within the not too distant future, we will begin to understand fields well enough to learn how to apply the forces of electricity and magnetism in such a manner that we can develop a propulsion force that will counter the Earth's gravitational field. If we can do that, then we can put to work in a true way the power of the atom or the nucleus in the form of a propulsion device, and space travel will really become not only a thing of possibility, but a matter of convenience and pleasure.

In Fig. 12 I have represented two things that are important in rocket propulsion—the specific impulse of the propellant and the thrust-to-weight ratio. In the solid propellant rockets the specific impulse of around 250 is the best we can do, and we can do a little better with liquid rockets. In our solid rockets we can develop thrust-to-weight ratios of between 6 to 1 and 100 to 1. This is a range in which we can work. With the liquid rockets, the good ones, we are doing well to get a thrust that is equal to 1.2 times the weight. Somewhere around 1.2 times the weight to 10 times the weight represents the liquid rocket range.

This is nothing to be ashamed of, we will note as we go on up to these different propulsion systems. Free-radical fuels I haven't said too much about. I don't know what kind of thrust-to-weight ratio we can get, because we have never made a rocket with them. Theoretically we can calculate a specific impulse of around 1500, if we could stabilize atomic hydrogen. This certainly would be an excellent propellant and it would be classified strictly as a free-radical fuel. By the way, the hydrogen arc-welding system works actually on free-radical hydrogen, which is stabilized in the jet that comes from the welding torch.

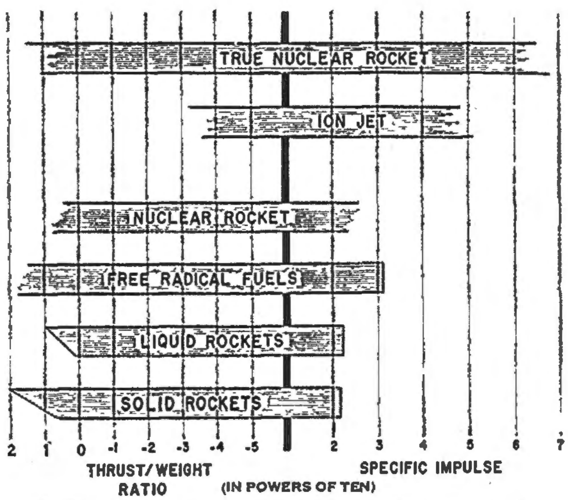


FIG. 12. Comparison of several types of rockets for thrust-to-weight ratio and specific impulse.

The nuclear rocket using the chemical working fluid we illustrated a while ago can produce enough thrust to take it off the ground against its own weight, if we are smart enough designers. However, I would guess that we are probably far from that stage at the present time. What we can achieve in the way of specific impulse is doubtful. It is quite doubtful, in my own mind at least, whether we will be able to do as well as we would with the free-radical hydrogen. So the specific impulse of somewhere in the 400 or 500 range might represent a very excellent achievement with the chemical working fluid put through a nuclear reactor as the heat source.

The ion jet is not a true rocket, because we get energy from another source. We can get specific impulses up in the 10,000 range or better by a very high acceleration of these particles in an electrical field and keeping them isolated by electrical and magnetic fields so we do not have to handle the extremely high temperatures that accompany the very high velocity. By the way, you can't have high velocity without temperature because temperature and particle velocity are essentially the same thing.

Now, the true nuclear rocket with the plutonium wire and the beaker full of liquefied neutrons is the one where we might get a specific impulse up in the millions of pound seconds per pound, and what it could do in the way of thrust-to-weight ratio is extremely problematical because the imaginary design shown in Fig. 10 has not yet been tested.

There is some very excellent work being done at the present time in the field of magneto-hydrodynamics, where it is possible that we will eventually learn enough to confine a fusion reaction within a magnetic "container" and actually achieve extremely high temperatures and extremely high velocities, giving specific impulses in the million range.

TABLE IV.—*Escape Velocity Missile.*

Stage load ratio	1/4	1/4	1/200	
W_p/W_e	0.92	0.92	0.99	
M_1/M_2	3.8	3.8	66	
I_{sp} equivalent value at 1000 psi	250	290	290	1×10^6
Take-off mass to payload mass	625	125	200	1.00125
No. stages	4	3	1	1

ESCAPE VELOCITY

We have considered many types of propulsion systems and their inherent capabilities. Now let us apply these to space travel in the form of an escape-velocity vehicle. Table IV is a calculation of escape velocities of various vehicles to give you some feeling for what we might encounter in going out into space. These are calculated on a stage-load ratio of 1 to 4. In other words, the rocket that is burning and propelling the stage weighs four times as much as everything it carries on top of it.

The propellant mass ratio—the weight of propellant divided by total engine weight—is 0.92 for the first two columns. The 1 to 4 load ratio and the 0.92 factor can be combined to show that the mass ratio in the missile itself is 3.8—in other words, initial mass divided by the final mass in each stage after the fuel is burned is 3.8.

This turns out to be a rather favorable number. By differentiating and carrying out a few other mathematical tricks I could show you that this mass ratio of 3 8 represents close to optimum so far as the conversion of fuel energy into kinetic energy of motion of the inert components of the rocket.

When velocity is calculated for a propellant specific impulse of 250, making appropriate corrections for the subsequent stages at high altitude in accordance with the C_F chart of Fig. 6, we find that it takes 625 pounds of take-off mass per pound of payload to achieve escape velocity. This gives an ideal velocity of about 39,000 or 40,000 feet per second because we must make certain allowances for drag losses and what we call g losses. It takes four stages to achieve this velocity.

This is probably better than we are demonstrating today. The Vanguard, which is supposed to be a very good rocket, does not achieve escape velocity—it only achieves satellite velocity (about 70 per cent of escape velocity). And the ratio of take-off weight to payload is about 1000 to 1.

With a specific impulse of 290 we can do better. We can actually eliminate a stage and this ratio between take-off mass and payload comes down to 125 to 1 to get to escape velocity.

I mentioned a while ago that if we could make the rocket consist of a high percentage of fuel, as we have in the third column where the engine is 99 per cent fuel, and M_1/M_2 becomes 66, we could achieve escape velocity with a one-stage missile. That is quite a difficult job and I will leave to you the choice of structural materials. You might notice that we are past the optimum ratio of 3.8 and you notice that the pounds of take-off weight to the pound of payload is actually higher with this one-stage rocket than with the three-stage vehicle.

How about atomic fuel? If we can make the true atomic-powered rocket work, with a specific impulse of about a million or better, we get into a very favorable situation. It only takes 1¼ pounds of fuel for a thousand pounds of take-off mass, 2½ pounds per ton, which would permit travel out in space in comfort and splendor. And if the landing field on the moon is fogged in when we get there, we can come back to an alternate landing field at LaGuardia! Under these conditions, space travel becomes an extremely simple matter. All we have to have is a gravity nullifier or a structural material that will contain materials at temperatures in the 10,000,000 degree range.

In the meantime, we can get vehicles into space with chemical rockets. The larger loads will require larger rockets in the ratios previously discussed, and the rockets will have to generate a thrust high enough to lift these heavy loads at take-off.

SATELLITE INSTRUMENTATION—RESULTS FOR THE IGY

BY

S. FRED SINGER[1]

It is a real pleasure to be here tonight to participate in this astronautics series and to discuss the many aspects of satellite instrumentation, particularly the scientific aspects. It is a subject which interests me very much.

It was really Newton who invented satellites, so that all of us today are Johnny-come-latelies. What Newton did two hundred years ago was to establish the fundamental physical laws which govern all satellites, whether they are natural satellites or man-made ones. What we are really doing today is practicing experimental celestial mechanics for the first time, by making up satellites to suit our own fancy instead of relying on satellites, such as the moon, which have been provided for us.

It took three hundred years to develop the necessary technologies to put these satellites into operation. I am thinking particularly of two developments: (1) the rockets which are the only devices known to produce the high velocities necessary for launching the satellites; and (2) the electronic instrumentation necessary for operating the satellites.

Newton, in formulating his third law, really laid the groundwork for rockets, although I am, perhaps, giving Newton more credit than he himself would take. However, I don't think Newton could have foreseen electronic instrumentation. In this field we have definitely advanced over him. In fact, I don't think Newton or anyone else a few years ago could have conceived of all the scientific applications which satellites will provide and are providing for us already.

Basically, of course, the scientific value of the satellite derives from the fact that it operates above the atmosphere for long periods of time, unlike high altitude rockets which merely go up and come down again. Having worked with high altitude rockets, including the V-2, the Aerobee and, more recently, our own small rockets Terrapin and Oriole, I feel very possessive about rockets. I would like to defend them against satellites by saying that rockets and satellites complement each other. While satellites study the events which take place outside the Earth's atmosphere, particularly the incoming extra-terrestrial radiations, rockets study the structure of the atmosphere itself, which varies with altitude.

Tonight I shall discuss three main topics: (1) what can be learned from the behavior of uninstrumented satellites; (2) what can be measured in a simple satellite carrying a minimum of instruments (represented by the

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Sputnik series, the Explorer series and the Vanguard series); and (3) what instruments were used in the satellites that have been launched, and how the Sputnik instrumentations compare with those of our satellites.

UNINSTRUMENTED SATELLITES

First of all, what can be done without instrumentation? Even though a satellite doesn't carry any instrumentation, its orbit is of interest. From simple observations of the satellite orbit, and in particular from the way the orbit changes with time, very many important things can be deduced.

Let me start, then, by discussing very briefly some aspects of satellite orbits

Figure 1 illustrates the fact that the orbits of the satellites are completely within our control. We can decide in a more or less democratic way (depending on what committee you sit on) how our satellite is going to go: whether it will go in an equatorial orbit which stays completely in the equatorial plane of the Earth; whether it will go in a polar orbit which I feel is probably the most useful one from the scientific point of view, but which has not yet been achieved; or whether it will go in an inclined orbit, that is, an orbit inclined to the equatorial plane of the Earth.

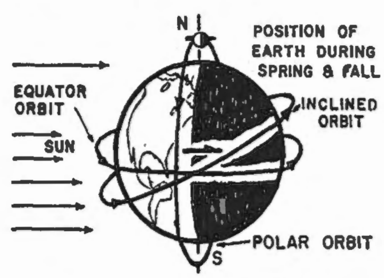

FIG. 1. Possible satellite orbits.

The factors which dictate the choice of the orbit are very many, and I don't want to go into them in too much detail except to say, as you may have guessed, that ultimately they are all guided by our pocketbooks. It costs a lot more to produce any orbit except an inclined orbit.

At least for our satellites which are launched into orbit from Florida, the angle of inclination of the orbit is fixed by the location of the range stations which the satellites must pass over. Considering the cost of the launching station and equipment in Florida, it would have been perhaps unwise for us to have insisted on a polar orbit, even though we are convinced that it is more useful scientifically. The Russians, by reason of geography, have used an orbit which is much more steeply inclined to the equator than ours—theirs is about 65 degrees as compared to our 40 degrees.

Figure 2 is a trace of the projection of our satellite onto the Earth. Remember that the orbit remains fixed in space, while the Earth turns underneath it. The orbit can be thought of as a ring, if you like, which stays fixed in space with the Earth turning on its axis underneath that ring. As shown in the figures, the U. S. satellite covers the range of latitudes from about plus 40 to minus 40 degrees. The Russian satellite covers the range from plus 65 to minus 65 degrees, which means that it can be seen over a wider region of the Earth and, in turn, the satellite itself can observe a larger region of the Earth underneath it.

Were the satellite in a polar orbit, we would indeed be able to cover all of the Earth, and this has very important applications. An obvious one, of course, is weather research, with all of the Earth under observation.

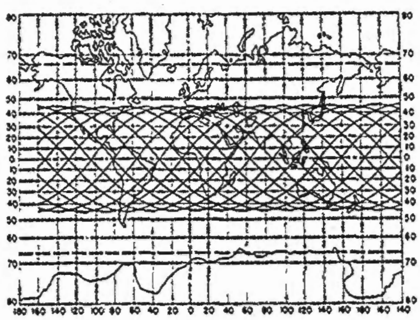

FIG. 2. Trace of an inclined-orbit satellite over the Earth's surface.

In order to have an equatorial orbit, the satellite must be launched at the equator. An equatorial orbit has certain advantages, perhaps for communication satellites, but for scientific satellites by and large we would prefer an orbit which is very much inclined to the equator, preferably 90 degrees.

I mentioned that the orbit stays fixed in space, that is, the ring (if you can think of the orbit as a ring) will remain fixed in space while the Earth turns underneath it; and as the Earth moves around the sun, it carries this ring along with it.

Now, in fact, the ring does not stay fixed in space and the reason is quite simple. Think of the satellite running along this ring and think of the ring, therefore, as rotating about its axis. It acts in essence as a gyro wheel and as such, it would keep its orientation in space forever, if it weren't for the fact that the equatorial bulge of the Earth produces a very small torque on this gyro. (The Earth is not a perfect sphere. The equator bulges out somewhat because of the rapid spin of the Earth, and this small bulge acting at the equator produces a torque which tends to pull the ring towards the equator.)

When a torque is applied to a gyro, it precesses. The same thing happens in a satellite orbit—the orbital plane of the satellite precesses. We can measure the rate of precession of the plane quite precisely by watching the satellite orbit, and from this we can immediately deduce the amount of bulging of the Earth's equator. Therefore, we can get very important information about the exact shape of the Earth. Again, I want to point out that this can be done without any instrumentation whatever on the satellite

Up to now I have talked about the orbital plane. Let us for a moment consider the actual shape of the orbit within the orbital plane. According to the planetary laws of Kepler and Newton's explanation of them, the normal orbit of any satellite is an ellipse A circle, of course, is a very special case of an ellipse and hardly ever occurs. The moon's orbit about the Earth is very nearly circular; the eccentricity is quite slight. But to be quite correct, all orbits we know of are ellipses, and the artificial satellite is no exception.

All of the artificial satellites to date have been launched into elliptic orbits; however, the orbit of the satellite doesn't remain fixed with time. At the beginning, the satellite moves in an ellipse, from a point of closest approach which we call the perigee, to the point of furthest approach which we call the apogee. This elliptical orbit would exist for all time if it weren't for the Earth's atmosphere, which decreases in density with altitude until finally it blends imperceptibly into the interplanetary gas. There is no real vacuum in the solar system—there is always a small density of gas apparently persisting.

At the normal sort of perigee we are talking about, perhaps a couple of hundred miles or perhaps only a hundred miles up, the density is very, very small. In fact, the density is less than the density in the best vacuum we can obtain here in our laboratories, but it isn't good enough because the satellite goes through the perigee many, many times and at very high speed. Even the few molecules and atoms which are present will take some of the energy from the satellite just due to collisions. So, aerodynamic drag (aerodynamic friction) has a small but steady effect on the satellite. This friction, however, takes energy from the satellite only at the perigee, because during the rest of the orbit, the satellite is traveling at such high altitudes (with correspondingly decreased density) that there is not enough atmosphere to cause friction.

The result of this energy loss is quite simple to evaluate. If one assumes that all of the loss takes place right at the perigee, it can immediately be shown by the simple law of conservation of angular momentum that the apogee has to move in (see Fig. 3). While the perigee stays relatively fixed, the apogee will move in steadily, until finally the orbit begins to approach that of a circle; and then, of course, the energy loss takes place throughout the orbit and the satellite will eventually spiral in towards the Earth's surface, probably melting or disintegrating along the way as it hits the denser layers of the atmosphere.

What can we learn by observing this progressive change in the orbit of the satellite and the shape of the orbit of the satellite? Quite a lot. Primarily we learn about the atmospheric density of the perigee and, strange as it may sound, we have really no good ideas of what the atmospheric density might be at 200 miles up. There have been some measurements from rockets, but they are suspect. There have been some calculations, but they can't be accepted finally until they are tested experimentally.

The satellite, then, is the only good method we know of for measuring density. We are already getting this from the Sputniks—in fact, we are getting the densities before the Russians because we have somewhat better observing and reducing setups. There have been already two publications which show that the densities are rather higher than we imagined them to be.

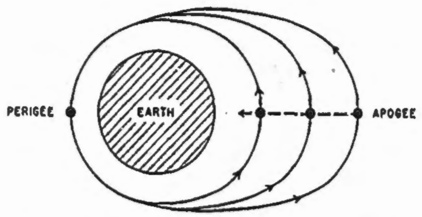

FIG. 3. Decrease of eccentricity of elliptical orbit due to drag at perigee.

One word of caution (this is a very special point): we are not one hundred per cent sure that we are really measuring the atmospheric density because there may be other effects which have not yet been evaluated completely. If the satellite acquires an electric charge in its travel around the Earth, then other forces may enter in, which will falsify some of these readings. One of the questions being debated very hotly right now is whether the satellite does acquire a charge, how much of a charge, and what influence this charge will have on the drag.

Let me explain the reason for this: because the atmosphere is composed principally of ions, that is, atoms which have lost their electrons and are, therefore, charged, electrical interactions may take place between ions and the charged satellite.

You have now seen two very important applications for a satellite which carries no instrumentation, but I forgot to mention one thing—you must be able to see the satellite! At least, you must be able to establish its orbit somehow, and by far the most convenient method is to see it optically.

This is not very easy, as many of you may have found out, since the physical conditions have to be just right. Clear weather is by no means the only requisite. Let me illustrate for you, since this is of some scientific importance, the physical conditions which have to be satisfied before you can see the satellite.

Since the satellite can be seen only by virtue of reflected sunlight, it cannot be observed when it is in the Earth's shadow. Figure 4 shows the sun on the left, with the satellite on the edge of the region of the Earth's shadow. As soon as the satellite hits the Earth's shadow it becomes dark.

In Fig. 4, the satellite is shown at the place where the probability for seeing it is highest. There are two reasons why this is so. In the first place, the phase of the satellite is good, since the whole disk is illuminated. If the satellite were on the other side of the observer, he would be looking at the unilluminated back side of the satellite and so, at best, he would see

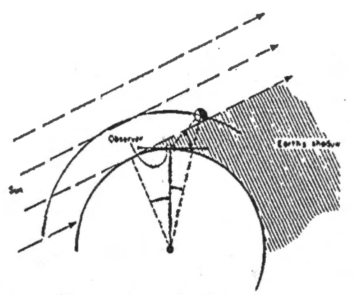

FIG. 4. Optical visibility of satellite.

only a crescent—if he could resolve it. (On the other hand, if the satellite were just a little to the left of where it is in Fig. 4, you might just be able to see it better because even though the phase is worse, it is a little closer to the observer. This matter has been worked out in some detail and has been published.)

A second condition which must be fulfilled is that the satellite must be bright against a dark background. It doesn't help to have the satellite illuminated if it is still daylight. You cannot see the stars in the daytime no matter how hard you try, unless you use elaborate optical equipment, and the same holds true for the satellite.

Under the conditions shown in Fig. 4, the atmosphere above the observer is in the Earth's shadow. In other words, for the observer the sun has already set (or has not yet risen) and he is in darkness and so is the atmosphere above him. Therefore, when he looks up, the sky, will appear to be dark, and under those conditions the satellite will be visible.

This explains, incidentally, why there are only certain regions on the Earth where the satellite can be seen at any particular time. It has to be at about local twilight, just after sunset or just before sunrise.

USES FOR INSTRUMENTED SATELLITES

To discuss comprehensively what can be measured in a simple satellite, we should really know a little about extra-terrestrial radiations, some of which can be measured by satellites and others by different methods.

"Radiation" is a very general word. As you know, there are two types of radiation coming from outer space: (1) electromagnetic waves having no mass (essentially a form or variation of light or radio waves); and (2) corpuscular radiations (particulate matter), having a mass.

Electromagnetic Waves

Let us look first of all at electromagnetic waves. Figure 5 is a graph of electromagnetic waves as we know them, ranging all the way from the very long waves which are hundreds of meters long, down to waves of the order of a few centimeters, millimeters, microns, and finally waves which are only a few Angstroms in wave length. In other words, we are going from long radio waves to short radio waves to extremely short radio waves (the so-called microwaves or millimeter waves) to the infra-red radiation through the visible radiation to the ultraviolet and into X-rays. This covers the whole spectrum of electromagnetic waves.

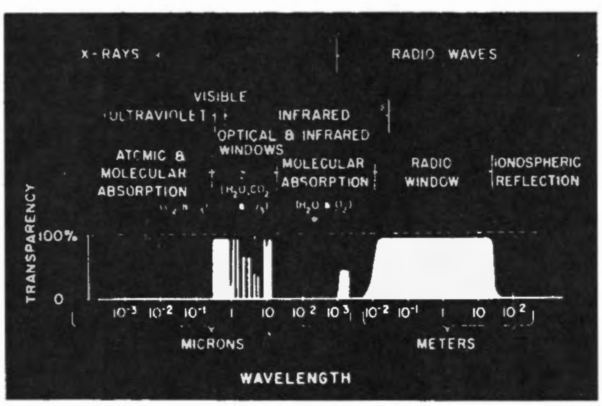

FIG. 5. Transparency of Earth's atmosphere for electromagnetic waves.

Heavenly bodies, such as the sun and any other radiant body in space, send out all of these waves However, here at sea level we generally observe only the visible radiation. There is a reason for this. The atmosphere is transparent (transparency is shown in white in Fig. 5) only in the visible part of the spectrum. There are a few transparency bands in the infra-red, and then there is a region of transparency again in the radio region called the radio window, the word "window" implying transparency.

This means that all of our knowledge of the universe is derived from the fact that the atmosphere is transparent If the atmosphere were not transparent in the visible, we would not be able to see the sun, the stars or the moon, and we would never know of their existence in all likelihood. We would have to hypothesize them as we hypothesize the inside of atoms now.

But thanks to a very fortunate coincidence, the atmosphere is transparent in the visible region, quite a narrow region, as you notice, called the optical window; but our eyes also happen to be sensitive in the visible region. It would do us no good if our eyes were sensitive only in the ultraviolet, because then we would still be in darkness. However, it so happens that the atmosphere is transparent in the visible and our eyes are sensitive in the visible region, and this is the region in which the sun puts out most of its radiant energy. This fortunate combination of circumstances makes it possible for us actually to see and to use the sun as a useful source of light. And, as I mentioned previously, all of our knowledge of the universe up until recently had been derived through visual observations.

In the last decade, a new science has arisen because of our advances in radio technology, and we can now observe the universe in the radio region, through the radio window. Large radio telescopes are being constructed all over the world. A tremendous radio telescope has been constructed in England recently, and the United States is now constructing its largest radio telescope in West Virginia. Some of the most difficult engineering problems associated with this U. S. telescope are being solved right here at The Franklin Institute.

If one considers how much has been learned about the universe from radio observations, it becomes apparent how much more we could learn if we could only observe freely in these other regions of the spectrum which are now completely inaccessible to us. For example, we cannot see any of the infra-red radiation being produced in space and coming towards the Earth because molecules in the Earth's atmosphere absorb the radiation before it can reach us. This infra-red region can be partially explored by going up in balloons to a height above most of the H_2O, CO_2 and most of the oxygen, too.

Long radio waves, on the other hand, present a real problem. We can't see them because when they come towards the Earth they are reflected back into space by the ionosphere, which exists at an altitude of about a hundred miles. The ionosphere, incidentally, acts as a reflecting layer in both directions, so that long radio waves cannot escape from the Earth because they are bounced back by this layer.

Perhaps the most interesting region for us to observe would be the ultraviolet region. Already we have had some very tantalizing glimpses at the sun, which is our nearest and therefore our most important star. By using rockets equipped with specially designed detectors, we have learned quite a lot about the sun, in the ultraviolet.

We have found out, for example, that during certain periods when the sun gets very excited, its output of ultraviolet increases by as much as a factor of 10,000. So, contrary to our usual impression of the sun as being quite a normal and fixed star (fixed in the sense of having a constant output), we find that in the ultraviolet region the sun is an extremely variable star, perhaps something like the flare stars in the visible region.

If we could only observe the sun in the ultraviolet continuously, we would discover much more about the processes which take place in the solar atmosphere, and we would learn more about solar physics and the way stars operate. Incidentally, we would also be able to interpret much better how this ultraviolet radiation from the sun affects us by affecting the Earth's atmosphere.

The ultraviolet radiation is all absorbed in the upper atmosphere and thereby produces the ionosphere, which is extremely vital to us for long distance radio communication. Changes are taking place in the ionosphere almost continuously, and these become very pronounced when the sun acts up. Therefore, we would like to be up there to look at the sun so we can correlate what we see coming from the sun directly with what we observe indirectly as happening in the ionosphere.

Corpuscular Radiation

Let us turn now to corpuscular radiation, which is composed of particulate matter. In Fig. 6, corpuscular radiation is plotted as a function of energy, expressed in units of the electron volt. Corpuscular radiation ranges from 1 electron volt (just about thermal energy) to a billion billion electron volts (possessed only by the extremely high energy cosmic rays).

Cosmic rays, which cover the energy range from about a billion electron volts on up, are the most energetic phenomenon we know of in nature. Primary cosmic rays are hydrogen atoms, with some helium and heavier atoms. They are really nuclei that have lost their electrons, moving with nearly the speed of light. These particles slam down into the atmosphere and produce nuclear disintegrations whose end effects we can finally observe at sea level.

As you probably know, the work of the Bartol Research Foundation during many years now has been concerned with studies of the cosmic radiation, both of the secondary radiation and pioneer experiments with the primary radiation using balloons which get fairly close to the top of the atmosphere from the point of view of cosmic rays.

Rockets or satellites are really needed only to study low energy cosmic rays, which cannot penetrate below the 100,000-ft. altitude (typical balloon altitude). We don't even know whether these low energy cosmic rays actually exist. Perhaps they exist only at certain times. Perhaps they are produced by the sun and exist only when the sun produces flares.

Particles of even lower energies, of the order of a million electron volts, are auroral particles which cause the aurora borealis, the northern lights and the southern lights. These auroral particles are probably protons which come from the sun with low energies and which are then somehow accelerated in the vicinity of the Earth (by a mechanism which we don't as

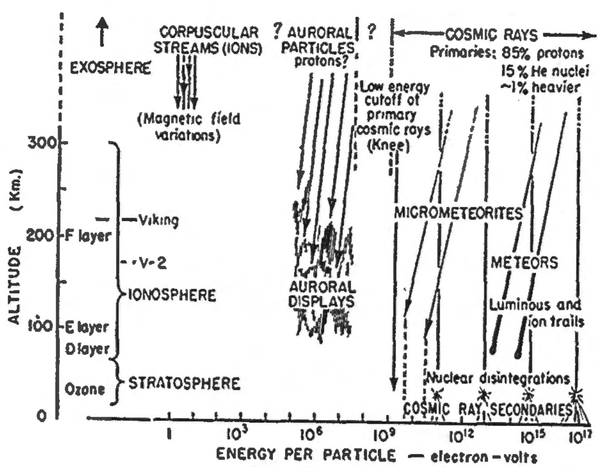

Fig. 6. Penetration of particles into the atmosphere.

yet understand) until they have energy enough to slam into the atmosphere and produce the beautiful displays which many of you, I am sure, have seen.

In going to even lower energies we have gas clouds, corpuscular streams which are shot out from the sun, again when the sun becomes very excited. Many of you may have seen the protuberances in pictures of the solar corona. When the sun is quite active, particularly during the high spot of the sun-spot cycle, these gas clouds are shot off very often. As they approach the Earth, they don't even hit the atmosphere because they are stopped many thousands of miles out by the Earth's magnetic field. When they hit the Earth's magnetic field, they produce quite violent changes which we call magnetic storms.

Measurements of Cosmic Phenomena by Satellites

Perhaps it would help if I described a typical series of events which follows a large solar flare, so you can see how complex the situation really is, how many different phenomena take place, and what we interplanetary physicists are up against, what we have to explain and how the satellites can help us explain all these facts.

Figure 7 is a schematic drawing of a typical series of events following a solar flare It looks complicated in the drawing, but it is much more complicated in practice.

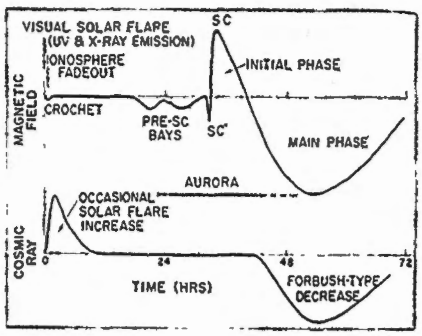

FIG. 7. Magnetic and cosmic ray variations following a solar flare.

The process starts with a big solar flare on the sun. These flares on the sun can be seen through an H Alpha Monochrometer, an instrument which filters out all of the radiation except the type which is emitted during flares.

During the visual flare there is intense emission of ultraviolet and X-rays from the sun which immediately gives a fade-out of radio signals in the ionosphere.

The Earth's magnetic field reacts very slightly to the flare because the ultraviolet radiation from the sun is absorbed in the ionosphere, affecting ionospheric conductivity and ionospheric currents. Every time there is a change in the ionospheric current, there is a corresponding change in the magnetic field, so that the Earth's magnetic field is slightly depressed, a so-called crochet.

About a day later things begin to happen. Suddenly the magnetic field strength at sea level over the world on both sides of the globe, day side and night side, rises by a large amount, stays up for several hours, then decreases by a large amount, and then stays low for several days. This is a result of the impact of the solar gas, but it is a very controversial subject—

the problem of just how the gas impacts on the Earth's field and how it produces the increase and decrease is a problem which is not yet conclusively solved.

At the same time most of the aurora begins to appear, probably because the gas shot out from the sun also contains these very high-speed particles which can penetrate deep into the atmosphere and produce the aurora.

Immediately following the flare, perhaps several minutes later, the intensity of cosmic rays suddenly rises, sometimes by as much as some few thousand per cent, indicating very strong production of cosmic rays right at or near the sun. This increase lasts for several hours and then the cosmic ray intensity returns to normal. However, when the magnetic field decreases, the cosmic ray intensity suddenly decreases by 10 or 20 per cent, and stays low for several days

Although these events are very complicated to describe and to explain, an understanding of them is vital. If we want to understand what really goes on in interplanetary space, we need to understand the magnetic conditions which exist out there. We are handicapped by not having sufficiently good data, but we think that the satellite will be able to give us adequate data to shed more light on these phenomena.

How do we measure these many phenomena that I have described? Many instruments can be crammed into a satellite, since some of the instruments are quite small, weighing only ounces or a fraction of an ounce in some cases.

Figure 8 is a schematic diagram of a satellite which I designed a few years ago. It has a feature that satellites really ought to have—some means for keeping their orientation in space. For example, if the sun is being studied, the solar detector should be pointed at the sun all the time; if the Earth is to be watched, the detector should be pointed towards the Earth. It does no good to have the detector pointing in other directions.

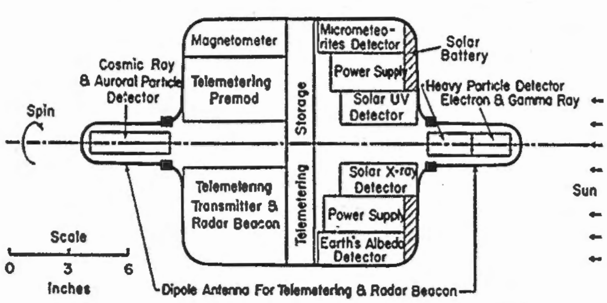

FIG. 8. Diagram of gyroscopic stabilization scheme, solar power supply and astrophysical instrumentation for Mouse satellite.

In order to do this, I adopted a principle of providing a spin axis and spinning the satellite about the axis. The spin axis points at the sun before take-off and, according to the gyroscopic principle, the axis will stay fixed in space and keep pointing to the sun for an appreciable period of time, long enough to make possible long periods of observations

Another instrument which every satellite ought to have is a telemetering transmitter, or some means of conveying the data back to the Earth. This requires a power supply to power the transmitter and the instrumentation. The power supply may be regular batteries, but they are relatively heavy. A more sophisticated approach is to use solar batteries to supplement the internal power supply. These solar batteries are attached to the side of the satellite which faces the sun.

Obviously it would be very uneconomical to cover all of the Earth with receivers. Therefore, the data gathered by the satellite are stored in the telemetering system on a tape recorder; then, when the satellite passes over a receiver, the transmitter is turned on by means of a radar beacon and the satellite spews forth all of the accumulated information.

Although these refinements are not in satellites now, I feel that eventually these features will be incorporated. The solar batteries have just been tried on the first Vanguard test satellite, and I think telemetering storage is close at hand. So far no one has tried attitude control by spin, but it is probably the simplest method for maintaining the orientation of the satellite.

Instruments to Use in Satellites

Now let us talk about the scientific instruments. How do you measure cosmic rays? The best and simplest method certainly is to use a Geiger counter, which is a gas-filled cylinder with a thin anode wire. These Geiger counters, particularly if they are thin-walled ones, will count cosmic rays and also auroral particles. Specialized types of Geiger counters can also be used to count gamma rays and the heavy component of the primary cosmic rays.

How do you measure the sun? The most direct way is with a photon counter, a special type of Geiger counter which can be used both in the ultraviolet and for the X-ray regions. Both counters have windows which face the sun (see Fig. 8).

The impacts of micrometeors can be measured by means of micrometeorite detectors. One such detector might be a microphone which simply counts the impacts; there are other more sophisticated types of detectors which I will discuss later.

A more difficult task is that of measuring the Earth's magnetic field at high altitudes above the ionosphere. The best instrument which we have developed up to now is a so-called proton precession magnetometer, which I won't describe here since it is described adequately in the literature.

Let us now consider more direct measurements in the satellite. I think micrometers are of some interest. A simple method has been devised to measure the impacts of micrometeors on the satellite, and it may have some practical importance for space flight.

Every time a meteor hits the satellite it blasts out a small crater, in the same way that a meteorite impacting on the Earth produces a crater. The physical processes are very similar, except that when little dust particles smaller than grains of sand hit the satellite the craters, of course, are very, very small. Nevertheless, the skin of the satellite will become eroded and eventually it will be worn completely through, leaving a hole in the satellite.

In order to measure how rapidly this skin abrasion takes place, we have devised the following experiment. We coat the outside of the skin with some radioactive material and then inside, within the satellite and at some separation from the skin, we have a specially designed Geiger counter with a scaling circuit. It is quite obvious that as the skin disappears the radio-

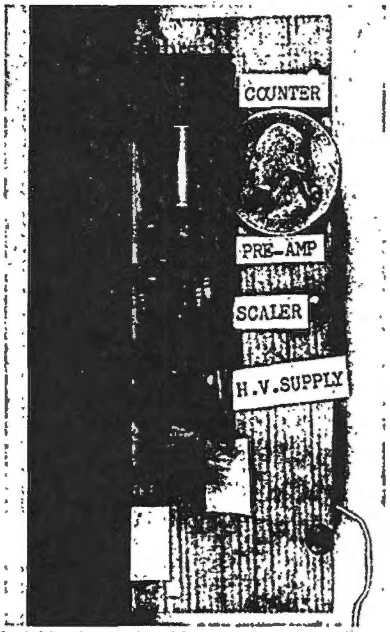

FIG. 9. Miniaturized version of instrument to measure skin abrasion.

FIG. 10. Meteorologist's concept of what the North American continent would look like as seen from a 4000-mile altitude.

activity disappears with it and at the same time, therefore, the counting rate of this monitor counter decreases. By telemetering to ground this counting rate, the amount of skin still remaining can be immediately determined.

Translated into practice, the instrument is extremely simple. Figure 9 shows a miniaturized version which has been prepared for a satellite experiment. It shows the special counter which is smaller than a quarter, a pre-amplifier, a scaling circuit and a high voltage supply. This high voltage supply is smaller than my thumbnail and supplies 700 volts to the Geiger counter.

Meteorological Applications

Perhaps one of the most intriguing things to measure, and certainly one of the most important from an economic point of view, is the so-called albedo of the Earth. By measuring this albedo, which is the amount of sunlight reflected from the Earth (mainly by clouds), we hope eventually to track storms and predict the weather. After all, our national income isn't going to be much affected by the intensity of cosmic rays and by what we can learn about micrometeorites. But the weather affects all of us and the satellite, in my opinion, is an invaluable tool for studying meteorology.

Figure 10, prepared by Dr. Harry Wexler of the U S. Weather Bureau, is his conception of what would be seen through a television camera looking down on the North American continent from an altitude of 4000 miles. The white fluffy areas represent clouds which are very good reflectors of solar radiation or visible radiation. They have an albedo (reflecting ability) of from about 70 per cent up to about 100 per cent. The Earth's surface is darker, since it has an average albedo of only 15 per cent, while the oceans, with an average albedo of only 4 per cent, appear to be quite black.

The satellite television camera can pick up these cloud patterns very easily, and the meteorologist can immediately recognize the type of weather pattern which they represent. For example, Fig. 10 shows a three-member cyclonic system extending from Hudson Bay towards Texas, with a well-defined front. Part of another cyclonic system is visible up near Alaska, with its other members extending over the Pacific. Notice the immense fog bank off Southern California and Newfoundland.

Detailed weather features are apparent. For example, there is a very small hurricane just forming in the West Indies. Hurricanes are very difficult to spot because the cloud patterns associated with them are not very prominent; but if you know that there isn't an island where a white blob shows, you suspect a hurricane, particularly in the hurricane season.

Little cumulus clouds show throughout the United States; also special types of lenticular clouds due to moist air being raised when it comes over the Rockies. Also shown are cloud patterns associated with the trade winds (so-called cloud streets) and cloud patterns associated with the equatorial convergence zone.

This type of display of the Earth and cloud patterns certainly permits a meteorologist to make much more accurate short-range predictions over several days; more than that, by studying the circulation of the atmosphere in this way, the meteorologist can learn how to make very reliable long-range predictions for such things as dry winters, wet summers and hot spells, over particular regions of the world.

This information, of course, is of tremendous economic importance and that is why I think that this particular application of the satellite is going to affect our way of life more than any other aspect of this space-flight business, except possibly for man to travel to the planets. It is a means whereby the space-flight business can pay its own way right now. The economic advantages of this type of knowledge are just tremendous. I think no one right now could estimate how much can be added to our national income by more accurate weather prediction. It probably runs into the billions.

PRESENT SATELLITE INSTRUMENTATION

I want to spend the last part of this lecture discussing present satellites and instrumentation. To date (March 18, 1958), four satellites have been launched: Sputnik I and II, Explorer I and, just recently, the first Vanguard test satellite, colloquially called "the grapefruit."

Let us discuss the Russian satellites first. We were all very impressed by their great size and heavy weight, but the main question is what they really do as compared to our satellites They are certainly indicative of a very high degree of development of large rockets, and they are quite reliable, too But instrumentally—and this is what I am mainly concerned with—they indicate that the science of miniaturization of instruments hasn't advanced very much, at least in a comparative way.

Most of the weight in Sputnik I consisted of batteries; some 120 pounds must have been batteries in order to operate the transmitter for the required period of time. The transmitter itself was a vacuum-tube transmitter, not a transistor. As a result, it dissipated a lot of power internally both in the filament and in the plate, and this caused problems, because the power generated heat had to be removed somehow.

But how do you remove heat in a vacuum? You can't do it very easily. Therefore, a forced circulation system for nitrogen (kind of air-conditioning system) had to be installed into their satellite. This required that the whole satellite be pressurized, which in turn meant that the skin had to be quite thick to withstand the atmospheric pressure of the internal air, or nitrogen.

You see how one difficulty leads to another and why it was necessary, therefore, to put a lot of weight into what we consider to be non-essentials. By going to transistors right away, we avoided these problems successfully.

Let us talk about other aspects. The solar batteries are certainly a great advance and I feel that both we and the Russians will probably go to them. However, if the instrumentation isn't transistorized, not much will be gained from using solar batteries because the power consumption would be so high that solar batteries would only add a little to the available power. On the other hand, if the instruments are transistorized, the solar batteries can supply all the power.

From a scientific point of view, in all cases both the Sputniks and our satellites have tried to measure internal and skin temperatures, in order to check up on the accuracy of the design. The temperature of the satellite depends primarily on the radiation properties of the skin, because the only exchange of heat energy is by means of the skin—radiation absorbed by the skin and radiation emitted from the skin.

In addition, other scientific experiments have been conducted, the most unique of which certainly is the experiment conducted on the dog. We cannot hope to match that until we have more payload, but in regard to experiments connected with physical radiations, we are in a very good position in comparison with the Russians.

It was only on Sputnik II where, for example, they measured cosmic rays. We are measuring cosmic rays on Explorer I. Figure 11 shows a cosmic ray instrumentation actually built for the Farside vehicle which was successfully fired last fall. It is very similar to the instrumentation which has been flown in Explorer I and it comprises a specially designed Geiger counter, a potted section containing a scaler to scale the counts from the

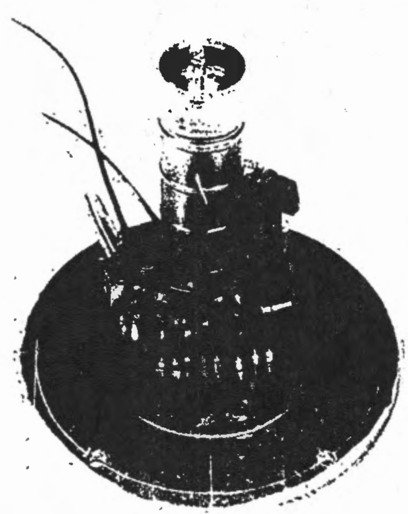

Fig. 11. Cosmic ray instrumentation built for Farside vehicle.

Geiger counter and feed them into the telemetering system, and finally, a high voltage supply to operate the Geiger counter. All of this weighs perhaps a couple of ounces and consumes only 18 milliwatts of power.

Figure 12 shows a Geiger counter scaler assembly built for a slightly different application, but it illustrates another type of miniaturization, including potting. Potting can be done successfully in transistorized circuits because the heat developed is very small. The shorter instrument shown in Fig. 12 is a little transmitter built for a rocket application. In fact, this whole assembly fits together inside a narrow pencil which forms one of our small research rockets, the Oriole. This small transmitter transmits an adequate amount of power for communication over distances of several hundred miles.

A few more words now about the types of measurements which will be made on future satellites. Of course, I can speak only about the American satellites. The next Explorer will probably contain a more elaborate cos-

FIG. 12. Miniaturized instruments for the Oriole rocket.

mic ray experiment. The next Sputnik, by announcement, will also contain a more elaborate cosmic ray experiment. In fact, I think the Russians are doing very well in their cosmic ray experiments.

The first real Vanguard sphere will contain a solar radiation experiment to measure ultraviolet radiation and perhaps X-rays from the sun. The Russian Sputnik II has already flown a solar radiation experiment somewhat different from ours, but quite interesting. We would very much like to know what results, if any, were obtained from it. It is a more difficult type of experiment than the cosmic ray experiment. They used a photo-multiplier. In front of the photo-cathode they placed a rotating filter wheel which carried a number of filter sections, and as it rotated, it allowed different parts of the solar radiation to enter into the photo-multiplier tube.

For the future Vanguards, several types of experiments have been planned, including one to fly a magnetometer to measure the magnetic fields at high altitudes. The fourth Vanguard satellite is to carry the first instrumentation to try to measure the Earth's albedo (the light reflected from the cloud systems) with a simple photo-cell. At very low altitudes, fairly adequate resolution can be obtained, giving a kind of fuzzy, out-of-focus picture of the cloud pattern.

What are some of the future experiments? I don't want to go too far, but I think one of the important experiments, certainly one of the interesting ones that we are working on right now, is the charge of the satellite. We would like to know what electric charge, if any, the satellite acquires, whether it is positive or negative, and how the charge changes as it moves about the Earth. This is a very difficult assignment and much thought is going into the design work for this type of measuring instrument. We have a proposal which we hope will work, but we will try it in the laboratory before putting it into the satellite.

Finally, you may wonder where further information can be obtained on this whole subject. There are many popular books on satellites, but at the present time, aside from the few listed below, there are no really serious detailed books on satellite scientific problems and instrumentation.

I hope that the proceedings of this astronautics series will serve as a means of putting on record some of our present thoughts. It would be interesting to see how they stand up against future developments.

REFERENCE BOOKS

EARTH SATELLITES AS RESEARCH VEHICLES, a symposium. 115 pages, diagrams, 6½ × 9½ in. Philadelphia, The Franklin Institute, 1956. Price, $2.50 (paper). (Editor's Note: This, unfortunately, is out of print.)

SCIENTIFIC USES OF EARTH SATELLITES, edited by James A. Van Allen. 316 pages, diagrams, 6¾ × 10 in. Ann Arbor, University of Michigan Press, 1956. Price, $10.00.

S. F. SINGER, "Geophysical Research with Artificial Earth Satellites." A chapter in "Advances in Geophysics," Vol. III, edited H. E. Landsberg. 378 pages, 6¼ × 9¼ in. New York, Academic Press, Inc., 1956. Price, $8.80.

CELESTIAL MECHANICS—ORBITS OF THE SATELLITES

BY

PAUL HERGET[1]

INTRODUCTION

Ladies and Gentlemen, I am overwhelmed by such a large audience. I think it is probably correct to infer that you are here because of the Sputniks; and, frankly, I am here for the same reason.

I was hesitant to accept the invitation to lecture. I have felt it was my duty to turn down all invitations religiously and not play favorites with anybody, but I changed my mind on one occasion and I changed it again for the same reason this evening. I gave a lecture to the Graduate Physics Seminar in our University because I feel it is important that somehow some young people (and I think we are missing a bet if we don't also include some who are not so young any more) must be willing to face up to the challenge.

It may well be that you all came here to be entertained this evening, but that is not why I came. I came for a serious purpose because I think this is a serious business. The thing that needs to be emphasized most of all so far as I am concerned is that the Americans have taken the term "complacency" too complacently. What we need to emphasize all through our school systems is that it is necessary to bring hard work to bear upon all of our educational system and this in general will improve science.

This is not a new idea to astronomers because astronomers have been engaged in prodigious tasks of one kind or another, at least those astronomers in history who have stood out; and I shall give you a few instances before the evening is over. There remain important tasks to be tackled in addition to those that are created by the artificial satellites.

We are most fortunate to be able to live in this age when these things are being done, and I think I can highlight this for the young people if I recall what my mother told me when I was about eight years old. She said, "It's too bad you were born too late. All the wonderful improvements have just been made and you missed them all."

In Cincinnati, where we have more hills than you have here, they no longer had cable cars. The cars ran up the hills under the power of their own electric motors, and my mother when she was young had ridden in the horse cars on the streets of Cincinnati. We had radios. There were airplanes in the sky. So she thought that everything had been done and it was too bad for me. Well, she is an old lady now and she doesn't under-

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stand all the things we have in electronics, and so on; and there is still more to come We make a sad mistake if we don't challenge our young people to the best of their ability.

WHAT CELESTIAL MECHANICS IS

The subject of Celestial Mechanics deals with the fundamentals of the various aspects of the operation of satellites In a manner of speaking, there is no sharp boundary to Celestial Mechanics. There is a sharp boundary to the problems that have been tackled in the past, but this is just because everybody was so compartmentalized in his own work. When we have a project such as these missile projects, people have to work together on teams and people have to bring different competences to these teams. So the sharp boundaries in technology are not going to remain so sharp as they were.

I would like to emphasize that there are various aspects of Celestial Mechanics, even though I won't be able to talk about all of them. First of all, I would like to emphasize that in order to deal with this field one has to depend upon the use of the law of gravitation and other physical laws. This is one aspect of the field of Celestial Mechanics.

Another aspect of this field is that one must be prepared to deal with mathematical analysis. You never can know too much mathematics for working in this field, and I wish I knew a lot more than I do.

Another aspect, in order to be fruitful, is some concept of what is involved in the use of observation. A great many people eventually find themselves out on some distant limb where they don't like to be simply because you have to be more than a theoretical physicist. Observations are an important part of what you have to deal with in this field.

There are two other items which are somewhat less specific, but I think both of them are important. It is necessary to have some kind of a practical bent even though the astronomer, dealing with the planets and the comets and the satellites of the solar system and the double stars of the universe, is someone you might think impractical; nonetheless, there is the necessity for a practical conception of what the problem is about and for arriving at some reasonable answer.

Finally, there is a requirement for a kind of intellectual curiosity or drive, and I say this because there is a tremendous amount of drudgery that goes on in order to obtain the solution to problems in the field of Celestial Mechanics. In one sense I am glad I wasn't born any earlier because now we have electronic computing machines and this is a tremendous lift to the whole field of Celestial Mechanics. I shall have some examples of this as we go along.

GENERAL PROBLEMS IN CELESTIAL MECHANICS

Before I get into the particular aspects of the subject, I would like to give you some perspective of the whole field of Celestial Mechanics and the problems in this field. These are really not confined solely to what has been considered the field of Celestial Mechanics in the past. They bridge over into the field of Astrophysics.

First, there is the problem of stellar structure which consists in studying and trying to formulate and give reasonably definite solutions to what the nature of the interior of stars is; and this, of course, is tied in very closely with the whole subject of Astrophysics.

In addition to studying the interiors of stars, the study of stellar and planetary atmospheres constitutes a field of problems in Celestial Mechanics.

Classically or historically the principal problem in Celestial Mechanics as such has been the problem of dealing with the motions of the planets, and this problem goes all the way back to ancient mythology and astrology. People had various systems in which they tried to reconstruct what the nature of the universe was insofar as the motions of the planets around the sun are concerned.

Another aspect of Celestial Mechanics very closely related to the motions of the planets is the motions of the comets, and if the occasion arises I use this as an example of how practical astronomy is because it is literally true that in ancient times people almost became insane in their conduct just at the appearance of a comet. After it became known what the nature of the orbits of comets really is, this whole foolishness was dispelled. In a manner of speaking, then, there are certain practical aspects to astronomy and to Celestial Mechanics.

Another closely allied field is the subject of the motions of the satellites around the planets and, of course, in astronomy the outstanding system of satellites around the planets is the system around the planet Jupiter.

The next planet having a large number of moons and having an entirely different set of circumstances associated with the moons is the planet Saturn. Our own moon around the Earth offers many problems entirely different from the moons of other planets.

Finally, we come to the artificial satellites of the last four or five months, which is, I assume, the subject in which you most interested and the one which I shall dwell upon the longest.

KEPLER'S WORK ON ORBITS

Now let me go back to the historical problem of the motions of the planets and describe to you, if you are not already familiar with it, a problem in Celestial Mechanics which was solved many centuries ago by Kepler before there were any telescopes—before optics had ever been developed to such a state and before there was anything except geometry for mathematics.

The Earth goes around the sun, which Kepler believed to be some kind of a nearly circular orbit, and out in the sky one can see the planet Mars. But one is not able to tell when Mars comes to 20th and Chestnut or to the stoplight at 18th Street and Market or any other place in the sky.

There is no intersection, no crossroad, no way of marking a point there for the person who sits on the Earth and looks out at the dark sky at night and sees the planet Mars, as people have done for thousands of years.

But what Kepler realized was that if the planet Mars was on the opposite side of the Earth from the sun, as shown in Fig. 1 at E_1 and M_1, then at a later date they would be at E_2 and M_2, and the relative rate at which the Earth overtakes Mars is represented by the angle E_2SM_2. The rate of the Earth's motion is shown by E_1SE_2; and the rate of Mars' motion is shown by M_1SM_2. Thus:

$$\sphericalangle E_1SE_2 = \sphericalangle M_1SM_2 + \sphericalangle M_2SE_2, \text{ or } \frac{1}{E} = \frac{1}{P} + \frac{1}{S},$$

where $E = 1$ year, $S =$ the observed period of Mars from opposition to opposition, and P is the actual period of Mars in its orbit, which cannot be directly observed. Since $S = 2.135$ years,

$$P = \frac{SE}{S - E} = 1.881 \text{ years.}$$

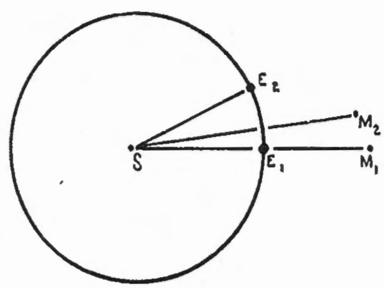

FIG. 1.

Kepler took observations, one of which was made at some date when the Earth was E_1 in Fig. 2 and another one was made after the Earth had gone all the way around 1.88 times to E_2. He simply chose his observations, paired them in such a way that they came at this interval of time apart; and then of necessity Mars was at the same place, M, in its own orbit even if Kepler didn't know exactly where this place was in space. Then, you see, if Mars is in the direction E_1M at one time and E_2M at the second time, they intersect at M and he has been able to locate the position of Mars in space only by assuming the explanation given above for finding P.

Let me point out one other thing. This is one pair of observations. Let us choose another pair of observations at E_3, E_4, and M_2, again separated by this period. Always the Earth goes around 1.88 times and in this way Kepler created a number of points on the orbit of Mars By plotting these points (as many as he could get by combining the observations that were at his disposal) he discovered that the path in space which Mars follows in going around the sun was not a circle as the Greeks would have insisted it must be for the sake of harmony and purity and beauty, and so on, but it was actually an ellipse.

There is an important point here to bear in mind: if, under the same circumstances, Kepler had successfully been studying the planet Venus, it just so happens that the orbit of Venus is almost exactly circular and he

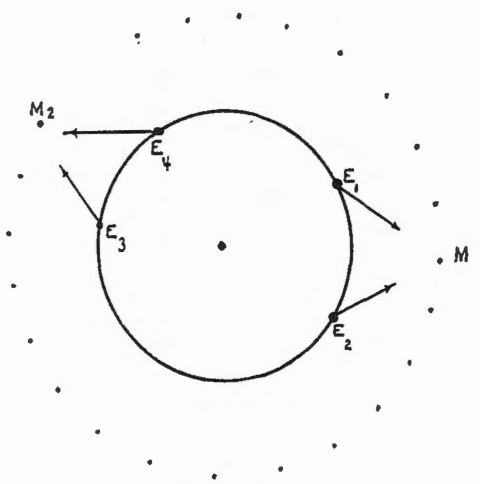

FIG. 2.

would have come to the conclusion that the Greeks were right. But it just so happens that he studied the planet Mars where the deviation from the circle is nearly 10 per cent, and with the methods that he used it was possible to discern this very distinctly, and so he demonstrated that the orbit of Mars is not a circle. Further than that, using the geometry at his disposal he was able to establish that its path is an ellipse, with the sun at the focus of the ellipse. Furthermore, having this clue to the whole process and applying it to the other planets, he discovered that the period of any planet is proportional to the 3/2 power of the distance of that planet from the sun, which means if you are going to count periods in years and if you are going to count the distance from the sun to the Earth as the unit of distance, then the period of the Earth is one year because $A^{3/2}$ is 1, but if you have an orbit which is four times as great as the Earth, then the period will be eight times as long as a year.

You can extend Kepler's third law to any object which is revolving freely around the central body. You can also extend this law to the period of a satellite going around the Earth as compared to the period of the moon which goes around the Earth. The period of the moon is 27.3 days—the period in which it takes the moon to go once around in its orbit through 360 degrees. The length of the month is 29.53 days, but that is because that is a little bit longer than once around for the moon.

On this basis you will find that if you want the period of the artificial satellite to be one day or 24 hours, then taking into account the distance of the moon in miles, if you wish, or whatever units you care to use, the distance of this satellite would then be something like 26,000 miles. This same law can be applied.

Kepler discovered three laws and this is one of the first triumphs in Celestial Mechanics because these laws are reasonably close to the truth. He discovered that the orbit of a planet around the sun is an ellipse. He discovered that the radius vector sweeps over equal areas in equal intervals of time. In other words, when the planet is close to the sun it moves more rapidly along its orbit, and when it is far away it moves more slowly.

Then he discovered this third law, that the pe[illegible] are proportional to the 3/2 power of the mean distance.

NEWTON'S WORK ON MOTION AND GRAVITY

All of Kepler's work was done before Newton's time. Newton, of course, is recognized as having made the outstanding contribution in history to the field of Celestial Mechanics, and what Newton did was to put all of the information which was already in Kepler's laws into the form of calculus. Newton invented the calculus under the stimulus of the challenge of the very problem which I have described to you which was solved in a descriptive sort of way, in agreement with the known observations. This is the beginning of the mathematical attack upon Celestial Mechanics and is embodied in Newton's law of gravitation and Newton's three laws of motion which he formulated as axioms. If you adopt these as being true, then the other things that you deduce can be derived mathematically in sequence.

There are many prominent mathematicians who have worked in the field of Celestial Mechanics since the time of Newton, such as Poincaré, all trying to use the best mathematical analysis and ingenuity which they could bring to bear to put into operation the laws which Newton had set forth in order that they may be used to portray by a mathematical model as accurately as possible the actual state of affairs in the solar system.

I want to emphasize that the law of gravitation enables us to provide a mathematical model which we hope to bring as closely as possible in conformity with the actual state of affairs, and the actual state of affairs is not governed by the law of gravitation. It is the other way around. You are fortunate if the law of gravitation is able to describe the actual state of affairs.

Incidentally, in passing I may say that it was the astronomers who discovered that fact in advance of Einstein's theory. It was impossible, by using the mathematical model that results from the law of gravitation to describe the motion of the perihelion of Mercury. This was known to the astronomers about three quarters of a century before Einstein's theory was first pronounced. I am not sure that Einstein knew about this and it doesn't make much difference, because he was a theoretical physicist and he had an approach to reality which was not the door the astronomers used.

What the astronomers tried to do, lacking Einstein's insight and perhaps his mathematical ability, was to experiment with the assumption that the law of gravity should be 1 over the distance not squared, but they were going to put about six or seven zeros and a one behind the decimal point (2.0000001) and see if that would work, hoping that it would serve the purpose for Mercury and not have much of an influence for Jupiter and the rest of the planets. It turned out that that wasn't so and it didn't work, but this was an experiment in Celestial Mechanics, in about 1880.

HALLEY'S WORK ON COMETS

Another important person in history is Edmund Halley, a contemporary of Newton. He was a genius and if he had not been outshone by Newton he would have been one of the most remarkable characters in history. He was the one who first propounded the idea of having actuarial tables for insurance. He was the first to prove definitely that the stars are moving, that they are not all standing still; and, among other things, following in the lead that Newton had pointed with his law of gravity, Halley worked out the orbits of twenty-four comets which had previously been observed in the history of astronomy. He says, in his own words, that "by a prodigious amount of labor" he arrived at these solutions. At the present time it would take less than a minute, a few milli-seconds maybe on a high-powered electronic computer, to do what Halley did by, in his own words, "a prodigious amount of labor."

Halley was able to prove that the motions of the comets were in accord, if you take a sufficiently general view, with Kepler's pronouncement about the orbits of the planets being ellipses. The orbits of the comets, according to Halley, were parabolas; and he discovered one comet which seemed to be on the same parabola at intervals of seventy-five years. He said: This is not really a parabola. We can't see the whole of it well enough from where we are. It is a very elongated ellipse.

I will give you some idea how elongated it is. If we imagine the distance from the Earth to the sun to be represented by 1 inch, so that the Earth goes around the sun on a circle of 2 inches in diameter, then Halley's comet comes within ½ inch on one side and it goes out about a yard on the other side, where it was in the year 1948. It has now started on the way back and it will be closest to the Earth again in about 1986.

The only portion of this whole orbit that could be observed is in the neighborhood of the Earth; all of the rest was invisible. On the strength of the work which he had done, the demonstrations he had made and the fact that there were three comets that came along this track at intervals of seventy-six years, Halley said that this comet will come back again. I forget his exact words, but he said in effect, "Let it be remembered that it was predicted by an Englishman." I am sorry that he didn't say "Let it be remembered that it was predicted by a Celestial Mechanic."

The point is that this is another illustration of the fact that under Newton's laws of motion and under the formulation of the law of gravity, the motions of all of the bodies around a central massive body will be some kind of a conic section. The simplest case of a conic section is a circle. If you cut the cone at any diagonal angle you get an ellipse, and the more diagonally you cut the more elongated the ellipse is; if you cut it at a certain critical place you get a parabola; and if you cut it by going over still a little bit farther you get hyperbolas.

All of these cases exist. For example, if a meteor from outer space were to come past the earth and the sun, come by the solar system, it would be on a hyperbolic orbit. We do have examples of all of these cases.

NATURAL SATELLITES

Now let us turn to the Jupiter system of satellites. I don't want to say that I predict this, but I am repeating what other people say. They say that we may have a great many satellites of the Earth, and they get carried away with the idea. They don't know quite where to stop, whether they should stop at six or ten or twelve or a hundred. So we may have a great many satellites in the not too distant future. At the present time it is known that the planet Jupiter has twelve satellites which have been observed from the Earth.

Four of these satellites were discovered by Galileo—three of them the night that he first looked at the planet in his first telescope. There are actually four of them that are easy to see. Most of you have probably seen them in the telescope already. They constitute a rather rapidly moving group. The period of the innermost one is 1¾ days; the period of the outermost of these four is 16 days, which is about half of a month of our moon. They lie practically in the plane of Jupiter's equator and they go around practically in circular orbits.

There is one little odd ball in the collection of these twelve moons of Jupiter. It was the first one to be discovered with a modern telescope. This fifth satellite of Jupiter is exceedingly close to the planet and it goes around in less than 12 hours. In this sense it is very similar to the artificial satellites except that it is practically in the plane of the equator whereas our artificial satellites are highly inclined, at least the ones we have had up to now.

The next satellite discoveries—the sixth and seventh—that were made for the planet Jupiter came in rather rapid succession in 1904 and 1905. Ultimately in 1938 another (the tenth) satellite was discovered, which is like the sixth and seventh, in that these are on orbits that are very large compared to the first four. They take about 260 days to go around the planet Jupiter once. They are about seven million miles from the planet and they are all tilted up at about 30 degrees. They immediately presented a problem, namely, to be able to represent the motion of these planets which in 1904 and 1905 was fairly difficult mainly because of the rather large inclination of 30 degrees which their orbits were tilted to the equator of Jupiter.

In 1908 the eighth satellite of Jupiter was discovered and since that time there have been three other ones like it—the ninth, the eleventh and the twelfth. These are on orbits which are about twice as big as the group comprising the sixth, seventh and tenth. They are about fifteen million miles from the surface of Jupiter, but they are exceedingly eccentric.

In the extreme case the distance of the satellite from the center of its orbit to the focus where Jupiter is, is four-tenths of the total radius of the orbit. This is an exceedingly high eccentricity of 0.4. There are now four of these satellites that have been discovered, and the other outstanding thing about them is that they all go around backwards, in the sense that is called retrograde motion which is contrary to the revolution of the planets around the sun and the revolution of the other satellites around the planets. They present an exceedingly difficult problem which still hasn't been solved with any high degree of satisfaction and which we are planning to work on at the Cincinnati Observatory if we ever get any time off from the Sputniks and other artificial satellites.

I might give you this sidelight. About three or four years ago we undertook to investigate the tenth satellite of Jupiter, which is the newest member of the intermediate group. Frankly, the reason was that we had set out to investigate the artificial satellite and methods by which its orbit should be worked out. Not knowing what the security requirements of the artificial satellite would be—the tenth satellite of Jupiter had already been discovered by astronomers before the security officers discovered it—we were going to use the tenth satellite of Jupiter as a kind of security blind if it was necessary.

Without trying to be facetious, the method which we were bringing to bear on the tenth satellite of Jupiter, which has this large inclination and which has resisted a solution to the problem in prior years, is exactly the same method that we wish to bring to bear upon the artificial satellites, because while the Vanguard satellites are being fired at an angle of about 35 degrees inclination, the Russian satellites were tilted at 65 degrees and there is interest in satellites which will go over the Pole. So, we wanted to have methods which would be sufficiently general and sufficiently powerful that we could do a good job in both cases; and it turned out that this is what happened.

CONDITIONS BEYOND THE ATMOSPHERE

One point I would like to get across is that in this whole approach to space problems it is necessary to recognize that these things are outside of your normal experience. The main thing which goes on in your normal experience as far as mechanical things are concerned, is friction; and out in empty space there is no friction.

Out where the artificial satellites are there still is friction and that is one of the purposes of the satellite experiment—to determine the density of the atmosphere. But the things I have been telling you about which are astronomical are outside the range of any atmosphere. There is no resistance, and therefore the whole normal mode of experience is different from anything you are accustomed to or are familiar with.

I would like to give you, therefore, some kind of picture which will enable you to think in terms of empty space and to understand why things are as they are out there.

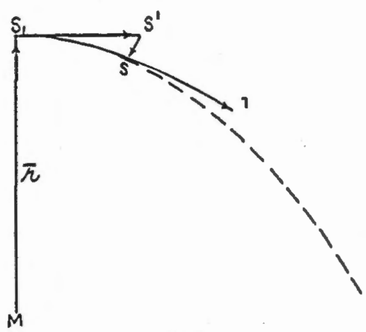

FIG. 3.

EFFECT OF VELOCITY ON SATELLITE ORBITS

Suppose that we have here a massive body, M in Fig. 3, which you can think of as being the center of the Earth if you wish or you can think of this as being the sun, in which case S_1 is some planet or some minor planet; if M is the Earth, then S_1 is an artificial satellite if you wish. This is the position vector of the satellite, MS_1, which simply says where the satellite is in space.

S_1S' is intended to represent the velocity vector which the little particle at this point has when it is at this position, S_1, so that if there were no attraction to the center, if only the law of inertia were operating, the particle would move out along this velocity vector, S_1S'. But if M is the center of the Earth, then in addition to the velocity along this velocity vector there will be a certain amount of fall of exactly the same kind that takes place when I let go of this piece of chalk.

As soon as I let go, the attraction of the center of the Earth draws the chalk down. Until I let go it is not the hypothetical particle we are talking about and so far as we are concerned sitting here in this room this piece of chalk has no velocity vector. The velocity vector shrank to zero If we were to get out in empty space and watch the rotation of the Earth, then the chalk has a velocity of nearly a thousand miles an hour. We are all a great deal farther away from where we were when I started, except we don't realize it.

In the small interval of time that S_1 would move from S_1 to S' it would also drop an amount $S'S$, and these two things are going on simultaneously, so as it begins to move out along the velocity vector it also drops down a little bit; and as it moves out a little bit farther it drops down a little bit more.

The point is that instead of the particle winding up at S', it has wound up at S as a result of a combination of the law of inertia and the amount of gravitational pull that it had. Now, from the way I have drawn the figure, if the attraction were just instantaneously cut off at S, then the particle would continue out along the velocity vector ST according to the law of inertia. But if the attraction is not instantaneously cut off, what happens? The attraction is toward the center, as before; the velocity is now ST instead of S_1S', and the whole process may be repeated.

I don't tell you from this diagram whether the arc is a hyperbola, a parabola or an ellipse, but it is a conic section if you do it according to the law of gravity.

ORBITS OF ARTIFICIAL SATELLITES

Now I am going to draw another figure (Fig. 4) under the same set of rules, but I am going to start out with the velocity vector S_1S' not being quite so long. In other words, we are considering now a second case in which the particle did not have as much velocity to start with as it did in the first instance. The attraction is still the same, so the amount that it is going to fall from S' to S is still the same and the curve is going to be steeper. This time the distance travelled in the same length of time is less, namely S_1S. Also the resulting velocity vector ST is shorter.

When does this curve become a circle? You have to have a very delicate balance between the amount of attraction, $S'S$, toward the center of the Earth (if we want to continue to talk about artificial satellites), which depends on how far away it is from the center of the Earth, and the velocity, S_1S' which you have, such that the combination of the two of them is exactly an arc of a circle from S_1 to S. If this delicate balance has been achieved and S_1S is an arc of a circle, then when the process is repeated, beginning again at S, it will continue to be an arc of a circle.

This is the situation which one always faces at the launching of an artificial satellite. At the position which the missile has when the last propellant is burned out, what is the length (and direction) of the velocity vector? Is it too short, as in Fig. 4? Is it exactly right for a circle? Or is it too long?

Now, this is where Von Braun succeeded in his project. He didn't even try to make it a circle. He just put as much velocity into this thing as he could get from his rockets up above the atmosphere. What did he get? He got a satellite which went way out, way down on the far side of

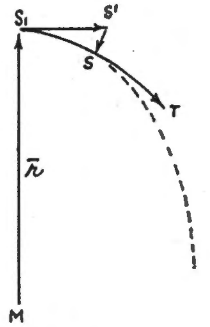

FIG. 4.

the diagram, and back up again to S_1. MS_1 was 220 miles above the surface of the Earth, and on the other side of the orbit it was about 1600 miles. The reason was that there was no attempt to balance the velocity. They just shot for all they were worth, and that is what comes of it—such an eccentric orbit.

As a first crude approximation, the orbit of an artificial satellite is an ellipse around the center. It is a crude approximation primarily because the whole attraction of the Earth cannot be considered to be the attraction at a point. If the entire Earth were composed of material exactly symmetrically located with respect to the center of the Earth, then the entire Earth could be considered to be the same as a massive point at the center. If that were the case, then the orbit of the satellite, the ellipse of the satellite, would remain fixed so long as there is no friction, that is, so long as the orbit were so big that it would be above the friction of the Earth's atmosphere.

However, that isn't the state of affairs because the Earth is turning on its axis and a point on the equator goes at the rate of a thousand miles an hour as the Earth turns on its axis. This is one thing that people are completely unaccustomed to visualizing because you have spent your whole life on this rotating Earth; you are used to it and you don't feel it, but it is going on. The Earth is not spherical. It bulges out at the equator such that the diameter is 27 miles greater than it is at the Poles. This means that the mass of material is not uniformly distributed, and this raises one of the two principal problems in connection with the artificial satellite.

PERTURBATION

Oblateness

If you allow me to draw an imaginary axis through the Earth and if you allow me to take out from the Earth all of the mass which is symmetrically distributed and concentrate it at the center, then what we have left is a kind of doughnut-shaped representation of excess mass around the equator, known as the "equatorial bulge" or equatorial protuberance. This material produces what in Celestial Mechanics we call perturbations upon the simple elliptic orbit. If any simple elliptic orbit were put up in the form of an artificial satellite, it would remain fixed but because of this excess mass it is unable to remain fixed because it is subjected not only to the attraction which is at the focus of the ellipse, but also to this additional attraction, which produces the oblateness perturbation.

Those of you who are interested in mathematics and who know something about it would write

$$V = -\frac{M}{r}\left(1 - \frac{C^2}{2r^2}P_2(\theta) + \ldots\right).$$

P_2 is the second Legendre polynomial. V is an expansion for the Earth's potential in spherical harmonics. The 1 is the principal term which would give you $1/r^2$-attraction for simple gravity. C^2 is of the order of magnitude of one part over 300, or 1/297 to be more exact. The remaining term is represented as the equivalent of this equatorial protuberance or the mass which causes the motion of the orbit planes. If this expression were put equal to zero, we would have a simple case, but in practice this doesn't stop here. There could be a plus, and so on, and you can have the fourth spherical harmonic and go as far as your observations will give you any information.

Now several things happen as a result of the oblateness. In Fig. 5 we have shown one octant of the rotating Earth and the initial position of the satellite orbit in space. The closest approach to the center of the Earth is assumed to be at P. The intersection of the orbit plane with the equatorial plane is called the node, N. This point moves westward so long as the inclination of the orbit is less than 90°. In the case of Explorer I the node moves at the rate of about 4¼° per day. Similarly, the perigee, P, moves forward in the orbit plane so long as the inclination is less than 63¼°. Again, for Explorer I this is going on at the rate of 6¼° per day. At this rate in two weeks' time the perigee moves through 90°.

I want to point out to you a little consequence of that which you probably hadn't thought about. When they fire the rocket, as you probably have read, the control system in American satellites is to spin the rockets so that if they fire off to one side, if there is an uneven balance in the firing chamber, this gets rotated sufficiently rapidly that it averages out and doesn't do any serious harm.

The rockets are rather rapidly rotated or rifled, as they are shot out. There are no fins, for the assembly is rotating so rapidly that fins wouldn't serve their usual purpose. As the rocket assembly goes around, the direction in which the missile itself—the satellite instrumentation and the

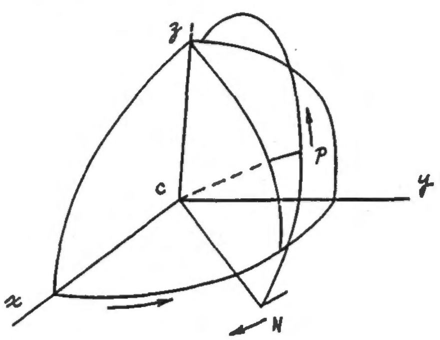

FIG. 5.

empty rocket chamber—points remains parallel to itself as it goes around. It is highly rifled and goes out to where there is practically no atmosphere and no resistance; then it comes back and bullets its way through the dense atmosphere when it gets down closest to the earth, about 220 miles above the surface of the earth.

In two weeks' time the orbit turns through 90°, as a consequence of this perturbation. When the satellite goes through the lowest atmosphere, it is going through broadside and so it produces or presents a maximum broadside face toward the resistance of the atmosphere.

I have given you the two extremes—when it bullets through and when it goes through broadside. In the meantime the orbit has been slowly turning; this broadside effect has gradually been setting in and so the center of gravity in the vehicle is forward (because the back part is now the empty rocket shell where the fuel was) and as it hits the low atmosphere broadside it would tend to be tumbled backward. This low density volume at the back would act as a fin. But because it is highly rifled it is not going to tumble this way in response to that broadside hit, but it is going to precess. So that the Explorer satellite has the most god-awful wobble you can possibly imagine!

You may well laugh, but to us this is a very serious business because it prevents us from making reliable predictions about the motion of the satellite, as we try to project it several weeks into the future, simply because it is impossible to observe what is going on. The satellite is only seven feet long—too small to see—and you have no way of knowing what it is doing. You can't tell what the average broadside is that it presents, so you have no way of predicting what the drag is going to be.

Drag

I have tried to show you descriptively that perturbation amounts to a regression of the node. It also amounts to an advance of the perigee, as the point closest to the center of the Earth is called. There is a force exerted on the vehicle in addition to the force of gravity. It is the force which is known as drag, atmospheric drag, and it is providing a resistance to the forward motion of the satellite.

This resistance has less effect when the mass is high, as it was in the Russian satellites. It also has less effect if the cross-sectional area is small. But the reason that the Vanguard satellite is spherical is that no matter how it would tumble and twist and turn or be rifled, if it is a sphere it must present the same cross-sectional area.

I emphasize this point not because I am associated with the Vanguard project. The main problem for which these satellites were intended to provide information is the density of the high atmosphere and with such a peculiarly-shaped object as the Explorer satellite, you are just at a loss to obtain the information. This is really the reason why the Vanguard satellite was designed to be spherical, and despite all the difficulties encountered in putting it up I hope that there will not be so much public acclaim for the success of putting up any object at all, that we shall ultimately lose sight of the benefit of the spherical satellite which will always present the same cross-section to the atmosphere and, therefore, it will not tumble and it will permit us to determine something about the density of the atmosphere. I doubt very much whether information about the atmosphere will be determined unless we do get a spherical satellite.

I want to describe one other thing to you in sort of a pictorial way. Suppose we come back to the state of affairs where the satellite is launched so that it goes out in an orbit which is very high on the south side, with a perigee of 220 miles and an apogee of 1600 miles. The first time around the satellite has whatever velocity the rockets were able to impart, and that is what produced the orbit. But the next time around the satellite is going to be subjected to drag as it gets into the low atmosphere and, therefore, drag is going to cause some kind of trouble.

What kind of trouble can you cause the satellite? All you can do is take some of its energy away from it. It has nothing else. It is a ballistic missile. It has no way to control itself. It is just coasting around The only way the satellite can be adversely affected is to steal some of its velocity vector.

The first time around the drag has the effect of stealing a little bit of the velocity vector so that the second time around the velocity vector is a little bit shorter than it was. As a result, the orbit will not be any closer to the earth at the perigee, but if it loses some of its velocity (the attraction to the center will be the same as it was because the distance is the same) it will come on a curve which is slightly inside of where it was and on the far side it won't swing out quite so far. Then it will come back in for the next time around, and at the low point of the atmosphere the atmosphere will steal a little more velocity away from the satellite. The orbit, therefore, keeps getting smaller and smaller.

When the total diameter of the orbit is smaller, (half of the total diameter is the quantity A which controls the period, according to Kepler's third law), so that from revolution to revolution, as the A gets smaller, this causes the period to get smaller, in other words, to get shorter. At the present time, on the basis of the observations which we have at our disposal, it appears that the period is getting shorter by, roughly, 0.005 minutes per day. This shortening of the orbit and the consequent shortening of the period are due to atmospheric drag.

Furthermore, the distance from the perigee (the place closest to the Earth) to the center of the Earth is changed very little but the distance from the apogee (the far side of the orbit) to the center of the Earth is getting smaller all the time, so that the eccentricity is also getting smaller. That is, the orbit is getting more and more circular.

Ultimately, the apogee comes down such an amount that it is also in the atmosphere of the Earth, and the satellite object meets with the same resistance as at the perigee. By that time the orbit is practically a circle—and there is no such thing as getting more circular. As a matter of fact, there is a point where the process of becoming circular stops. What is the state of affairs if the velocity vector is shorter than the one required for circular orbit?

What happens is this: The object does not have enough velocity to swing out so that it is much lower on the far side than it is on the near side, and the original apogee becomes the perigee in that case. As soon as drag has set in to such an extent that the orbit is nearly circular, then the velocity is being slowed up at every point in the orbit and the orbit is spiraling inward. If any of you are on the Moonwatch teams you know that this happened after the time that Sputnik I got down to a period of about 90 minutes; then practically all of the orbit was in the reasonably dense part of the atmosphere, the resistance was going on continually and shortly the satellite came down, nobody knows where. It was no longer to be seen because the resistance reduced the velocity so greatly that the satellite just dropped.

I want to make one point more before I close. In all of this discussion there is one other equation that you can use if you are interested in dabbling around and experimenting with these things, and that is the very important equation: $V^2 = \frac{2}{r} - \frac{1}{a}$. V is the magnitude of the velocity vector, r is the distance from the center of the Earth, and a is the semi-major axis of the ellipse, or half of the diameter of the ellipse which is the same as the one in the formula for Kepler's third law.

This equation tells you that if you have a very special velocity (and what that velocity has to be depends upon the mass of the central body and the distance that you are away), such that you would get a circular orbit, then $V^2 = 1/a$. In other words, r is equal to a because no matter how far you are from the center of the Earth it will always be the radius of the circle. That is the special velocity that you have to have or you can say it is equal to $1/r$, which is the same thing in this case.

If you have less than this velocity, and if the two vectors are at right angles to each other, then you are at the apogee of the orbit. If you have more than this velocity and the two vectors are at right angles to each other, then you will be at the perigee of the orbit and the orbit will swing out on the far side. If the two vectors are not at right angles it is a little more complicated and beyond the scope of this lecture.

METHODS OF TRACKING ARTIFICIAL SATELLITES

There are two ways of dealing with perturbations. One of them is the step by step process whereby you have the formula for the second derivative; you estimate what the position of the satellite is so that you can compute the attraction; and you also estimate what the velocity is so that you can compute the drag and in this way you can estimate what the position would be at the end of the next interval of time.

In our case, with the programs that we have prepared for the satellite computations, we do this at intervals of one minute and we perform what is known as numerical integrations. We can do it in several ways. One of them is directly on the position vector.

There is a more general way of dealing with this problem of the oblateness (but not of the drag), and that is to express what those of you who are mathematicians will recognize as Fourier series; and we have a scheme whereby we can compute what these periodic series should be as they apply to the satellite. The principal period, of course, is the period in which the satellite goes around its orbit. The second harmonic would be a term which goes through a complete cycle in just half of a revolution of the satellite. These are the most sizeable; these are the largest terms of the perturbation because the path that the satellite experiences above the equator is practically the same as the one it experiences below the equator, at least to a first approximation, and therefore you have the second harmonic terms rather important in this Fourier series expression.

These are what are known as the general perturbations because they are not restricted to any interval of time. You can simply specify a time and starting from there for that single time alone you can compute the position of the satellite at any time in the future, using the numerical integration method. For use on the electronic computer, this is also a very powerful method.

PROBLEMS TO BE SOLVED

I want to mention one other thing along these same lines and it is still a reasonably unsolved problem. The method by which it can be done has been pointed out more than a century ago by a mathematician and astronomer named Delaunay. This is an even more powerful approach in which you solve the orbit once and for all, and when you get finished you will have elaborate formulae in which the quantity A appears, the quantity E (the eccentricity) appears, the quantity I (the inclination) appears, and you simply substitute the values for whatever particular orbit you have.

There are little technical difficulties in the numerical integration method, such as getting small divisors when you integrate, which present difficulties for this method in special cases. Those difficulties are overcome in Delaunay's method, and in the case of Jupiter 6, as I mentioned to you before, it was pointed out that the methods were inadequate because of the large inclination.

We are now back to that same point. Because of the high inclination satellites, we are presented with the problem of developing Delaunay's method in such a way that it is not restricted by the high inclination. This can be done. It is just a piece of work that has to be done. It hasn't been done yet.

There are several other problems in Celestial Mechanics. One of the most challenging is to try to express the motions of the major planets, particularly Jupiter and Saturn, as they go around the sun, for exceedingly long intervals of time such as 100,000 years or more. At the present time no one has succeeded in doing this. The orbits which we have for the major planets are good in terms of a century or two or three centuries, but not for thousands and hundreds of thousands of years. Here, then, is a challenging problem which certainly must take off from the methods which are known now.

I have tried to give you some kind of a panoramic array of some of the details of Celestial Mechanics at this very minute, and some of the broad challenging problems that face us for the future.

THE EXPLORER

THE EXPLORER STORY AND THE TEMPERATURE CONTROL OF THE SATELLITE PACKAGE

BY

GERHARD HELLER[1]

This series would not be complete without the story of the U. S. Satellite, Explorer I, this country's first "step into space."

The last time I talked on the Army satellite was about a year ago at a secret meeting of the Army Science Symposium. Now that the situation has improved, and we can tell the story publicly, I will give you a description of the missile, its payload, and some of the obtained results.

It is needless to say that the Explorer started the space age for the United States. We no longer discuss whether we should explore space or whether we should have manned space vehicles and eventually manned space flight, for everyone has accepted this as a matter of fact. It is definitely not something for a future generation. We are in the space age now.

It is generally accepted now that the exploration of space by man is an objective in itself. We do not need any more explanation or justification than this.

Space science is advancing so rapidly that science fiction writers are having quite a bit of trouble. The best thing for them to do is to find a new field because science has now caught up with them.

STEPS OF SPACE EXPLORATION

Before going into the detailed story of Explorer I, I would like to say a few words about the stages of space exploration as I see it. First, we will use ballistic missiles with a high degree of reliability and proven recovery techniques that have been successfully applied for several years, both for instruments and, eventually, man.

The next step will be manned flight over great distance, say a thousand or more miles, and only after manned flights in rockets have been made and a high chance of survival has been realized should we proceed to flights with orbital speed and maneuvers in space. Only after gaining sufficient experience and learning enough from these flights, can we think of establishing a space platform. Proceeding simultaneously with activities directly associated with manned flights, there will be many instrumented flights for the purpose of perfecting

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both vehicles and recovery techniques, as well as obtaining measurements of the space environment. Eventually, there will be manned exploration of our solar system with landings on other celestial bodies.

I would like to emphasize the fact that we should use highly reliable ballistic missiles at first. This is specifically important if we think of entrusting the lives of men to such vehicles.

EARLY HISTORY OF THE EXPLORER

The story of the Explorer is one of a dream that came true.

In 1953, a few space enthusiasts got together and said, "We have many reports, many ideas and projects about space flight. Is there any chance that we can do something about it in the near future, or is it still a dream of ten or twenty years?"

At this meeting the Explorer idea, or "Project Orbiter" as it was called at that time, was born. Dr. Wernher von Braun presented his idea to combine the Redstone vehicle, which was in the advanced research and development stage, with existing solid propellant rockets. The first project that came out of this proposal was very modest—to put five pounds of satellite into an orbit, using available hardware, just to prove the idea and to do it as cheaply as possible.

This idea found general acceptance among space enthusiasts, including, at that time, members of all three Armed Services. It was planned as a joint undertaking, and was started as such, each branch having its role in the undertaking. The project got under way using a Redstone as the first stage booster with three upper stages of solid propellants.

At the end of 1954, the original booster and upper stage performance were improved to the extent that the payload weight went up to 15 pounds. After this promising start the project was suddenly discontinued as an active satellite project.

During 1955, however, the Army was assigned the development work on the Jupiter IRBM. One of the difficult problems was the re-entry of the warhead into the atmosphere at supersonic speeds after traveling the full range of the missile. For this task, the original four-stage vehicle found an immediate application, although, as it turned out, only three stages were needed. Still, the basic work done on the Explorer prior to 1955 could be utilized for the Jupiter IRBM.

In 1956, such a Jupiter-C missile went over a range of 3600 nautical miles. The configuration of this vehicle was similar to that of the original "Orbiter," even to the inclusion of a dummy fourth stage. The charge was taken out of this stage because it was to be a re-entry vehicle, not an orbiting one. Therefore, the 1956 model, essentially a three-stage vehicle, was actually a test vehicle for the later Jupiter-C flights with simulated nose cones, fired over the full range of the IRBM.

Several of these re-entry flights were completed successfully prior to

October, 1957, when the first Russian satellite was put into orbit. In November, the Army Ballistic Missile Agency was assigned the task of dusting off the Jupiter-C and modifying it to carry a scientific IGY satellite. It took only 84 days from the date of the assignment to the successful firing at the end of January, 1958. The complete vehicle, which has a total length of 68½ feet, is shown in Fig. 1. The Redstone

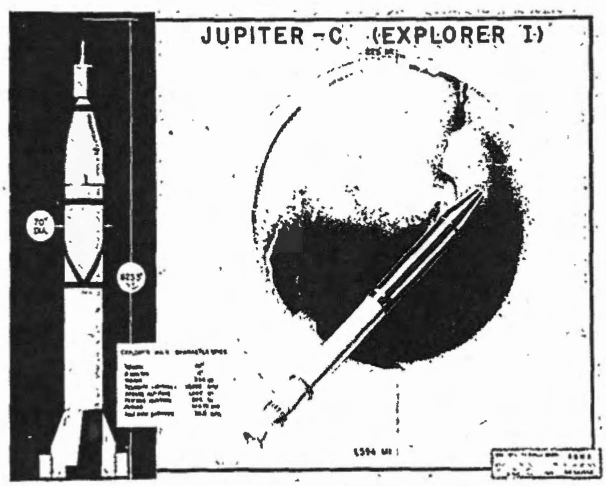

FIG. 1.

booster, with a diameter of 5 feet 10 inches, has been elongated to provide additional tank space for both oxygen and fuel. In this case the fuel used was a high performance propellant called Hydine that increased the specific impulse considerably.

The instrument compartment contains control equipment and measuring instruments necessary for the flight performance and quite a number of instruments that are necessary for flight safety purposes.

At the forward end of the instrument compartment is the rotational launcher which is spun up by two electric motors located inside the instrument compartment. The rotational launcher or "tub" contains the three high speed stages of the solid propellant cluster. The second stage of the vehicle is a cluster of eleven Sergeants inside of which the third stage (three Sergeants) is telescoped. The fourth stage (a single Sergeant located above the tub) is topped by the 18¼-pound payload.

The fourth stage, 80 inches long and 6¼ inches in diameter, weighs 30.8 pounds and has a perigee velocity of 18.400 miles per hour. The perigee altitude is 225 (statute) miles, and the apogee altitude is 1594 (statute) miles. The initial period is 114.78 minutes, and the inclination to the equator is 33.3°.

TRAJECTORY

The missile takes off vertically from the firing table and is then tilted toward the flight direction. It follows a ballistic trajectory during the 2½ minutes of powered flight. The cut-off precedes the separation of the instrument compartment from the booster by a few seconds. Within the following one-half minute, the instrument compartment is tilted into the correct firing position. This is accomplished by four pairs of supersonic air jets, whose nozzles are attached to

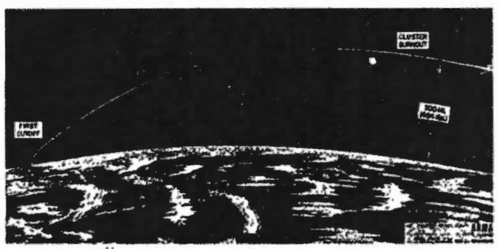

FIG. 2. Typical ascent trajectory (phase 3 orbital carrier).

the rearward end of the instrument compartment. The thrust characteristic of these nozzles is proportional to the linear displacement of centrally-located control needles. The firing position, about 95° from the vertical, compensates for the curvature of the Earth and leaves the velocity vector after the fourth-stage burn-out parallel to the surface of the Earth. Total free-coasting time from cut-off to upper stage ignition is 4 minutes 10 seconds. The upper stages are ignited about 15 seconds before the apex of the booster trajectory is reached by means of a manually-operated computation device and a command link. It uses the weighted results of the following measurements as input:

1. Cut-off velocity obtained through booster telemetry;
2. C-band radar evaluation of velocity vector at two points of the free-coasting trajectory;
3. S-band radar data, evaluated by an IBM 704 computation program, the computation being performed during the flight; and
4. Doppler frequencies of Dovap transponder of booster.

The three solid propellant stages are fired in a rapid sequence, the third and fourth stages being ignited by an automatic timer. Burning time is approximately 6 seconds for each stage, with an 8-second interval between the ignition of the stages to allow for complete burn-out of each stage and for temperature dependency of burning time. The solid propellant stages are stabilized by spin that is imparted prior to the launching and then increased from 450 to 750 rpm during the boosting flight. A dispersion analysis showed that with this method the angular deviation of the cluster will not exceed 1.5° circular probable error in 95.5 per cent of all cases (2 sigma value).

The error angle of Explorer was less than 1°, well within the prediction.

The trajectory of the Explorer is shown in Fig. 2.

LAUNCHING THE EXPLORER

The Explorer undertaking turned out to be a combined three-Service affair, almost as it was originally visualized. The vehicle itself was developed by the Army combining the capabilities of ABMA and the Jet Propulsion Laboratory—the latter working specifically on the upper stages. It was launched from the Missile Test Center, Cape Canaveral, Florida. The Navy tracking facilities and computation centers, set up for the IGY, were utilized.

The loading of the booster into a C-124 Globemaster for transportation from the AMBA to the Missile Test Center is shown in Fig. 3. Figure 4 was taken at the Center as the booster and instrument compart-

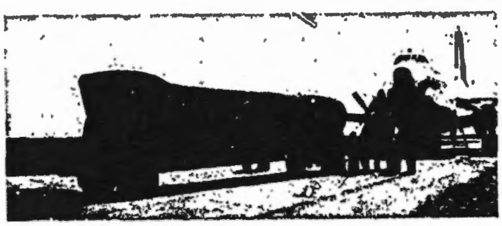

FIG. 3. Booster of Jupiter-C ready for loading into C-124 Globemaster for shipping to the Missile Test Center, Cape Canaveral, Florida.

ment were being checked out in a hangar. In Fig. 5 the two sections are suspended and ready to be bolted together. Figure 6 shows the fourth stage being checked out on a spin balancing rig. The missile, surrounded by working platforms, is erected on the launching site in

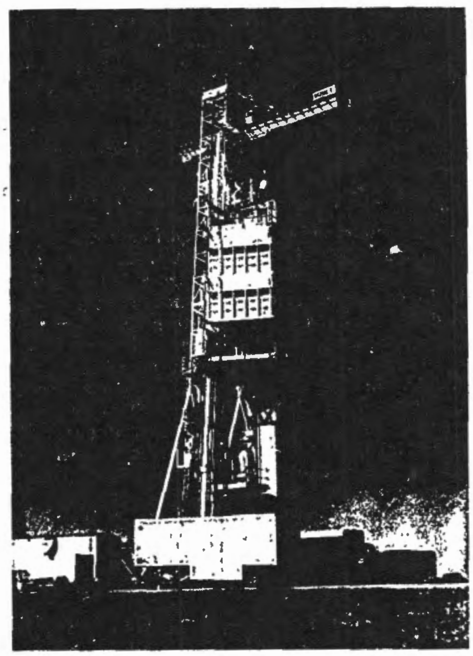

FIG. 10. Jupiter-C Missile (Explorer) on launching pad at the Missile Test Center, Cape Canaveral, Florida.

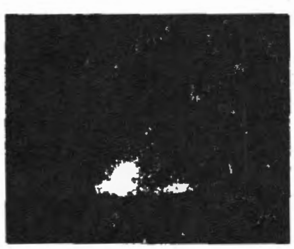

FIG. 11. Jupiter-C missile (Explorer) at take-off.

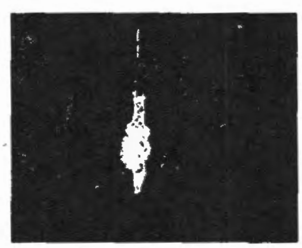

FIG. 12. Jupiter-C missile (Explorer) on its way.

Fig. 7. Figure 8 was taken as the spin launcher and the three solid propellant stages are checked out in a dynamic balancing rig, and Fig. 9 shows the enclosed upper platform with the fourth stage ready to be mounted on the missile. Figure 10 shows the Explorer on the launching pad at the Missile Test Center.

Figure 11, taken at X minus zero, shows the flame spreading out from the base of the missile as it takes off. In Fig. 12 the missile is on its own heading for space.

During the launching of the satellite I was in the telemeter room with Dr. E. Stuhlinger. We followed the booster flight by looking at the telemeter data that were recorded and displayed on instruments These were exciting minutes, any slight movement of the instrument needles was followed with apprehension. The missile worked perfectly. After cut-off of the Jupiter-C main stage, we obtained the separation signal, that indicated that the instrument compartment had separated from the booster. There followed four busy minutes in which data obtained from the C-band radar station were obtained and evaluated for apex prediction. This was one of the input data used for the apex prediction and the determination of the firing time. Then followed the ignition and the confirmation that the signal had come through and the upper stages were burning. We knew that 22 seconds after the ignition signal the first U. S. satellite would be in orbit, provided that everything worked all right. A rapid evaluation of tracking data received at Cape Canaveral and the Antigua station was made to determine the initial conditions, and after one hour the confirmation came that Explorer I was in orbit.

INSTRUMENTATION

Figure 13 shows the details of the instrumentation of Explorer I, which consists of the empty fourth stage motor case and the payload proper. The stainless steel shell is shaped like a cylinder and topped

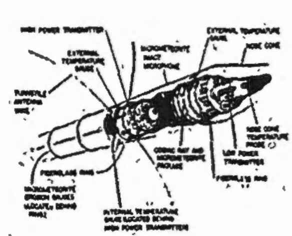

FIG. 13. Explorer I cut-away of instrumentation.

FIG. 14. Antenna array of Microlock station.

by a rounded cone. This shape has been chosen in order to fulfil design requirements for easy packaging as well as temperature requirements during both the orbiting and the boosting phases of the vehicle.

The instrumentation consists of: (*a*) a high power transmitter located at the rear of the payload (with a continuous transmitting power of 60 milliwatts); (*b*) a low power transmitter inside the nose cone, operating at 10 milliwatts; (Both are Microlock transmitters developed by the Jet Propulsion Laboratory of the California Institute of Technology) (*c*) the main measuring device, a Geiger tube for cosmic ray counts, located at the rear of the Cosmic Ray and Micrometeorite package, developed and built by Dr. van Allen of the State University of Iowa (each of these instruments has an independent battery package); (*d*) a nose cone temperature probe; (*e*) an external temperature gauge located at the Fiberglas ring between cone and cylinder; (*f*) a temperature gauge on the low power transmitter (not shown); (*g*) an external temperature gauge located at the Fiberglas ring between the payload and the Sergeant motor; (*h*) an internal temperature gauge located on the high power transmitter; (*i*) a micrometeorite erosion gauge located behind the rearward stiffening ring of the motor; and (*j*) a micrometeorite impact microphone located on the cylinder.

The antenna arrangement consists of: (*k*) a Fiberglas ring between the cone and the cylinder as a dipole gap for the low power transmitter; and (*l*) four turnstile antennas consisting of steel wire with a steel ball at the end mounted on a Fiberglas ring between the payload and the motor. The transmitters operate at 108 and 108.03 megacycles, respectively, and serve for both telemetry and tracking purposes. Signals can be received both by the Minitrack network established for IGY and by the IGY-Microlock stations such as the one shown in Fig. 14. Telemetry is accomplished by using phase modulation.

TEMPERATURE CONTROL

The instrumentation of Explorer I operates properly only in the particular temperature range for which it is designed. The lower limit, 0° C., is determined by the efficiency of the mercury batteries, which decreases rapidly if the temperature drops below this mark. The upper limit of the temperature is set at +65° C. by the transistors of the electronic equipment. However, in the evaluation of the temperature control problem, the upper and lower limits were not considered equally important. No permanent damage is incurred if the temperature drops below the lower limit and then rises again, for the power output would decrease or drop out only until the temperature rises again. The upper limit is more serious, because, if it is exceeded considerably, permanent damage is incurred. The desired temperature was, therefore, set at about 20° C., with foreseeable deviations not expected to exceed ±25° C. This choice would guarantee functioning for an extended time even if the deviations exceed ±25° C. The problem of temperature control and the measured results are treated in some detail in this paper, because the author worked on these problems. It was part of the missile assignment to provide the proper temperature range for the functioning of the scientific equipment flown under IGY auspices in the first U. S. satellite.

Environment of Satellite

The temperature of a satellite is mainly determined by a radiation equilibrium. In most cases aerodynamic heating is negligible. This can be shown by Table I, if the heat flux of the last column is compared to the solar constant of 1200 Kcal/m^2h.

TABLE I.

Altitude	Density (Smithsonian) $\frac{(kg\ sec^2)}{m^4}$	Heat Flux by Aerodynamic Heating Kcal/m^2h Circular orbit	Average Heat Flux by Aerodynamic Heating for Orbit with Eccentricity $e = 0.13$
100	8×10^{-9}	170,000	17,000
150	6×10^{-10}	1,300	130
200	7.5×10^{-11}	60	16
250	1.8×10^{-11}	39	3.9
300	7×10^{-12}	15	1.5
350	3.3×10^{-12}	7.1	0.7
400	1.7×10^{-12}	3.7	0.4
450	1.0×10^{-12}	2.2	0.2
500	6×10^{-13}	1.2	0.1

For eccentric satellites with a perigee of 250 km or higher, the heating effect of the interaction of the satellite with air molecules can be neglected. The thermal environment is determined by radiation coming from the sun and the Earth. Figure 15 illustrates the radiation household of the Earth. The incoming radiation from the sun, or solar constant, is 100 per cent. However, 36 per cent of this radiation leaves the atmosphere because of reflection and scattering. This percentage of the radiation called the albedo of the planet Earth, has basically the same spectral distribution as the incoming radiation. Since the albedo varies with the cloud cover, land and sea areas, a variation between 24 per cent and 54 per cent has been used for temperature considerations. Using the average value of the albedo (36 per cent) we find that 64 per cent of the solar radiation has been absorbed by the atmosphere and the Earth's surface. This energy is re-radiated as infrared light, 17 per cent being radiated directly from the Earth and 47 per cent from the atmosphere. There are slight differences in the spectral distribution of these infrared radiations due to day and night, latitude, and seasons; however, they are close enough together to allow us to use the spectral distributions of a black body at 255°K. as a mean value.

Summarizing: The satellite temperature is determined by four modes of radiation—(*a*) direct radiation from the sun; (*b*) reflected solar radiation; (*c*) infrared radiation from Earth; and (*d*) infrared radiation from the skin of the satellite. We have to place the satellite within this given environment and operate it for two or three months without exceeding the temperature limits imposed by the instrumenta-

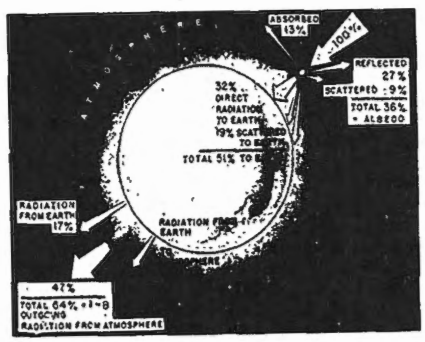

FIG. 15. Radiation equilibrium of Earth and atmosphere.

tion. Maximum temperatures that could occur in this environment are 400 to 450°K. and minimum temperatures of about 200°K. can be expected. The problem is to find out whether this range can be reduced to meet the instrument requirements.

In order to do this, a study has been made of all parameters that influence the temperatures and the variations of these parameters that can be expected during the two or three months of orbiting time.

First we have to consider the orbital characteristics which can be expressed by the parameters (in equatorial spherical coordinates):

i = inclination of orbit to equatorial plane,
e = excentricity of orbit,
R_p = radius of perigee,
ω = argument of perigee,
Ω = right ascension of ascending node, and
M = mean anomaly from injection point of satellite to initial perigee.

The inclination, i, is determined by the firing azimuth and the geocentric latitude of the firing site (which for the Missile Test Center is 28.45°). The azimuth was 110° from true north. Variations of the inclination are due to angular dispersion in yaw of the high speed stages, errors in booster trajectory and cut-off flight direction, and changes in final velocity.

It turned out that the variations of i are important in the determination of the time the satellite is in sunlight.

The eccentricity e is determined mainly by the altitude of the booster apex, the injection velocity after fourth stage burn-out, the error angle in pitch due to timing error of upper stage ignition and cluster dispersion in pitch. The eccentricity is also dependent on time after launching because the friction caused by interaction with air molecules at perigee of each revolution decreases the eccentricity.

The radius of perigee is equal to the booster apex velocity for zero error angle in pitch. Any error angle decreases the perigee altitude. The effect of various perigee altitudes on the long time temperature balance was studied. The argument ω of the perigee is the spherical angle measured from the ascending node to perigee in the orbital plane. For a given set of booster characteristics and upper stage performance, ω is a constant if the pitch error angle is zero. The influence of variations of ω has been studied in connection with simultaneous changes of e and R_p. The right ascension Ω of the ascending node, a function of the day and hour of the launching, is an important factor because it determines the plane of the orbit in space and thus its inclination to the ecliptic and the angular distance to the position of the sun within the ecliptic. Changes of Ω through 360° have been studied. In addition to the variation of the initial values of ω and Ω, it has to be considered that these angles are time dependent. The change of ω, called the progression of the line of apsides, is 6.41° per day for Explorer I. The change of ω or the regression of the nodes is −4.27° per day.

The mean anomaly M from injection point of the satellite to the perigee of the first revolution is proportional to the time difference between injection point and perigee. For zero error angle in pitch, M is zero.

In addition to the orbital characteristics described above, the position of the sun also has to be considered. This position can be expressed in equatorial spherical coordinates by the inclination angle of the ecliptic to the equator and the right ascension of the sun.

Properties of the satellite itself that have an effect on the temperature equilibrium are: (*a*) emissivity of satellite skin with respect to solar light; (*b*) emissivity of satellite skin with respect to infrared radiation; (*c*) heat capacity of skin; (*d*) heat capacity of instrumentation; (*e*) coefficients of heat conduction between skin and instruments; (*f*) coefficients of radiative heat transfer between skin and instruments; (*g*) internal heat release of instruments; and (*h*) shape of satellite.

The nose cone and cylinder of the satellite are separated by an insulating ring, and must be considered individually because of their differing shapes and instrumentation. The layout was made so that only small temperature differences would develop between the two parts of the payload.

The parameters (*c*), (*d*) and (*e*) are determined by design considerations and the functions of the payload, and were varied only until the final payload design was chosen.

The radiation equilibrium can be influenced by the proper selection of the emissivities (*a*) and (*b*). For example, if a polished aluminum surface is selected, the average instrument temperature would be 400

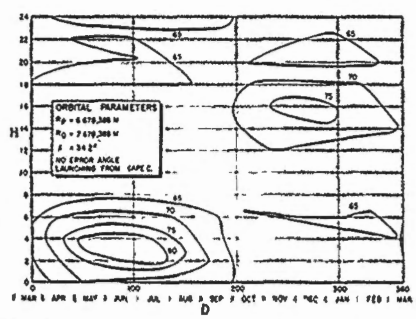

FIG. 16. Time in sunlight for satellites as a function of day of firing and hour of firing. Maximum values at date of firing.

to 450°K., and if a white paint on titanium dioxide or a similar white oxide base is used, the average skin temperature would be near 200°K.

By using sandblasted steel, the temperatures would be between 280 and 380°K., the exact value depending on a combination of the other influencing parameters. Materials not subject to change by environmental conditions for a period of three months were selected for the outside surfaces for Explorer I. Since rough surfaces were expected to give the most constant emissivity value, the selection fell on a combination of sandblasted stainless steel with a special coating of aluminum oxide called Rokide A, applied in lengthwise strips. Due to the spin of the satellite the heat flux to and from the surface is equalized and no lateral temperature gradient (in the shell) can develop. The aluminum oxide strips effect a decrease in the average temperatures.

In order to keep the temperature fluctuations of the instruments small, the instruments are insulated from the skin, but the great variation in temperature between the time the satellite is in the shadow of the Earth and in sunlight will still affect them. For this reason, the relative time in sunlight is an important factor for thermal equilibrium.

The variations of all these parameters and their influence on the temperature of the Explorer was set up as a computation program in an IBM 704 electronic computer. Since each of the parameters was considered as an independent variable, the complete analysis cannot be illustrated in graphical form. However, some of the more important effects can be shown if only two parameters are considered at a time. The parameters that are used in the diagrams, the hours of firing H, the firing day D, and the ratios of emissivities are the ones that could be most freely isolated.

Figure 16 shows the percentage of time in sunlight at the day of firing in an $H-D$ diagram. The percentages of time in sunlight are given as contour lines similar to lines of constant altitude in maps. The two maxima occur for firing at $H=3$ around the summer solstice and at $H=15$ around the winter solstice. This graph is typical for the orbital characteristics listed, but for more eccentric orbits the maxima will increase. The time in sunlight changes due to the regression of the nodes, the progression of the line of apsides, and the mean motion of the sun and may reach 100 per cent even if the initial value is near a minimum. If the computation is extended for an orbiting time of 60 days with the maxima occurring at any time during this period, all values increase, the maxima and minima interchange and the values of Fig. 17 result.

The influence of the eccentricity e is shown in Figs. 18 and 19. In this case all parameters are kept constant except the hour of firing H

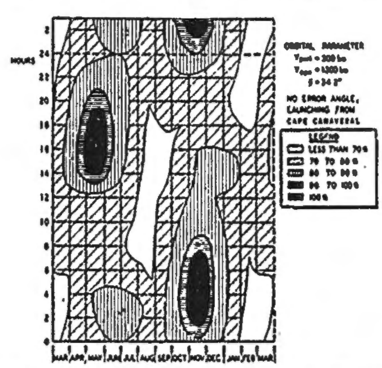

FIG. 17. Time in sunlight for satellite as a function of day and hour of firing. Maximum values during first sixty days of orbiting.

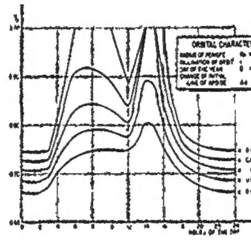

FIG. 18. Time in sunlight (T_s) *versus* hour of the day of launching. Maximum values during first 30 days of orbiting.

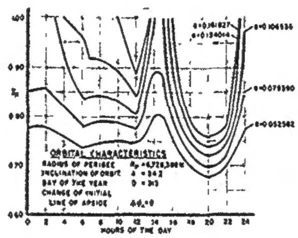

FIG. 19. Time in sunlight (T_s) *versus* hour of the day of launching. Maximum values during first 90 days of orbiting.

and the eccentricity e. Figure 18 gives maxima of time in sunlight occurring during 30 days of orbiting and Fig. 19 shows the same for 90 days. For low eccentricity the time in sunlight barely exceeds 80 per cent at $H=14$ in both graphs. For the high eccentricities, 80 per cent is always exceeded except for the hours between $H=18$ and $H=3.5$ on Fig. 18 and between $H=18$ and $H=22$ on Fig. 19. The eccentricity of Explorer 1 turned out to be 0.1399 which is between the upper two values of Figs. 18 and 19.

In order to study the influence of the emissivities α and ϵ on the temperature, it has been found that it is more convenient to use the ratio of α over ϵ as one parameter and α as the other.

Figures 20 to 22 illustrate the effects of the ratio α/ϵ with the time in sunlight as a parameter. Three typical cases of environmental conditions and of satellite attitude are shown here. Figure 20 shows the

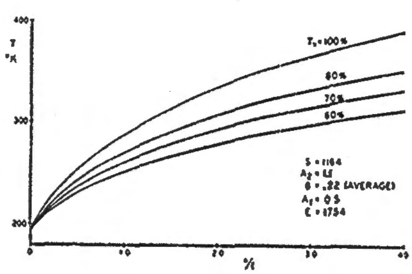

FIG. 20. Temperature of orbiter as a function of ratio of emissivities α/ϵ and time in the sunlight (T_s).

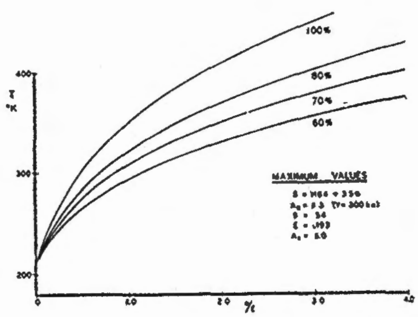

FIG. 21. Temperature of orbiter as function of ratio of emissivities α/ϵ and time in the sunlight (T_s).

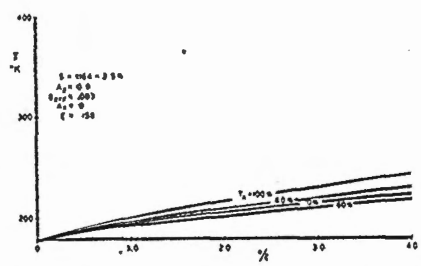

FIG. 22. Temperature of orbiter as function of ratio of emissivities α/ϵ and time in the sunlight (T_s).

relations for a set of parameters that can be considered to be average. Two simultaneous differential equations have to be solved.

$$A_1\alpha SD_1 + A_2\alpha BSD_2 + A_3\epsilon ES - A_4\sigma\epsilon T_s^4 - C_5\dot{T}_i - C_6\dot{T}_s + Q = 0$$
$$C_6\dot{T}_s = C_7(T_s^4 - T_i^4) + C_5\sigma(T_s^4 - T_i^4)$$

wherein

A_1' = projected area with respect to sun direction depending on the attitude angle (it is different for cone and for cylinder due to different geometry)
α = emissivity for sunlight
S = radiative heat flux from sun per unit area and unit time (solar constant)

D_1 = step function ($D_i = 1$ for sunlight, $D_i = 0$ for shadow)
A_2 = effective area of satellite with respect to albedo radiation. It depends on altitude of satellite and the attitude angle with respect to the radius vector and on angle of sun to orbital plane
D_2 = step function ($D_2 = 1$ for orbit in hemisphere $\pm 90°$ from sun direction; $D_2 = 0$ for orbit $\pm 90°$ from opposite of sun direction)
A_3 = effective area of satellite with respect to Earth radiation
ϵ = emissivity with respect to infrared
E = ratio of infrared heat flux radiation from Earth to S
A_4 = total surface area of satellite
σ = Boltzman constant
T_s = skin temperature
C_5 = heat capacity of skin
$\dot{T}_s$ = rate of temperature change of skin
C_6 = heat capacity of instruments
T_i = instrument temperature
$\dot{T}_i$ = rate of temperature change of instruments
Q = heat release of instruments
C_7 = heat transfer coefficient by conduction
C_8 = heat transfer coefficient by radiation.

Figures 20 and 21 are the averages of instrument temperatures T_i obtained by integration of the above equations. The graphs show very

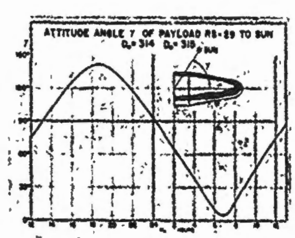

FIG. 23.

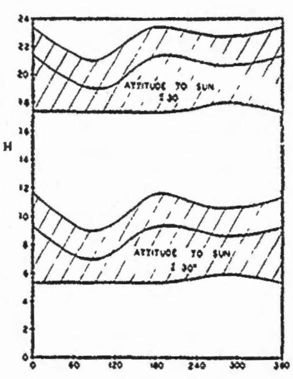

FIG. 24. Attitude angle of the axis of the payload to the sun as a function of day and time of missile firing.

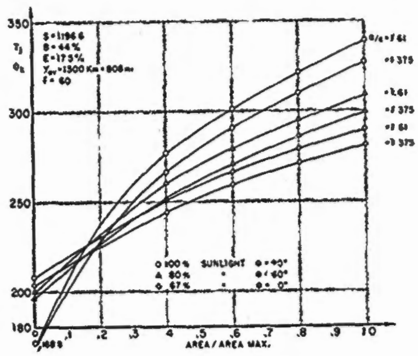

FIG. 25. Instrument temperature (average skin temperature).

clearly the wide range of temperatures that can be expected for a satellite like the Explorer. Figure 22 shows the case in which the projected area to the sun is zero, that is, when the cylinder is pointed directly toward the sun. This is not the case for the cone, however, because of its shape. For firing around $H = 7$ and $H = 19$, the attitude angle will be small but will never be zero, because it is not possible to fire directly into the direction of the sun or away from the direction of the sun with the given conditions of the Explorer I. This is illustrated by Fig. 23, which shows the attitude angle of the satellite axis to the sun as a function of the hour H. The smallest attitude angle, 5°, is reached at $H = 7$, whereas the greatest angle for the rearward sun position is 150° at $H = 19$. These minima or maxima attitude angles change throughout the year as shown on Fig. 24. The shaded areas are those for which the attitude angle is smaller than 30° or higher than 150°, expressed on the graph also as ≤ 30 measured from a line away from the sun. The upper limits are indicated by two lines which indicate respectively the situation at the day of firing and the situation during 60 days of orbiting (wider limits).

In order to illustrate the dependency of the temperature on the area ratio of A_1 to the maximum value of A_1, it is necessary to plot this relation for a number of typical cases where numerical values have been fixed for other parameters. In Fig. 25 the parameters chosen are explained. The parameter f is the ratio of the area A_2 to its maximum value computed at sea level. The curves show three combinations of the parameters:

1. 100 per cent sunlight $\theta = 90°$
2. 80 per cent sunlight $\theta = 60°$
3. 67 per cent sunlight $\theta = 0°$

For each of these combinations, two values of α/ϵ have been used. The angle θ in this graph is the angle between the sun direction and orbital plane. Figures 16 through 25, described on the previous pages, give an illustration of the dependencies for certain sets of parameters. The complete analysis has shown that temperatures of the instrumentation can be kept within the desirable limits by entirely passive control which does not add to the missile complication, and even more important, costs only little weight. The three parameters chosen were:

1. Time of firing $H = 0$ with a tolerance of 2 hours. This takes care of the influence of the time in sunlight and the attitude to the sun.
2. Ratio of Rokide coating to sandblasted stainless steel (25 per cent for cylinder, 30 per cent for cone).
3. Insulation such that the deviations of the instrument temperatures from a mean value during one revolution do not exceed ±5°C.

Figure 26 is a plot of the expected instrument temperature of the

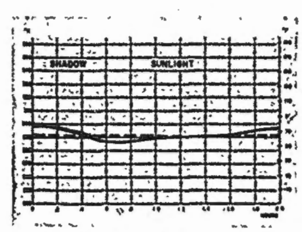

FIG. 26. Explorer I cylinder instrument temperatures. Theoretical curve for second revolution.

high powered transmitter, based on the actual orbital characteristics as evaluated after the launching. The shadow period is indicated by a shaded area. The graph has been computed for the parameters of the day of the firing, but is valid for approximately 10 days of orbiting.

Results of Temperature Measurements from Explorer I

Temperature data have been evaluated from the Microlock transmitter signals received by the following stations: Cape Canaveral (C); Jet Propulsion Lab (J); Earth Quake Valley (E); Temple City (T); Singapore (S); and Nigeria (N).

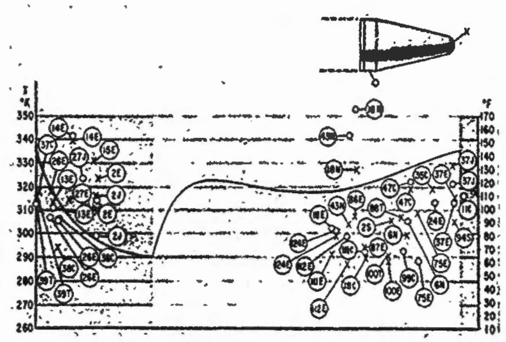

FIG. 27. Cone and skin temperatures of Explorer I. Numbers indicate number of revolutions.
C = Cape Canaveral
J = Jet Propulsion Laboratory
E = Earthquake Valley
T = Temple City
S = Singapore
N = Nigeria

Figure 27 shows a plot of the temperature measurements on the skin of the nose cone of the payload. Circles indicate values measured on the side and crosses indicate data measured on the rounded tip. The points have been labeled with the station that received the measurement and with the number of the revolution at which it was taken. All data fall between 287 and 241°K. and are plotted *versus* the time of one period, measured from the time the satellite enters the shadow. The length of time of the shadow period is 0.515 hour. The light area corresponds to the period in sunlight, which is 1.400 hours. Relative time in sunlight is 73 per cent.

Theoretical skin temperatures are obtained by a computation program in which the equations of motion are solved by step-wise integration and by solving the thermodynamic equations for each step on an IBM 704 computer. Results for the nose cone skin temperatures computed for the conditions at the day of firing are included in Fig. 27 as a solid line. Maximum temperature at the end of the sun period is 335°K.; minimum temperature at the end of the shadow period is 290°K. It can be seen that the variation of the skin temperature during each sun-shadow cycle is larger than the variation of the instrument temperatures as shown in Fig. 26 for the temperature of the high power transmitter.

Temperature measurements of the high power transmitter were obtained from the Microlock Stations at Cape Canaveral, Jet Propulsion Laboratory and Temple City and from the Minitrack Stations at Antofagasta, Chile, and Quito, Ecuador. Figure 28 is a plot of data received during the first 137 revolutions of Explorer I. Temperatures are plotted *versus* the relative time of the shadow-sunlight cycle, as in Fig. 27. The arithmetic mean of the data is marked by the heavy line at 292°K. All points fall between 270 and 310°K. The great scatter of data can be expected if the possible changes of environmental conditions and the dependency on a great number of parameters is considered. Comparison with Fig. 26 shows that the scatter exceeds the theoretical variation of ±5°.

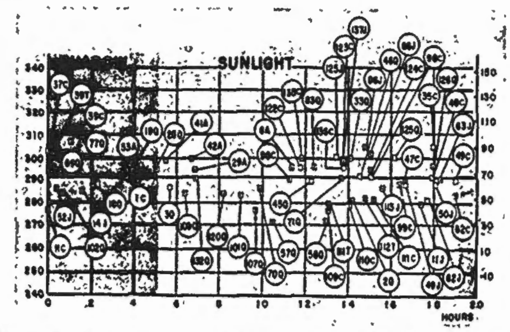

FIG. 28. Explorer I cylinder instrument temperatures (Feb. 1 to Feb. 12, 1 to 135 revolutions).

A = Antofagasta
C = Cape Canaveral
J = Jet Propulsion Lab
T = Temple City
Q = Quito

The following influence factors apply for the variations of some of the parameters:

Parameters	Symbol	Standard Value	Variation	Change of Temperature, deg.
Albedo	B	0.44	±0.2	±5
Earth radiation	E	0.175	±0.1	±8
Solar radiation	S	$1164 \frac{\text{Kcal}}{m^2h}$	±3.5%	±2.5
Time in sunlight	T_s	0.73	±0.05	±5
Projected area to sun	A_1	0.9	±0.1	±8
Altitude of satellite	H	1300 km	±100 km	∓1
Emissivity ratio	α/ϵ	1.4	±0.1	±5

The discussion of the projected area to the sun A_1 and the areas A_2 and A_3 would not be complete without an analysis of the direction of the spin axis. The satellite is spin stabilized and the initial spin axis can be expected to coincide within 1° with the velocity vector at the injection point. The satellite may have a small initial tumbling motion but this would not have an effect on the attitude in space and on the angle with respect to the sunline. Since the rotation of the satellite occurs around the minor axis, it is not stable over a long period of time. Any tumbling motion increases, if energy is dissipated by internal friction. The spin around the longitudinal axis will finally cease and

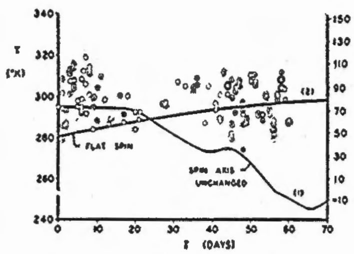

Fig. 29. Temperature of Explorer I; nose cone instrument temperature.

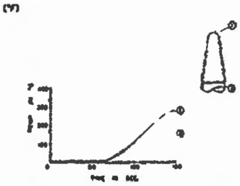

Fig. 30. Payload No. 29, temperature during boost.

the satellite will spin around the lateral axis. The momentum vector is not expected to change, because momentum exchange with the surrounding is very small for a satellite of higher than 200 miles perigee.

This expected increase of the tumbling motion is exactly what happened with Explorer I. However, it occurred considerably faster than anticipated. A plausible explanation is the rapid dissipation of energy by the flexing of the wire antennas of the payload. The tumbling motion has a randomizing effect on the attitude angles to sun and Earth. Explorer I ceased to spin around the longitudinal axis within the first day from launching, and within five days, the tumbling motion converted to a "flat spin," the two-dimensional rotation around the lateral axis with the highest moment of inertia.

The effect of the mode of spin on the satellite temperature is shown in Fig. 29 which shows two temperature curves as a function of the days after launching. The temperature for the case of unchanged spin axis drops off after about 20 days and reaches a minimum after about 65 days. The curve computed for the flat spin case starts lower than the first case. It intersects the first curve after about 23 days and rises steadily to a maximum at 65 days. Measuring points of nose temperature obtained from a great number of different stations have been plotted in the same graph. The scatter during the 60 days of orbiting stays about the same with ±20°C. around a mean value. The trend of the temperature measurements follows first the curve for unchanged spin, but then changes over to the curve for the flat spin and follows it for the remainder of time in which measurements were obtained

Design considerations for the payload include requirements for protection against aerodynamic heating during the boosting phase. The steel nose cone with a rounded tip fulfills this requirement. The

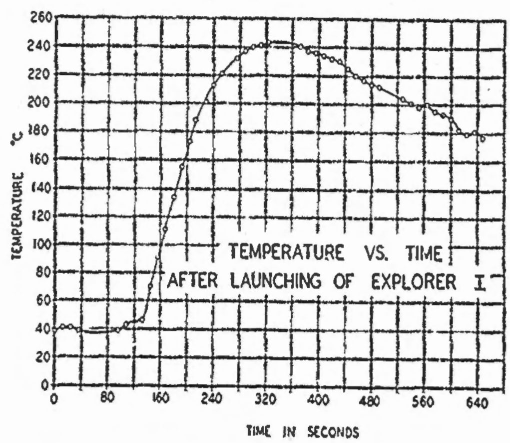

FIG. 31.

highest temperature occurs at the stagnation point. The heat capacity at this point has been increased by increasing the wall thickness to 0.125 inches, which means for the small area of the hemispherical tip only a small weight penalty. The thickness of the steel shell at the rear is 0.015 inches. Theoretical temperatures were obtained for an anticipated trajectory and are shown on Fig. 30. Curve 1 gives the expected temperature at the stagnation point, the maximum value being 290°C. Curve 2 gives values for temperatures at the side wall of the payload shortly downward from the cone frustum. Highest expected temperature at the end of the first stage propulsion is 140°C. The effect of this skin temperature on the internal instrumentation is small because the insulation between skin and instruments has a time constant of about 1 hour. Initial instrument temperatures measured at Cape Canaveral immediately after injection into the orbit are about 10°C. higher than the average of later measurements. This is due mainly to the higher temperature in Florida on the ground that existed prior to the firing. It was 28°C. compared to the average instrument temperature of 20°C.

Temperature measurements obtained during boosting flight from the measuring gauge at the tip of the payload are plotted in Fig. 31. The maximum temperature is somewhat smaller than the theoretical value, but a comparison of the time shows the peak is later. The difference in time is considerable if counted in seconds, but is small if counted in hours, which is the time unit used for satellite orbits. The time constant of the measuring gauge was small compared to temperature gradients during orbiting flight, but not sufficient for the boosting phase. Considering this effect, the agreement of the temperature during boosting with the expected value is sufficient to confirm the fact that the method applied for heat protection has been successful.

ERROR ANALYSIS OF SINGLE AND TWO-FORCE FIELD SPACECRAFT ORBITS

BY

KRAFFT A. EHRICKE[1]

1. INTRODUCTION

A spacecraft is defined here as a controlled celestial body with the purpose of carrying a payload along a predetermined Keplerian orbit. Basically, one can distinguish between closed orbits and open orbits or transfer orbits. A closed orbit is generally a terminal orbit, that is, either a departure orbit, or a target orbit. Closed orbits are possible only in a single force field which ideally is a central force field (center body can be replaced by a mass point); under most actual conditions, however, it is a quasi-central force field, a field which is not spherically-symmetrical, but distorted by perturbations, caused either by the center body proper, due to inhomogeneity of internal mass distribution and non-spherical shape, or by a powerful companion, for example, the Moon. These perturbations are not part of astronautical error analysis.

Transfer orbits connect terminal orbits. They can be flown either within a given single force field or between two force fields. Launching a spacecraft from the Earth into interplanetary space is a typical example. A two-force field transfer orbit always involves two types of orbits, namely a hyperbolic escape orbit from lower-order force field (for example, the geocentric field) and another conic (usually an elliptic transfer orbit) in the higher-order force field (for example, the heliocentric field).

It is the purpose of astronautical error analysis to express mathematically the effect of errors or deliberate changes in velocity on the orbital elements of either transfer orbit or terminal orbit.

2. DISCUSSION OF ERRORS

In general, an error is defined as a deviation in position or in velocity vector from the predetermined conditions at termination of powered flight and beginning of (Keplerian) coasting. The predetermined conditions will also be referred to as reference conditions and reference orbit. In the case of two-force field transfer, as explained in the preceding section, an error measured in the coordinate system whose origin is in the center of the lower-order force field, will be designated generally as *planetocentric* error, the other as *heliocentric error*. Obviously, no such distinction is required in the single-force field error analysis.

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Emphasis on certain errors depends on the nature of the mission under consideration. For example, in establishing a satellite orbit, or generally, a new closed orbit in a single force field, one is less concerned with the transfer orbit than with the target orbit. Emphasis is therefore on the errors existing at the moment of the injection into the predetermined final orbit. On the other hand, if the target orbit already exists in the sense that it is the orbit of a body with which a close encounter is desired, such as in launching a Moon probe or a planet probe, then primary emphasis is on the transfer orbit. In the case of the planet probe, first errors in the geocentric transfer orbit (hyperbola) then errors in the heliocentric transfer ellipse leading to the target planet must be considered.

Errors can be divided into three basic categories, namely velocity errors, position errors and timing errors. Position errors are of importance either if the inclination of the target orbit with respect to a reference plane is significant, or if one wishes to encounter a target body.

A position error can have three causes. Two of these are: errors in the powered flight path or errors in launching[3] time. The first cause will necessarily involve simultaneous velocity errors, compared to which the position error is quite small and may be expressed in terms of an additional velocity error. The second cause is not likely to happen accidentally, but may occur as an emergency measure, involving comparatively large changes in launching position as well as in the position of the target body. Such case requires a re-definition of the reference cut-off conditions and therefore does not really belong into the framework of error analysis. The third cause, most likely to occur in the case of a powered maneuver out in space, is uncertainty of one's position. Velocity errors pertain to the magnitude or direction of the velocity vector in the predetermined orbital plane and to orthogonal velocity components, normal to the predetermined orbital plane. Errors within the orbital plane may also be expressed in terms of errors in the radial and azimuthal velocity components.

In the case of a desired encounter with a target body, a velocity error not only will cause a displacement of the vehicle with respect to the desired intersection point with the target orbit, but due to the change in transfer time it will also cause a displacement of the target body with respect to the intersection point because the transfer was too fast or too slow for the departure constellation of vehicle and target body.

Subsequently, these various errors will be analyzed. Derivations of equations will not be given wherever they have been presented already elsewhere. In this case, the reference will be given for the benefit of the interested reader.

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[3] Launching is the beginning of powered flight, no matter whether on the ground or in a departure orbit.

3 ERROR ANALYSIS OF THE ELLIPTIC ORBIT

The error analysis of the elliptic orbit can be divided into five parts, namely, firstly the effect of arbitrary combinations of radial and azimuthal velocity errors on the position of a body at a different place in the orbit, secondly, the effect of velocity errors in the orbital plane on the orbital elements, thirdly, the effect of orthogonal velocity components on orbital elements, fourthly, the effect of a velocity error on the transfer time, fifthly, the effect of a position error. Knowledge of the orbital elements is presumed. They are explained in every astronomical textbook.

3.1 Effect of Arbitrary Combinations of Radial and Azimuthal Velocity Errors

Figure 1 explains the nomenclature used in conjunction with the elliptic orbit. Shown are the apsidal distances r_A and r_P, the polar

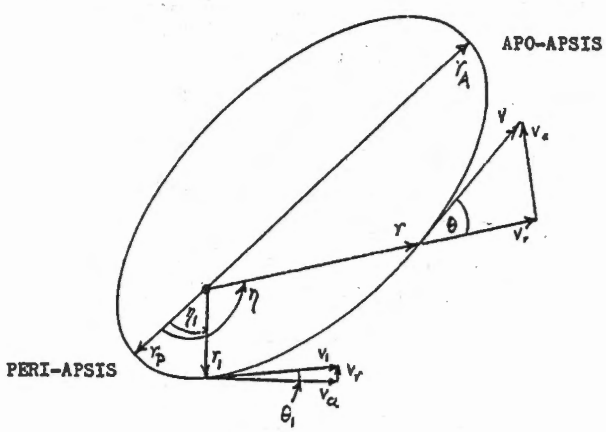


FIG. 1. Elliptic orbit.

coordinates of a point in the orbit, namely radial distance r and true anomaly η. The tangential velocity is v_1, the path angle θ and the radial and azimuthal velocity components are $v_r = \dot{r}$ and $v_a = r\dot{\eta}$. Subscript 1 refers to a particular condition, for example, the cut-off condition and beginning of Keplerian flight.

One can now write the polar equation of the ellipse in the form (1),[*]

[*] The boldface numbers in parentheses refer to the references appended to this paper.

$$r = \frac{r_1 v_1^2 \cos^2 \theta_1}{K(1 + e \cos \eta)}, \tag{1}$$

where e is the eccentricity of the ellipse and $K = gr^2$, the product of gravitational acceleration at distance r and the square of the distance r_1

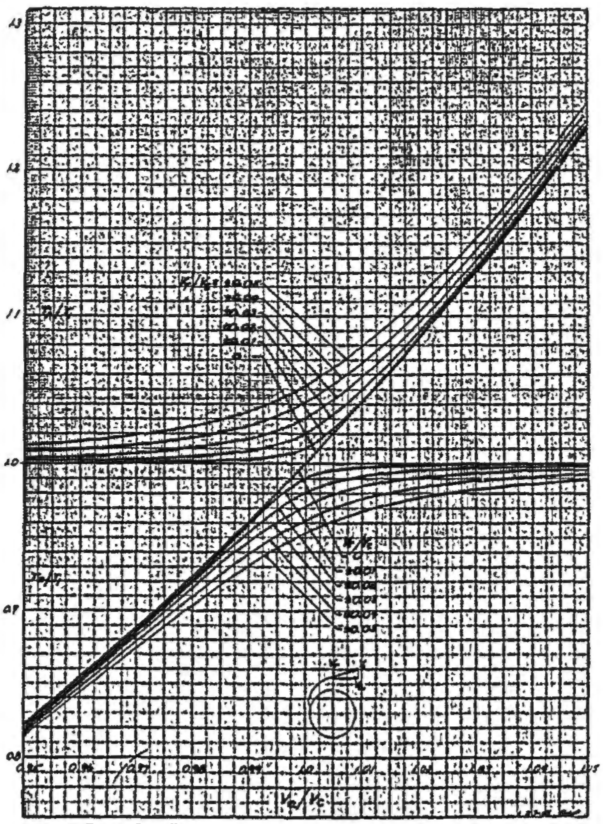


Fig. 2. Cut-off error spectrum with lines of constant radial velocity. (Distance error $\Delta r = 0$)

is the gravitational parameter, a constant characterizing the particular single force field. By proper conversion (1, 2) this equation can be written in the form, expressing r in terms of r_1 to make it dimensionless

$$\frac{r}{r_1} = \frac{(v_{a_1}/v_{c_1})^2}{1 + \chi(v_1)\cos\eta} \tag{2}$$

where $v_{c_1} = \sqrt{K/r_1}$ is the circular velocity at distance r_1 and

$$\chi(v_1) = \sqrt{\left[1 - \left(\frac{v_{a_1}}{v_{c_1}}\right)^2\right]^2 + \left(\frac{v_{a_1}}{v_{c_1}}\right)^2\left(\frac{v_{r_1}}{v_{c_1}}\right)^2}. \tag{3}$$

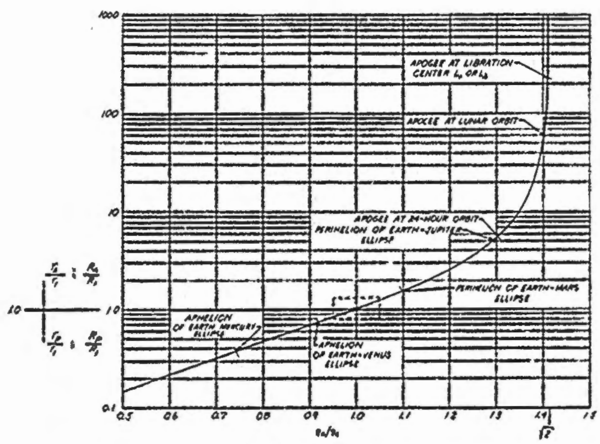


FIG. 3. Distance error as function of the cut-off velocity error for the Earth-Moon system and the solar system.

It should be noted that η in Eqs. 1 and 2 pertains to r, not to r_1, as also indicated in Fig. 1. In other words, these equations define the distance r at a given true anomaly η for given initial conditions v_1, θ_1, or v_{a_1}, v_{r_1}.

Figure 2 shows a general plot of Eq. 2 for the case $r = r_A$ and $r = r_P$, that is, for $\eta = 180°$ ($\cos\eta = -1$) and $\eta = 0°$ ($\cos\eta = 1$), respectively (1, 2). This graph shows, for example, that if $v_r/v_c = 0$ and $v_a/v_c = 1$ then $r_A/r_1 = r_P/r_1 = 1$, that is, the orbit is indeed an exact circle. If $v_r/v_c = 0$, but $v_a/v_c = 1.01$, that is, 1 per cent higher than local circular, then $r_A/r_1 = 1.04$, but $r_P/r_1 = 1$, that is, the orbit is an ellipse and cut-off is given at the peri-apsis. If $v_r/v_c = 0$ but $v_a/v_c = 0.99$, then $r_A/r_1 = 1$ but $r_P/r_1 = 0.96$. The orbit is an ellipse and cut-off was given at the apo-apsis. If $v_r/v_c \neq 0$, the orbit is an ellipse, even if $v_\theta/v_c = 1.0$, but the cut-off point no longer agrees with one of the apsides. For example, let $v_r/v_c = 0.02$ and $v_\theta/v_c = 0.99$. Then $r_A/r_1 = 1.099$ and $r_P/r_1 = 0.954$, that is, the cut-off point in this case was close to the apo-apsis. This graph is generally valid for any central force field. It will be noted that the effect of radial velocity components is greatest when $v_\theta/v_c = 1.0$ and rapidly diminishes as $v_\theta/v_c \gtrless 1.0$. This may be noted in Fig. 3 where the apsidal distances in terms of cut-off distance are shown over a considerably wider range of v_θ/v_c. The dashed square around $v_\theta/v_c = 1$ shows the range of Fig. 2. The lines $v_r/v_c = 0.05$ are barely recognizable. For the range of Fig. 3 we have shown therefore only the line $v_r/v_c = 0$. It is seen that this line becomes vertical, indicating infinite error sensitivity as $v_\theta/v_c = \sqrt{2}$. At that point the ellipse has become a parabolic orbit. On the other end, the curve becomes horizontal for $v_\theta/v_c = 0$ (vertical fall toward the center), indicating zero error sensitivity.

The gradient of this curve (as well as of any curve $v_r/v_c \neq 0$) is of great interest in astronautical error analysis, since it defines the error sensitivity of the apsides (or any other distance $r(\eta)$) for the given initial condition v_{θ_1}/v_c and v_{r_1}/v_c. The general equation for this gradient has been derived in (1). For a given value v_{r_1}/v_{c_1}, the change in distance r at a given true anomaly η, for a given change in v_{θ_1} is

$$\frac{\partial\left(\frac{r}{r_1}\right)}{\partial\left(\frac{v_{\theta_1}}{v_{c_1}}\right)} = 2\frac{v_{\theta_1}}{v_{c_1}} \frac{1 + \chi(r_1)\cos\eta - [\chi(r_1)]^{-1}\left(\frac{v_{\theta_1}}{v_{c_1}}\right)^2 \cos\eta\left[\left(\frac{v_{\theta_1}}{v_{c_1}}\right)^2 + \frac{1}{2}\left(\frac{v_{r_1}}{v_{c_1}}\right)^2 - 1\right]}{[1 + \chi(r_1)\cos\eta]^2}. \quad (4)$$

This equation is perfectly general. For cut-off at one of the apsides ($v_{r_1}/v_{c_1} = 0$) the equation for the variation of the apsides with variation in v_{θ_1} simplifies to (1),

$$\frac{d\left(\frac{r_{\text{apsis}}}{r_1}\right)}{d\left(\frac{v_{\theta_1}}{v_{c_1}}\right)} = \frac{4\left(\frac{v_{\theta_1}}{v_{c_1}}\right)}{2 - \left(\frac{v_\theta}{v_c}\right)^2}. \quad (5)$$

This relation is shown in Fig. 4. If the error or change is very small, compared to the flight velocity, which is usually assumed in error analysis, one may replace dr by Δr etc. Thus, if $v_r/v_c \neq 0$, the value of $(\partial r/\partial r_1)/(\partial v_{\theta_1}/\partial v_{c_1})$ must be computed from Eq. 4 and at the particular value of η, the distance error is given as function of $\Delta v_{\theta_1}/v_{c_1}$ by

$$(\Delta r)_{\eta, v_r/v_c} = \frac{\partial\left(\frac{r}{r_1}\right)}{\partial\left(\frac{v_{\theta_1}}{v_{c_1}}\right)} \cdot r_1 \cdot \frac{\Delta v_{\theta_1}}{v_{c_1}}. \quad (6)$$

If Eq. 5 can be used, the displacement of the apsis opposite to the cut-off apsis becomes

$$\Delta r_{\text{apsis}} = \frac{d\left(\frac{r_{\text{apsis}}}{r_1}\right)}{d\left(\frac{v_{a_1}}{v_{c_1}}\right)} \cdot r_1 \cdot \frac{\Delta v_{a_1}}{v_{c_1}}. \tag{7}$$

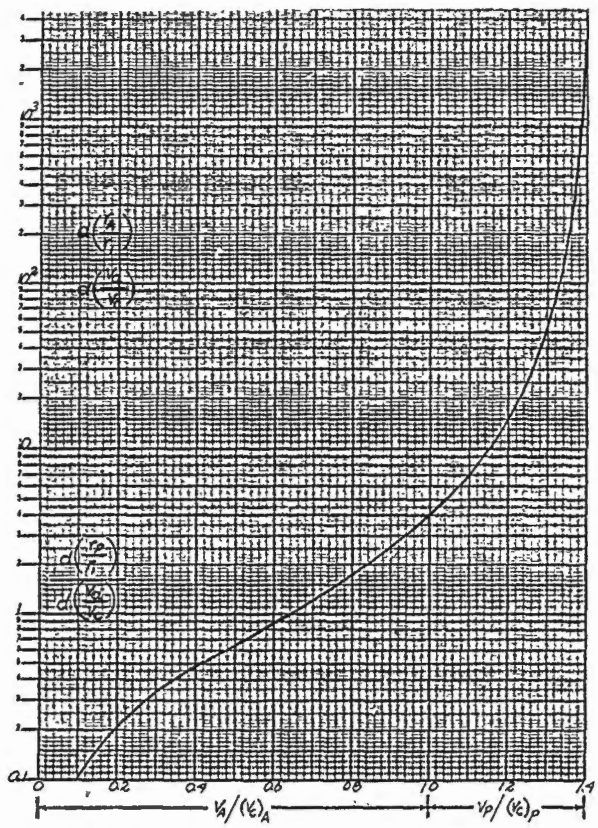


FIG. 4. Rate of change of apsidal distance with magnitude of apsidal velocity

Example: At the Earth's surface ($r_1 = r_P = 1$) a vehicle is launched into an ellipse whose apogee lies at a distance of 60 Earth radii (about Moon distance). The departure velocity is $v_{e_1}/v_{c_1} \approx 1.39$, corresponding to about 35,800 ft./sec. For this value of v_{e_1}/v_{c_1} the gradient is shown in Fig. 4 to be 1500. Application of Eq. 7 gives for the apogee displacement in the case of $\Delta v_{e_1} = 1$ ft./sec. error,

$$\Delta r_A = 1500 \cdot 1 \cdot \frac{1}{35{,}800} = 0.0418 \text{ Earth radii}$$

or about 440 n.mi./(ft./sec.). This is a considerable error sensitivity, but nevertheless one order of magnitude smaller than for the nearest

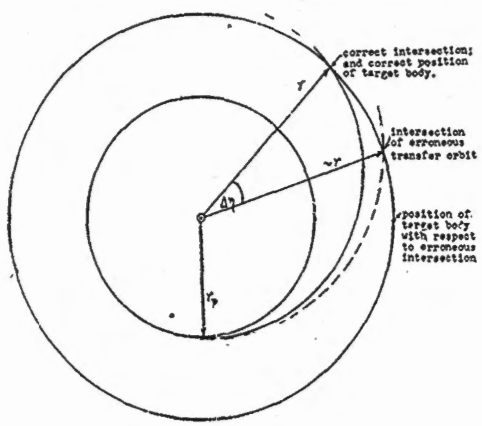


FIG. 5. Error sensitivity of intersecting orbit.

planets. One finds an analogous displacement of aphelion or perihelion of a transfer orbit departing from the Earth's orbit (but not the Earth) and touching the mean circular orbits of Mars and Venus, respectively, about 6000 n.mi./(ft./sec.) for Mars and about 2000 n.mi./(ft./sec.) for Venus.

Often the apsidal displacement is of lesser interest than the displacement of the intersection of the transfer orbit with the target orbit. In other words, instead of defining the change r for a given η, one wishes to determine the change in η for a given r of the target orbit (Fig. 5). Unless the change in true anomaly, $\Delta\eta$, is very large, one can write for the displacement

$$\Delta s = r\Delta\eta^{\text{rad}} \tag{8}$$

where $\Delta\eta$ is found from the relation (3),

$$\frac{\Delta\eta}{\Delta v_1} \approx \frac{d\eta}{dv_1} = \frac{\frac{2v_1}{K/a}\left(\frac{ea}{r}-\frac{a}{er}\right) - \frac{2}{v_1}\left(\frac{v_1}{v_e}\right)^2\left(\frac{a}{e^2r}+\frac{a}{r}-\frac{1}{e^2}\right)}{\sin\eta} \tag{9}$$

where a designates the semi-major axis of the transfer ellipse. It will be observed that $\Delta\eta$ becomes infinite for a cotangential transfer ellipse between circular orbits, since in this case $\eta = 180°$ and $\sin\eta = 0$. The above equation is therefore not applicable for this case. Actually the transfer ellipse merely contacts the target orbit, but does not intersect. In order to obtain the effect of a small velocity error in this case, it is preferable to use Eq. 1 solved for η,

$$\cos\eta = \frac{1}{e'}\left(\frac{r_1}{r}\frac{(v_1+\Delta v_1)^2\cos^2\theta_1}{K} - 1\right). \tag{10}$$

Assume v_1 for the correct 180° transfer to distance r is known. Then, for an error $v_1 + \Delta v_1$, the new value of η is found from Eq. 10 if the new value e' is used, where e' is given by

$$e' = \frac{(v_1+\Delta v_1)^2}{K/r_1} - 1 \tag{11}$$

if $\theta_1 = 0$. If $\theta_1 \neq 0$, the departure point is not an apsis. In this case, e' follows from

$$e' = \sqrt{(1-q)^2 + q^2\tan^2\theta_1} \tag{12}$$

where

$$q = \left(\frac{v_1+\Delta v}{v_c}\right)^2\cos^2\theta_1. \tag{13}$$

3.2 Effect of Planar Errors on the Elliptic Orbital Elements

Turning now to errors within the orbital plane, the effect of a radial velocity on the following orbital elements is given by the subsequent equations (4, 3)

semi-major axis,

$$\frac{\partial a}{\partial v_{r_1}} = -\frac{2a^2v_{t_1}}{K}\sin\eta_1 \tag{14}$$

eccentricity,

$$\frac{\partial e}{\partial v_{r_1}} = -\sqrt{\frac{1+e\cos\eta_1}{K/r_1}}\sin\eta_1$$

mean angular motion,

$$\frac{\partial \mu}{\partial v_{r_1}} = \frac{3}{a\sqrt{1-e^2}} \sin \eta_1 \tag{16}$$

true anomaly of station 1,

$$\frac{\partial \eta_1}{\partial v_{r_1}} = \frac{\cos \eta_1}{e\sqrt{h(1-e^2)}} \tag{17}$$

where

$$h = v_1^2 - \frac{2K}{r_1} \tag{18}$$

where v_1 refers to the correct cut-off velocity.

A radial differential impulse if given at one apsis ($\sin \eta_1 = 0$) is seen to have no effect on the orbital elements, except on the position of the apsis itself, since according to Eq. 17, η is changed. In fact, this change is a maximum if the impulse occurs at the original peri- or apo-apsis. The reason why the other elements are not changed if the impulse is given at one apsis is that in this case a radial component coincides with a normal velocity component (normal to the tangential velocity). A normal component never changes the elements a and μ. It also leaves e unchanged if given at one apsis (4, 3).

The effect of an error in azimuthal velocity on the orbital elements is summarized by the following equations (4, 3),

$$\frac{\partial a}{\partial v_{a_1}} = \frac{2a^2}{K} v_{a_1} \tag{19}$$

$$\frac{\partial e}{\partial v_{a_1}} = \frac{1-e^2}{er_1\sqrt{Ka}} (r_A r_P - r_1)^2 \tag{20}$$

where r_A and r_P are the correct apsidal distances,

$$\frac{\partial \mu}{\partial v_{a_1}} = -\frac{3}{r_1}\sqrt{1-e^2} \tag{21}$$

$$\frac{\partial \eta_1}{\partial v_{a_1}} = \sqrt{\frac{2 + e\cos \eta_1}{eK/r_1}} \sin \eta_1. \tag{22}$$

In the above equations, either v_a or v_r was considered in error. Unless $v_r = 0$, an error in v_a or v_r represents an unspecified combination of scalar and directional error. Therefore, it is sometimes useful to have the error coefficient expressed in terms of v and θ. If the flight path angle θ is correct, then only the scalar value of the velocity can be in error. In this case the following relations apply (3),

$$\frac{\partial a}{\partial v_1} = \frac{2v_1 a^2}{K} \tag{23}$$

$$\frac{\partial e}{\partial v_1} = \pm \frac{2}{v_1}\left(\frac{v_1}{v_{c_1}}\right)^2 = \pm \frac{2}{v_1}\nu_1^2 \quad (\nu \gtrless 1) \tag{24}$$

$$\frac{\partial \mu}{\partial v_1} = -\frac{3v_1}{\sqrt{Ka}} \tag{25}$$

$$\frac{\partial \eta_1}{\partial v_1} = \frac{2}{ev_1}\sin \eta_1. \tag{26}$$

Furthermore,

$$\frac{\partial r_A}{\partial v_1} = \frac{2v_1 a r_A}{K} \pm \frac{2a}{v_1}\nu_1^2 \quad (\nu \gtrless 1) \tag{27}$$

$$\frac{\partial r_P}{\partial v_1} = \frac{2v_1 a r_P}{K} \mp \frac{2a}{v_1}\nu_1^2 \quad (\nu \gtrless 1). \tag{28}$$

In the case of an error in angle θ one computes the resulting normal velocity component (normal to v)

$$\Delta v_{n_1} = v_1 \tan(\Delta\theta_1); \quad \frac{\partial v_{n_1}}{\partial \theta_1} = v_1 \tag{29}$$

and the resulting change in orbital elements (4, 3),

$$\frac{\partial a}{\partial v_{n_1}} = 0 \tag{30}$$

$$\frac{\partial e}{\partial v_{n_1}} = -\frac{r_1}{av_1}\sin \eta_1 \tag{31}$$

$$\frac{\partial \mu}{\partial v_{n_1}} = 0 \tag{32}$$

$$\frac{\partial \eta_1}{\partial v_{n_1}} = \frac{2ae + r_1 \cos \eta_1}{aev_1}. \tag{33}$$

It will be noted that in this section only four orbital elements, a, e, μ, η_1 were considered. The fifth and sixth elements are the nodal line and the orbital inclination. They are the non-planar elements, determining the orientation of the orbital plane with respect to a given reference plane. Neither the direction of the nodal line nor the orbital inclination is affected by errors or changes in the planar velocity components.

3.3 Effect of an Orthogonal Velocity Component on the Orbital Elements

The three axes which define the orientation of the elliptic orbit in space are shown in Fig. 6. In analogy with aeronautics, rotations about

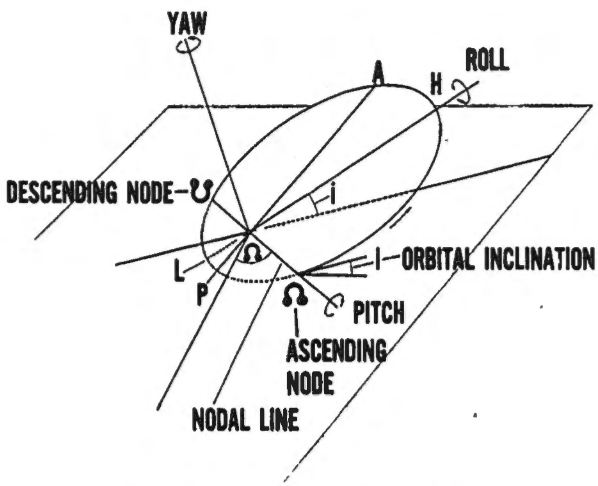


FIG. 6. The three axes of rotation of the elliptic orbit.

one of these axes may be designated as pitch, yaw and roll motion. Of these, yaw motion does not change the orbital plane, but merely rotates the apside line *P-A*, while the nodal line and the *L-H* line remain fixed. Yaw, therefore, amounts to a change in true anomaly of a given point in the orbit (that is, the longitude of the perigee), and has been discussed in the preceding section. The nodal line ☊ ☋ is the line of intersection of the orbit plane with the reference plane. Rotation about this line (pitch) changes the orbital inclination. The line normal to the nodal line intersects the ellipse at its lowest and highest point, respectively, below and above the reference plane, wherefore we call it for the sake of brevity the *L-H* line. Rotation about the *L-H*·line causes a change in orientation of the nodal line, measured by the angle ☊ with respect to a fixed direction in space, usually the vernal equinox ♈.

This rotation will be designated as change in nodal angle, or canting or rolling the orbit.

In general, an orthogonal impulse Δv_w does not change the elements a, e, μ, η, but only orbital inclination and the nodal angle. The tangential velocity vector is not changed in scalar magnitude, but only tilted out of the original plane by the angle

$$\left.\begin{aligned} \alpha &= 2\sin^{-1}\left(\frac{\Delta v_w}{2v}\right) \\ \cos\alpha &= 1 - \left(\frac{\Delta v_w}{v}\right)^2 \\ \sin\alpha &= \sqrt{\left(\frac{\Delta v_w}{v}\right)^2 - \frac{1}{4}\left(\frac{\Delta v_w}{v}\right)^4} \end{aligned}\right\}. \tag{34}$$

This angle will have a varying effect on the amount of tilting, Δi, and canting, $\Delta\Omega$, respectively, depending on where in the orbit the impulse is given, because the ratio of azimuthal to radial velocity component, v_a/v_r, varies. It can be shown that merely v_a affects the changes Δi and $\Delta\Omega$. Suppose Δv_w is produced at the ascending node Ω. The local orbital velocity is v_Ω, the local azimuthal velocity is $v_{a\Omega}$. Then it is

$$\sin \tfrac{1}{2}\alpha_\Omega = \tfrac{1}{2}\frac{\Delta v_w}{v_\Omega}$$

but the change in orbital inclination is

$$\sin \tfrac{1}{2}\Delta i_\Omega = \tfrac{1}{2}\frac{\Delta v_w}{v_{a\Omega}}. \tag{35}$$

A change in orbital inclination can be brought about only where an azimuthal component normal to the nodal line exists. This component normal to the nodal line shall be called $(v_a)_n$ and it is given by

$$(v_a)_n = v_a \cos(\eta - \eta_\Omega) \tag{36}$$

where v_a is the local azimuthal component at a point of true anomaly η and η_Ω is the true anomaly of the nodal point. The change in orbital inclination caused by an orthogonal impulse at any arbitrary point in the orbit, Δi, is therefore generally

$$\sin \tfrac{1}{2}\Delta i = (\sin \tfrac{1}{2}\Delta i_\Omega)\frac{v_{a\Omega}}{v_a}\frac{(v_a)_n}{v_a} = \frac{1}{2}\frac{\Delta v_w}{v_a}\cos(\eta - \eta_\Omega); \quad di = \frac{dv_w}{v_a}\cos(\eta - \eta_\Omega). \tag{37}$$

It is seen that $\Delta i = 0$ at $\eta = \eta_\Omega \pm 90°$ where $(v_a)_n = 0$, that is at the points L or H. At these points the orbital velocity is by definition parallel to the nodal line. However, Δi is a maximum not only if

$\eta = \eta_{\Omega}$ or $\eta = \eta_{\mho}$ (that is, if Δv_w is given at one of the nodes), but also if v_s is a minimum, that is, if $r = r_A$ (line of apsides coinciding with the nodal line).

If Δv_w is given at points L, H, no rotation takes place about the nodal line. This, however, does not necessarily mean that no change in orbital inclination takes place, as will be seen below. However, if given at L, H, the primary effect is a change in nodal angle $\Delta\Omega$. Consider Fig. 7 which shows the projection of two orbits, reference orbit

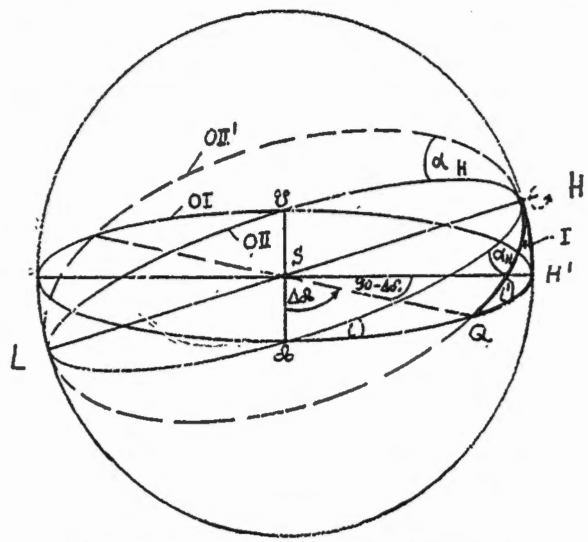


Fig. 7. Determination of orbit canting and associated change in inclination.

OI and inclined orbit OII on the celestial sphere. Suppose an orthogonal impulse is given at point H, rolling the orbit about the angle α_H. The resulting change in nodal angle $\Delta\Omega_H$ is then given by consideration of the spherical triangle $\Omega Q H \Omega$,

$$\sin \Delta\Omega_H = \frac{\sin I}{\sin i} \sin \alpha_H$$

$$= \frac{\sin I}{\sin i} \sqrt{\left(\frac{\Delta v_w}{v_{sH}}\right)^2 - \frac{1}{4}\left(\frac{\Delta v_w}{v_{sH}}\right)^4}, \qquad (38)$$

whence

$$\sin I = \sin i \frac{\sin \Delta\Omega_H}{\sin \alpha_H}. \tag{39}$$

These two equations must be solved simultaneously. The term $\sin I$ is closely connected with the before-mentioned change in i upon an orthogonal impulse in L or H, even if no rotation about the nodal line takes place. The new value of i is

$$\sin i' = \frac{\sin \alpha_H}{\sin \Delta\Omega_H} = \frac{\sin i}{\sin I}. \tag{40}$$

The side I (Fig. 7) is given by

$$\cos I = \sin \Delta\Omega \cos i \tag{41}$$

whence

$$\sin i' = \frac{\sin i}{\sqrt{1 - \sin^2 \Delta\Omega \cos^2 i}}. \tag{42}$$

This relation shows that $i' \to i$ if either $\Delta\Omega$ is small or if $\cos i \to 0$ ($i \to 90°$). In either case, $\cos I \to 0$ and if at the same time $\Delta v_w/v_e \ll 1$), Eq. 38, defining the change in nodal angle $\Delta\Omega_{L,H}$ due to an orthogonal impulse given at L or H, simplifies to

$$\sin \Delta\Omega_{L,H} \approx \Delta\Omega_{L,H} = \frac{\Delta v_w}{v_{e,L,H} \sin i}. \tag{43}$$

If the orthogonal impulse is given at any other point in the orbit one has in analogy to the previous consideration for Δi,

$$\sin \Delta\Omega = \frac{\sin I}{\sin i}\sqrt{\left(\frac{\Delta v_w}{v_e}\right)^2 \sin^2 (\eta - \eta_\Omega) - \frac{1}{4}\left(\frac{\Delta v_w}{v_e}\right)^4 \sin^4 (\eta - \eta_\Omega)} \tag{44}$$

and the simultaneous equation

$$\sin I = \sin i \frac{\sin \Delta\Omega}{\sin \alpha_H} \tag{45}$$

where α_H represents the angle by which the axis LH is turned, where $\sin \alpha_H$ is equal to the square root term in Eq. 44. Again, if $\cos I = \sin \Delta\Omega \cos i \to 0$, hence $\sin I \to 1$ and $\Delta v_w/v_e \ll 1$ (which is the case in most astronautic conditions), then the simplified relation for $\Delta\Omega$ due to an orthogonal impulse at any point in the orbit is

$$\sin \Delta\Omega \approx \Delta\Omega = \frac{\Delta v_w}{v_e}\frac{\sin (\eta - \eta_\Omega)}{\sin i}. \tag{46}$$

The angle $\Delta\Omega$ is a maximum if Δv_w occurs at L or H and zero if it occurs at ☊ or ☋.

The above analysis is more than mere mathematics The position of the orbital plane is of great importance in many astronautical missions. For example if a spacecraft enters a satellite orbit about the Moon or another planet and wishes to return to Earth, the orbital orientation with respect to the departure direction may, due to precession or erroneous maneuver, be so unfavorable that only a small component of the vehicle's velocity vector can be used for the departure. Therefore, corrections may be necessary which either vary the orbital inclination or the nodal angle.

3.4 Effect of a Velocity Error on the Transfer Time

The orbital period is given by

$$T = \frac{2\pi}{\sqrt{K}} a^{3/2}. \tag{47}$$

The variation in orbital period as function of a scalar error in departure velocity is therefore

$$\frac{dT}{dv_1} = \frac{dT}{da}\frac{da}{dv_1} = 6\pi \frac{v_1 a}{\sqrt{K^3/a^3}} \tag{48}$$

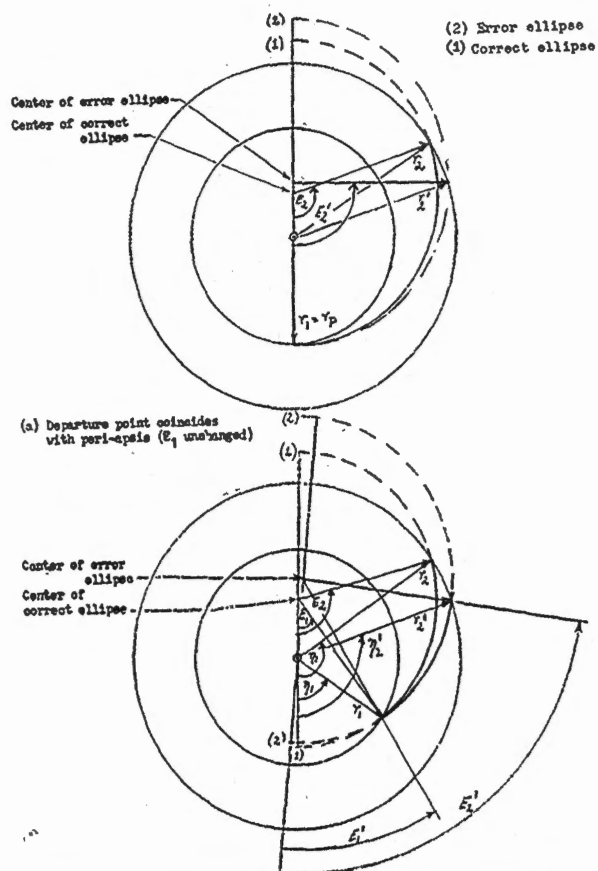
where a represents the original semi-major axis. Equation 48 is equally valid for the half-period.

In the case of an intersecting orbit, let the correct transfer time be Δt. Then, an error in departure velocity, Δv_1, can be shown (3) to result in the change $\Delta(\Delta t)$ in transfer time,

$$\Delta(\Delta t) = \frac{\Delta E_2(1 - e\cos E_2) - \Delta E_1(1 - e\cos E_1) + \Delta e(\sin E_1 - \sin E_2) + \frac{3}{2}\Delta t\,\Delta a\sqrt{\frac{K}{a^5}}}{\sqrt{K/a^3}} \tag{49}$$

with the auxiliary relations

$$\left.\begin{aligned}
\Delta a &= \frac{2v_1 a}{K}\Delta v_1 \\
\Delta e &= \frac{2}{v_1}\left(\frac{v_1}{v_c}\right)^2 \Delta v_1 \\
\Delta E_1 &= \frac{\frac{\Delta e}{e^2}\left(\frac{r_1}{a} - \frac{1}{e^2}\right) - \frac{\Delta a}{a^2}\frac{r_1}{e}}{\sin E_1} \\
\Delta E_2 &= \frac{\frac{\Delta e}{e^2}\left(\frac{r_2}{a} - \frac{1}{e^2}\right) - \frac{\Delta a}{a^2}\frac{r_2}{e}}{\sin E_2}
\end{aligned}\right\} \tag{50}$$



(b) Departure point does not coincide with an apsis (E_1 and E_2 changed by error)

FIG. 8. Effect of transfer time error.

where a and e are original elements, r_1, r_2 two radial distances between which the change in transfer time $\Delta(\Delta t)$ is to be determined and E_1, E_2 the values of the eccentric anomaly belonging to r_1 and r_2 in the original orbit (Fig. 8). If the departure point coincides with one of the apsides of the transfer ellipse, then $E_1 = 0°$ or $180°$. A tangential error would not change the value of E_1. Hence, $\Delta E_1 = 0$, $\sin E_1 = 0$ and Eq. 49 takes the simplified form

$$\Delta(\Delta t) = \frac{\Delta E_2(1 - e\cos E_2) + \frac{3}{2}\Delta t\Delta a\sqrt{\frac{K}{a^3}} - \Delta e\sin E_2}{\sqrt{K/a^3}}. \tag{51}$$

The effect of $\Delta(\Delta t)$ can, especially in interplanetary transfer, be very significant. For example, in a 156-day transfer orbit to Mars with cotangential departure from the Earth's orbit, a heliocentric velocity excess of $\Delta v_1 = 1$ ft./sec. resulted in a displacement of the intersection point (Fig. 5) by $\Delta S \approx 18{,}000$ n.mi. and a reduction in transfer time by $\Delta(\Delta t) \approx 3.6$ hr. Mars travels at about 48,000 n.mi./hr., so that the displacement of the planet from the intersection point is about 170,000 n.mi. The time effect therefore completely overshadows in this case the effect of the vehicle's displacement error. This is especially characteristic of fast transfer orbits to the Moon or the planets, while near-cotangential transfer orbits show a lesser sensitivity with respect to errors in transfer time.

3.5 Effect of a Position Error

Errors in conditions on a certain point in the orbit affect the entire orbit because they change the orbital energy at the given point from its pre-determined value. Errors in velocity change the local kinetic energy of the space vehicle. Errors in position alter the local potential energy of the vehicle. Since potential energy at a given flight condition plays usually a lesser role than kinetic energy, in terms of effects on the entire orbit, it follows that position errors (that is, comparatively *small* differences in desired and actual position) will be less consequential than even a small velocity error.

We ask first the question: Suppose a vehicle enters a transfer ellipse at its peri-apsis r_P. However, the distance determination is in error and the actual value deviates by the error dr_P from the assumed value. Then, what is the position error dr at a true anomaly η in the transfer orbit which the vehicle has entered at a given peri-velocity v_P? From the polar equation in the form

$$r = \frac{r_P^2 v_P^2}{K\left[1 + \left(\frac{v_P^2 r_P}{K} - 1\right)\cos\eta\right]} \tag{52}$$

one finds

$$\frac{dr}{dr_P} = v_P^2 \frac{r_P^2 v_P^2 \cos \eta - 2r_P K\left[1 + \left(\frac{r_P v_P^2}{K} - 1\right)\cos \eta\right]}{K^2\left[1 + \left(\frac{r_P v_P^2}{K} - 1\right)\cos \eta\right]^2} \qquad (53)$$

where r_P is the assumed peri-distance and v_P the actual peri-velocity at cut-off.

As a second case, suppose the peri-apsis cut-off velocity is in error as well as the peri-distance. In this case one obtains from Eq. 52 for the error in distance, dr, at true anomaly η,

$$\frac{dr}{dr_P} = r_P v_P\left[v_P + 2r_P \frac{dv_P}{dr_P}\right] \frac{r_P v_P^2 \cos \eta - 2K\left[1 + \left(\frac{r_P v_P^2}{K} - 1\right)\cos \eta\right]}{K^2\left[1 + \left(\frac{r_P v_P^2}{K} - 1\right)\cos \eta\right]^2}. \qquad (54)$$

The values dv_P and dr_P are not functionally correlated, but are individual errors. No equation of definition therefore exists.

Of further interest is the effect of a position error on the orbit's semi-major axis and eccentricity. These are obtained by using proper relations for the orbital energy

$$h = v^2 - \frac{2K}{r}; \quad \frac{dh}{dr} = \frac{2K}{r^2} \qquad (55)$$

$$h = -\frac{K}{a}; \quad \frac{dh}{da} = \frac{K}{a^2} \qquad (56)$$

whence

$$\frac{da}{dr} = \frac{da}{dh}\frac{dh}{dr} = 2\frac{a^2}{r^2}. \qquad (57)$$

Furthermore, from the relation

$$h = \frac{K}{p}(e^2 - 1) \qquad (58)$$

where p is the semi-latus rectum one obtains

$$dh = \frac{K}{p^2}(1 - e^2)dp + 2e\frac{K}{p}de$$

or, since

$$p = a(1 - e^2); \quad dp = da(1 - e^2) - 2aede \qquad (59)$$

one has

$$\frac{dh}{de} = \frac{K}{p^2}(1 - e^2)\left[(1 - e^2)\frac{da}{de} - 2ae\right] + 2e\frac{K}{p}.$$

With the use of the relation

$$e = 1 - \frac{r_P}{a}; \quad de = -\frac{dr_P}{a} + \frac{r_P}{a^2}da. \tag{60}$$

The following analysis is greatly simplified by neglecting the term dr_P/a, since, for a given error in r at a point different from the peri-apsis, the associated error in peri-apsis, dr_P, is not immediately known. However, since only small deviations are involved in error analysis (for example, $dr/r \ll 1$) it follows that generally the term dr_P/a will be very much smaller than unity so that it may safely be neglected as an additive term. Therewith, one obtains

$$\frac{da}{de} = \frac{a^2}{r_P} \tag{61}$$

where r_P is the original value. Replacing furthermore p by $a(1 - e^2)$ one obtains

$$\frac{dh}{de} = K\left[\frac{1}{a^2(1-e^2)}\left\{(1-e^2)\frac{a^2}{r_P} - 2ae\right\} + \frac{2e}{a(1-e^2)}\right]. \tag{62}$$

Therewith the effect of a position error dr at an arbitrary point in the orbit on the orbit's eccentricity is

$$\frac{de}{dr} = \frac{de}{dh}\frac{dh}{dr} = \frac{2a(1-e^2)}{r^2\left[\frac{1}{a}\left\{(1-e^2)\frac{a^2}{r_P} - 2ae\right\} + 2e\right]}. \tag{63}$$

It is now possible to express the error in r_P in terms of the error at any arbitrary place r in the orbit. Assume, for instance, cut-off is given at a distance r_1 and an error dr_1 has been made in determining r_1. The error in peri-apsis distance, dr_P, is then

$$\frac{dr_P}{dr_1} = \frac{dr_P}{da}\frac{da}{dr_1}. \tag{64}$$

Since $r_P = a(1 - e)$ it follows that

$$\frac{dr_P}{da} = 1 - e - a\frac{de}{da}$$

or, with the aid of Eq. 61

$$\frac{dr_P}{da} = 1 - e - \frac{r_P}{a}. \tag{65}$$

Substituting this term and Eq. 57 in Eq. 64 yields

$$\frac{dr_P}{dr_1} = \left(1 - e - \frac{r_P}{a}\right)\frac{2a^2}{r_1^2} \tag{66}$$

Here again, e, a, r_P and r_1 represent the original values (not considering the error dr_1).

We are now ready to answer the question asked at the beginning of this section in a more general form by eliminating reference to the periapsis. If cut-off is given at a distance $r_1 \neq r_P$ and r_1 is in error by dr_1, what is the error dr_2 at a different true anomaly η_2? Suppose the space vehicle discontinues propulsion at a certain velocity v_1 and distance r_1. The resulting (original) ellipse has the eccentricity e and the peri-distance r_P. Then, at true anomaly η_2 the distance would be (polar equation),

$$r_2 = r_P\frac{1 + e}{1 + e\cos\eta_2}.$$

By differentiating r_2 with respect to r_P and e, and by dividing both sides by dr_1, one obtains

$$\frac{dr_2}{dr_1} = \frac{dr_P}{dr_1}\frac{1 + e}{1 + e\cos\eta_2} + r_P\frac{\cos\eta_2 - 1}{(1 + e\cos\eta_2)^2}\frac{de}{dr_1} \tag{67}$$

where dr_P/dr_1 is given by Eq. 66 and de/dr_1 by Eq. 63, replacing r by r_1. The above equation defines the radial distance error dr_2 at η_2 as result of an error dr_1 (not necessarily at one of the apsides) at the beginning of Keplerian flight.

The change in mean angular motion, μ, as function of a distance error follows from

$$\mu = \sqrt{\frac{K}{a^3}};\quad d\mu = \frac{3}{2}\sqrt{\frac{K}{a^5}}\,da \tag{68}$$

whence

$$\frac{d\mu}{dr_1} = \frac{3}{2}\sqrt{\frac{K}{a^5}}\frac{da}{dr_1} \tag{69}$$

where da/dr_1 is given by Eq. 57. Likewise, one obtains for the effect of an error in r_1 on the period, or semi-period, of the orbit

$$T = 2\pi\sqrt{\frac{a^3}{K}};\quad dT = 3\pi\sqrt{\frac{a}{K}}\,da \tag{70}$$

$$\frac{dT}{dr_1} = 3\pi\sqrt{\frac{a}{K}}\frac{da}{dr_1}. \tag{71}$$

In both cases, the terms $\sqrt{K/a^3}$ and $\sqrt{a^3/K}$ are very small, indicating a high degree of insensitivity of μ and T to position errors.

The position of a body in an elliptic orbit is actually not only given by r, but also by η. However, an error in η cannot be made, unless a velocity error is made. Errors in η, therefore, belong in the category of velocity errors as discussed in Section 3.2.

Therewith, the error analysis of the elliptic orbit is completed. This analysis is the core of central force field analysis.

4. ERROR ANALYSIS OF THE HYPERBOLIC ORBIT

Although the hyperbola, like the ellipse, is strictly a central force field concept, the hyperbola is of interest primarily in conjunction with two-force field transfer.

The hyperbola can be visualized as transfer orbit from one central force field into another. The space vehicle "escapes" along a hyperbolic orbit. Planetocentric errors therefore refer to the hyperbolic path. They will affect the subsequent heliocentric orbit. This orbit, which generally is of elliptic shape, can then be treated according to central force field analysis.

In the following we will designate velocities and distances measured in the heliocentric system, by capital letters, all other symbols pertaining to the heliocentric system will have the symbol of the Sun ($\odot$) as subscript. Planetocentric symbols will be in small letters and without astronomical subscript. Let U be the velocity of the departure planet,[4] V_1 the heliocentric departure velocity of the space vehicle. The velocity vector V_1 is attained when the planetary attraction has become negligible compared to the heliocentric attraction. Geometric "planetocentric infinity" has been achieved when the vehicle's hyperbolic path has merged with one of the two asymptotoes of the hyperbola. This process and correlated symbols are explained in Fig. 8. The difference between planetary velocity U and V_1 determines the hyperbolic excess velocity v_∞ for which the vehicle performance must be laid out. The square of this velocity is by definition equal to the orbital energy constant of the hyperbola,

$$h = v_\infty^2 = U^2 \sin^2 \beta + (\Delta V_1)^2 \tag{72}$$

where

$$\Delta V_1 = |V_1 - U \cos \beta| \tag{73}$$

is the difference between U and V_1, and β is the angle between the velocity vectors U and V_1. In the case of cotangential heliocentric

[4] If a distinction between departure and arrival must be made, departure has the subscript 1, arrival the subscript 2, as before.

departure one has the well-known relation

$$V_1 - U = v_\infty. \tag{74}$$

The resulting hyperbolic departure velocity in planetocentric space is

$$v_1 = \sqrt{\frac{2K}{r_1} + v_\infty^2} \tag{75}$$

where v_∞ is determined either by Eq. 72 or by

$$v_\infty = \sqrt{U^2 + V_1^2 - 2UV_1 \cos \beta} \tag{76}$$

in the case of planar departure. In the case of non-planar heliocentric departure at inclination i and planar angle β,

$$v_\infty = \sqrt{U^2 + V_1^2 - 2UV_1 \cos \beta + (2V_1 \sin \tfrac{1}{2}i)^2}. \tag{77}$$

It should be pointed out that the hyperbola as escape path is a simplification inasmuch as solar gravitational torque during the escape was neglected. This effect plays a role, especially if the vehicle escapes at an angle with respect to the planet's motion, since in this case the vehicle's distance from the Sun varies. Since the Earth's field is not really a single force field inasmuch as it is superimposed over the gravitational field of the Sun, the solar torque changes slightly the path from that of a perfect hyperbola. This effect on Earth-Moon transfer orbits is treated in (5). If it were not neglected here, the formulation of closed mathematical expressions would not be possible. This would mean a considerable loss in lucidity without adding to the results of the error analysis.

Whether or not the cut-off point coincides with the vertex of the hyperbola (peri-apsis), this vertex is well defined by the vehicle's hyperbolic cut-off vector v_1. The vertex distance r_P and the hyperbolic excess v_∞ given, the eccentricity of the hyperbola is given by

$$e = 1 + \frac{r_P}{K} v_\infty^2. \tag{78}$$

If r_1, v_1 and θ_1 are given, the eccentricity follows from

$$e = \frac{v_1^2 \cos^2 \theta_1}{K/r_1} - 1. \tag{79}$$

The half angle of the asymptotes is given by

$$\sec \phi = e. \tag{80}$$

The angle ϕ determines the heliocentric departure direction with respect to the original planetocentric departure direction (direction of v_1 or v_P), *if* the heliocentric departure is cotangential to the planet's orbit. The above equations show that even if the planetocentric departure direction is correct, but the departure velocity (v_1) wrong, the angle ϕ will be changed and therewith the heliocentric departure direction. This is a significant difference from the single force field error analysis where only an error in direction, but not an error in scalar velocity at the apsides, can change the orbital departure direction (that is, the orientation of the major axis of the elliptic orbit). Since only small values of $\Delta\beta$ will have to be considered in practice, one can write (Figs. 9, 10, 11)

$$\cos(\Delta\beta) \approx V'/V_1 \tag{81}$$

$$V_1 = U + v_\infty \tag{82}$$

$$V_1' = U + (v_\infty + \Delta v_\infty)\cos\phi \tag{83}$$

$$\Delta V_r = v_\infty \sin(\Delta\phi) = v_\infty \sin\left[\frac{2\left(1 + \frac{v_\infty^2}{K/r_P}\right)}{v_\infty \sec^2\phi}\Delta v_P\right] \tag{84}$$

$$\Delta\beta \approx V_r/V_1 = \frac{v_\infty}{V_1}\sin(\Delta\phi) \approx \frac{v_\infty}{V_1}\Delta\phi \tag{85}$$

where ΔV_r is a small, roughly radial, velocity component caused by the change in direction $\Delta\beta$. This change is liable to be extremely small within the range of practical velocity errors of a few ft./sec.

The correct cotangential departure from a circular orbit is shown in Fig. 9, left. Figure 9, right hand side, shows the heliocentric angular error $\Delta\beta$ due to the planetocentric error $\Delta\phi$. In Fig. 10 the same process is shown for departure from a non-circular orbit. Finally, Fig. 11 shows correct and erroneous intersecting heliocentric departure. One has, therefore, for small errors $\Delta\phi$ planar heliocentric error

$$\Delta\beta = \frac{v_\infty}{V_1}\Delta\phi \tag{86}$$

and, likewise for a planetocentric orthogonal error $\Delta\alpha$ at hyperbolic departure the error in heliocentric orbital inclination is

$$\Delta i = \frac{v_\infty}{V_1}\Delta\alpha. \tag{87}$$

The first step is now to establish the equations which determine the effect of an error at planetocentric departure on the initial heliocentric

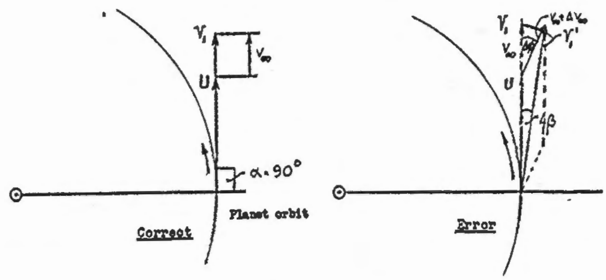


FIG. 9. Planetary orbit circular; V_1 parallel to U.

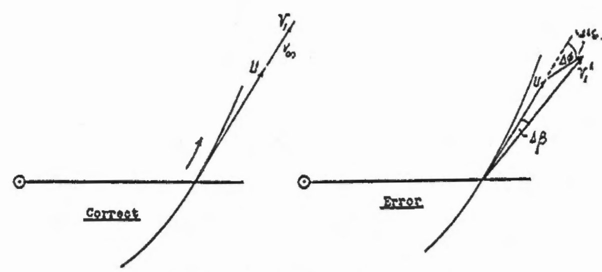


FIG. 10. Planet orbit elliptic; V_1 parallel to U.

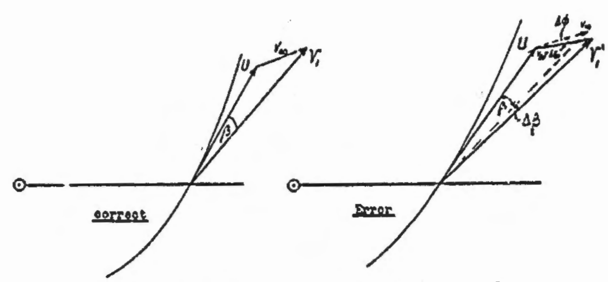


FIG. 11. Planet orbit circular or elliptic; V_1 not parallel to U.

flight conditions and to proceed from there, using elliptic single force field relations. One obtains for the change in ϕ (6) the following relations. If the cut-off conditions are v_1, θ_1, r_1 and if v_1 is scalarly in error, then

$$\frac{\partial \phi}{\partial v_1} = \frac{2\left(1 + \frac{v_\infty^2}{K/r_1}\right)\cos\theta_1}{v_\infty \sec^2\phi} \tag{88}$$

where ϕ is the original angle. For the peri-apsis conditions this relation becomes

$$\frac{\partial \phi}{\partial v_P} = \frac{2\left(1 + \frac{v_\infty^2}{K/r_P}\right)}{v_\infty \sec^2\phi}. \tag{89}$$

If the cut-off velocity v_1 is broken down into the radial and azimuthal components v_r and v_a, one finds, if either one is in error,

$$\frac{\partial \phi}{\partial v_{r_1}} = \frac{C}{K}\sin\eta_1\,(\sin\phi - \operatorname{cosec}\phi) \tag{90}$$

where $C = r_1 v_1 \cos\theta_1 = r_1 v_{a_1} = r_P v_P$ is the Keplerian area constant and η_1 the true anomaly of the cut-off point, and

$$\frac{\partial \phi}{\partial v_{a_1}} = \frac{C}{Kaer_1}(b^2 - r_1^2)(\operatorname{cosec}\phi - \sin\phi) \tag{91}$$

where $b = C/v_\infty$ is the semi-minor axis of the hyperbola. If both components are in error, the total differential becomes

$$d\phi = \frac{2v_{a_1}\left[v_\infty^2 dv_{a_1} + \frac{1}{2}v_{a_1}v_{r_1}dv_{r_1}\right]}{\left(\frac{K}{r_1}\right)^2 \tan\phi\sec^2\phi}. \tag{92}$$

If the tangential velocity v_1 is correct scalarly, but the path angle θ_1 is in error, one obtains

$$\frac{\partial \phi}{\partial \theta_1} = -\frac{1}{2}\frac{v_1}{v_\infty}\frac{r_1}{r_P}\frac{\sin\theta_1}{\sec^2\phi}\left[\frac{(K/r_P)\cos\phi}{\left(\frac{K/r_1}{v}\right)^2\tan^2\phi + \frac{K}{r_P}\cos^2\theta_1}\left\{3\sec^2\phi + 2\left(\frac{v_1^2\cos^2\theta_1}{K/r_1}\right)^2 + 1\right\} - 2\right]. \tag{93}$$

If v_1 as well as θ_1 are in error, the resulting change in ϕ is

$$d\phi = \frac{\partial \phi}{\partial v_1}dv_1 + \frac{\partial \phi}{\partial \theta_1}d\theta_1 \tag{94}$$

where the partials are given by Eqs. 88 and 93.

For the error in v_∞ one finds

$$\frac{dv_\infty}{dv_P} = \frac{v_P}{v_\infty} \tag{95}$$

$$\frac{dv_\infty}{dv_1} = \frac{v_1}{v_\infty} \tag{96}$$

$$dv_\infty = \frac{v_\theta dv_\theta + v_r dv_r}{v_\infty}. \tag{97}$$

Therewith the errors $d\phi$ and dv_∞ are known as functions of the departure errors dv_1, $d\theta_1$ or dv_{θ_1}, dv_{r_1} and the new heliocentric departure values β and V_1 can be computed.

5. CORRELATION OF HYPERBOLIC ERRORS WITH HELIOCENTRIC DEPARTURE VALUES

The second step in the two-force field error analysis is to correlate the geocentric departure errors in the hyperbola with the heliocentric departure values. For cotangential or nearly cotangential heliocentric departure ($\beta \simeq 0$) one finds (6)

$$\frac{dV_1}{dv_1} = \frac{v_1}{v_\infty} \quad (\beta \simeq 0). \tag{98}$$

This equation can be used with good accuracy in the case of cotangential or near cotangential departure and where small errors in v_1 (a few ft./sec. against several ten thousand ft./sec.) are involved, which do not change β significantly. If $\beta \neq 0$ one has

$$\frac{dV_1}{dv_1} = \frac{dV_1}{d\phi}\frac{d\phi}{dv_1} \quad (\beta \neq 0) \tag{99}$$

with the auxiliary equations

$$\frac{dV_1}{d\phi} = \left(\frac{\partial V_1}{\partial v_1}\right)_\beta \frac{dv_1}{d\phi} + \left(\frac{\partial V_1}{\partial \beta}\right)_{v_1} \frac{d\beta}{d\phi} \tag{100}$$

$$\left(\frac{\partial V_1}{\partial v_1}\right)_\beta = \pm \frac{v_1}{V_1 - U\cos\beta} \quad (V_1 \gtrless U) \tag{101}$$

$$\frac{dv_1}{d\phi} = \frac{1}{\dfrac{\partial \phi}{\partial v_1} + \dfrac{\partial \phi}{\partial \theta_1}\dfrac{d\theta_1}{dv_1}} \tag{102}$$

where $\partial\phi/\partial v_1$ follows from Eq. 88, $\partial\phi/\partial\theta_1$ from Eq. 93 (the two errors

$d\theta_1$ and dv_1 are not functionally correlated) and

$$\left(\frac{\partial V_1}{\partial \beta}\right)_{v_1} = \mp \frac{V_1 U \sin\beta}{V_1 - U\sin\beta}. \tag{103}$$

Finally, $d\beta/d\phi \approx v_\infty/V_1$ according to Eq. 85. The second term on the right hand side of Eq. 99 is the reciprocal of Eq. 102. Equation 99 may therefore be written in the form

$$\frac{dV_1}{dv_1} = \left(\frac{\partial V_1}{\partial v_1}\right)_\beta + \left(\frac{\partial V_1}{\partial \beta}\right)_{v_1}\frac{d\beta}{d\phi}\frac{d\phi}{dv_1}. \tag{104}$$

The second term is very small for small errors in v_1 (say 5–10 ft./sec.), so that it often may be neglected.

A direct tie-in with the elliptic error analysis for the heliocentric elements can be achieved by correlating the azimuthal and radial components, V_{θ_1}, V_{r_1}, of heliocentric departure velocity V_1 with the planetocentric departure velocity v_1. For the azimuthal component one has

$$\frac{dV_{\theta_1}}{dv_1} = \frac{dV_{\theta_1}}{d\phi}\frac{d\phi}{dv_1}. \tag{105}$$

The second term is given by Eq. 102, while the first term is (6)

$$\frac{dV_{\theta_1}}{d\phi} = \frac{dV_{\theta_1}}{dV_1}\frac{dV_1}{d\phi} \tag{106}$$

with the auxiliary equations

$$\frac{dV_{\theta_1}}{dV_1} = \cos\beta - V_1\sin\beta\frac{d\beta}{dV_1} \tag{107}$$

$$\frac{d\beta}{dV_1} = \frac{V_1 - U\cos\beta}{v_1\dfrac{dv_1}{d\beta} - UV_1\sin\beta} \tag{108}$$

$$\frac{dv_1}{d\beta} = \frac{U}{\dfrac{v_1}{V_1}\operatorname{cosec}\beta - \dfrac{dV_1}{dv_1}} \tag{109}$$

where dV_1/dv_1 is given by Eq. 99 and its auxiliary equations. The second term in Eq. 106 is given by Eq. 100. One may therefore write

$$\frac{dV_{\theta_1}}{dv_1} = \left[\cos\beta - \frac{V_1\sin\beta(V_1 - U\cos\beta)}{\dfrac{Uv_1}{\dfrac{v_1}{V_1}\operatorname{cosec}\beta - \dfrac{dV_1}{dv_1}} - UV_1\sin\beta}\right]\frac{dV_1}{dv_1} \tag{110}$$

where dV_1/dv_1 is given by Eq. 104 and preceding auxiliary equations.

Similarly, one may write

$$\frac{dV_{r_1}}{dv_1} = \frac{dV_{r_1}}{d\phi}\frac{d\phi}{dv_1} \tag{111}$$

where

$$\frac{dV_{r_1}}{d\phi} = \frac{dV_r}{dV_1}\frac{dV_1}{d\phi} \tag{112}$$

$$\frac{dV_{r_1}}{dV_1} = \sin\beta + V_1\cos\beta\frac{d\beta}{dV_1} \tag{113}$$

where $d\beta/dV_1$ is given by Eqs. 108 and 109. One may therefore write

$$\frac{dV_{r_1}}{dv_1} = \left[\sin\beta + \frac{\dfrac{V_1\cos\beta(V_1 - U\cos\beta)}{Uv_1}}{\dfrac{v_1}{V_1}\operatorname{cosec}\beta - \dfrac{dV_1}{dv_1}} - UV_1\sin\beta\right]\frac{dV_1}{dv_1}. \tag{114}$$

Planetocentric and heliocentric radial and azimuthal velocity components are easily correlated by the key equations

$$\frac{dV_{e_1}}{dv_{e_1}} = \frac{dV_{e_1}}{d\phi}\frac{d\phi}{dv_{e_1}} \tag{115}$$

where $d\phi/dv_{e_1}$ follows from Eq. 92,

$$\frac{d\phi}{dv_{e_1}} = \frac{2v_{e_1}\left[v_\infty^2 + \frac{1}{2}v_{e_1}v_{r_1}\dfrac{dv_{r_1}}{dv_{e_1}}\right]}{\left(\dfrac{K}{r_1}\right)^2\tan\phi\sec^2\phi}. \tag{116}$$

Likewise

$$\frac{dV_{r_1}}{dv_{r_1}} = \frac{dV_{r_1}}{d\phi}\frac{d\phi}{dv_{r_1}} \tag{117}$$

$$\frac{d\phi}{dv_{r_1}} = \frac{2v_{e_1}\left[v_\infty^2\dfrac{dv_{e_1}}{dv_{r_1}} + \frac{1}{2}v_{e_1}v_{r_1}\right]}{\left(\dfrac{K}{r_1}\right)^2\tan\phi\sec^2\phi}. \tag{118}$$

$dV_{e_1}/d\phi$ and $dV_{r_1}/d\phi$ are given by Eqs. 106 ff. and 112ff., respectively.

6. CORRELATION OF PLANETOCENTRIC ERRORS WITH THE ELEMENTS OF THE HELIOCENTRIC TRANSFER ORBIT

We are now ready to correlate the hyperbolic errors in planetocentric space directly with the elements of the heliocentric transfer orbit, that is with the equations given in Section 3. The elements pertaining to the heliocentric orbit will be designated by the subscript $\odot$.

Suppose one wants to know the effect of an error in v_r on the semi-major axis $a_\odot$ of the heliocentric orbit. Taking Eq. 14, re-writing it in heliocentric nomenclature,

$$\frac{da_\odot}{dV_r} = -\frac{2a_\odot^2 V_r}{K_\odot} \sin \eta_\odot \tag{119}$$

and multiplying it by Eq. 117 yields

$$\frac{da_\odot}{dv_r} = \frac{da_\odot}{dV_r}\frac{dV_r}{dv_r}$$

$$= -\frac{2a^2_\odot V_r}{K_\odot} \cdot \frac{2v_{e_1}\left[v_\infty^2 \frac{dv_{e_1}}{dv_{r_1}} + \frac{1}{2}v_{e_1}v_{r_1}\right]}{\left(\frac{K}{r_1}\right)^2 \tan\phi \sec^2\phi} \frac{dV_{r_1}}{d\phi} \sin \eta_\odot. \tag{120}$$

In the same manner one finds

$$\frac{de_\odot}{dv_r} = -\frac{C_\odot}{K_\odot}\frac{dV_r}{dv_r} \sin \eta_\odot \tag{121}$$

$$\frac{d\mu_\odot}{dv_r} = \frac{3}{a_\odot \sqrt{1 - e_\odot^2}}\frac{dV_r}{dv_r} \sin \eta_\odot \tag{122}$$

$$\frac{d\eta_\odot}{dv_r} = \frac{\cos \eta_\odot}{e_\odot \sqrt{h_\odot (1 - e_\odot^2)}}\frac{dV_r}{dv_r} \tag{123}$$

and for the effect of an azimuthal planetocentric error on the elements of the heliocentric transfer orbit

$$\frac{da_\odot}{dv_a} = \frac{2a_\odot^2}{K_\odot} V_a \frac{dV_a}{dv_a} \tag{124}$$

$$\frac{de_\odot}{dv_a} = \frac{C_\odot(b_\odot^2 - R_1^2)}{K_\odot a_\odot e_\odot / R_1}\frac{dV_a}{dv_a} \tag{125}$$

$$\frac{d\mu_\odot}{dv_a} = -\frac{3}{R_1}\sqrt{1 - e_\odot^2}\frac{dV_a}{dv_a} \tag{126}$$

$$\frac{d\eta_\odot}{dv_a} = \frac{2 + e_\odot \cos \eta_\odot}{e_\odot K_\odot / R_1}\frac{dV_a}{dv_a} \sin \eta_\odot. \tag{127}$$

Finally, if the direction of the planetocentric velocity vector is correct and only its scalar value in error, the following relations are obtained,

$$\frac{da_\odot}{dv_1} = \frac{da_\odot}{dV_1}\frac{dV_1}{dv_1} = \frac{2V_1 a_\odot^2}{K_\odot}\frac{dV_1}{dv_1} \tag{128}$$

$$\frac{de_\odot}{dv_1} = \pm\frac{2}{V_1}\left(\frac{V_1}{V_{c_1}}\right)^2\frac{dV_1}{dv_1} \quad \left(\frac{V_1}{V_{c_1}} = \frac{V_1}{\sqrt{K_\odot/R_1}} \gtrless 1\right) \tag{129}$$

$$\frac{d\mu_\odot}{dv_1} = -\frac{3V_1}{\sqrt{K_\odot a_\odot}}\frac{dV_1}{dv_1} \tag{130}$$

$$\frac{d\eta_\odot}{dv_1} = \frac{2}{e_\odot V_1}\frac{dV_1}{dv_1}\sin\eta_\odot \tag{131}$$

$$\frac{dR_A}{dv_1} = \frac{dV_1}{dv_1}\left[\frac{2V_1 a_\odot R_A}{K_\odot} \pm \frac{2a_\odot}{V_1}\left(\frac{V_1}{V_{c_1}}\right)^2\right] \quad \left(\frac{V_1}{V_{c_1}} \gtrless 1\right) \tag{132}$$

$$\frac{dR_P}{dv_1} = \frac{dV_1}{dv_1}\left[\frac{2V_1 a_\odot R_P}{K_\odot} \mp \frac{2a_\odot}{V_1}\left(\frac{V_1}{V_{c_1}}\right)^2\right] \quad \left(\frac{V_1}{V_{c_1}} \gtrless 1\right). \tag{133}$$

Additional relations can easily be obtained for errors in orthogonal or normal velocity components, using the results of Section 3.

7. EFFECT OF ARBITRARY PLANETOCENTRIC AZIMUTHAL AND RADIAL VELOCITY ERRORS ON HELIOCENTRIC DISTANCE

In the preceding section, two equations are formulated, giving the change in R_P or R_A as function of an error in v_1 in terms of V_1 and $a_\odot$. In this section the question as to a change in R at any value $\eta_\odot$ (that is, at any point in the transfer orbit) and as to the change in $\eta_\odot$ for a given arbitrary distance R, both as function of an error in planetocentric departure velocity v_1, will be answered.

Of particular interest is the latter question. It relates the heliocentric displacement of the spacecraft at the intersection with the target orbit (cf. Section 3) to the planetocentric error in departure velocity v_1.

In order to preserve a certain degree of simplicity, suppose the planet orbits are circular and heliocentric departure is cotangential. Then $V_1 = V_{c_1} \pm v_\infty$ is either the perihelion or the aphelion velocity of the transfer ellipse. In the following, we will make the velocities dimensionless by dividing them by $V_{c_1} = \sqrt{K_\odot/R_1}$. They are designated by an asterisk superscript. Then one has for the heliocentric departure velocity, remembering that $v_\infty^* = \sqrt{v_1^{*2} - v_p^{*2}}$,

$$V_1^* = 1 + \sqrt{v_1^{*2} - v_p^{*2}}. \tag{134}$$

Writing the polar equation, as in Section 3.1 for the heliocentric case, one has

$$\frac{R}{R_1} = \frac{V_c^{*2}}{1 + \chi(V_1)\cos\eta_\odot} \tag{135}$$

where $\chi(V_1)$ has the same definition as in Section 3.1. The use of this equation in conjunction with two-force field transfer can be greatly simplified if term containing V_r, is neglected in the expression for $\chi(V)$. Since indeed errors in v_1 will in practice be small enough (some 10–20 ft./sec.) not to cause a significant heliocentric radial component, the disregard of V_r from this view point will be permissible. The general validity of disregarding V_r is, however, limited to cases of cotangential or nearly cotangential heliocentric departure (that is, where the predetermined value of β is zero or very small, so that $V_r/V_c \ll 1$). With this restriction in mind we eliminate V_r whence $\chi(V_1)$ becomes $1 - V_{c_1}^{*2} = 1 - V_1^{*2}$ and Eq. 135 simplifies to

$$\frac{R}{R_1} = \frac{V_1^{*2}}{1 + (1 - V_1^{*2})\cos\eta_\odot} \tag{136}$$

or, because of Eq. 134

$$\frac{R}{R_1} = \frac{1 + \sqrt{v_1^{*2} - v_p^{*2}}}{1 + \cos\eta_\odot[1 - (1 + \sqrt{v_1^{*2} - v_p^{*2}})^2]}. \tag{137}$$

Solving for $\cos\eta_\odot$, one obtains

$$\cos\eta_\odot = \frac{1 - \sqrt{v_1^{*2} - v_p^{*2}} - \dfrac{R}{R_1}}{\dfrac{R}{R_1}[1 - (1 + \sqrt{v_1^{*2} - v_p^{*2}})^2]}. \tag{138}$$

With Eqs. 137 and 138 one has two relations defining the distance R (in terms of departure distance R_1), for a given planetocentric departure velocity v_1^*, as function of the true anomaly $\eta_\odot$ and defining the true anomaly $\eta_\odot$ at which a given distance R/R_1 (for example, the distance of the target orbit) is attained at a given departure velocity v_1^*.

Differentiating $\eta_\odot$ with respect to v_1^*, using Eq. 138 yields

$$\frac{d\eta_\odot}{dv_1^*} = \frac{2v_1^*}{\sqrt{v_1^{*2} - v_p^{*2}}} \frac{1 + \frac{1}{2}v_1^* - v_1^{*2} - v_p^{*2} - \dfrac{R}{R_1}(1 + v_1^{*2} - v_p^{*2})}{\sin\eta_\odot \dfrac{R}{R_1}[1 - (1 + \sqrt{v_1^{*2} - v_p^{*2}})^2]^2}. \tag{139}$$

Thus, for a given small error $dv_1^* \approx \Delta v_1^*$, the change in heliocentric true anomaly $d\eta_\odot \approx \Delta\eta_\odot$ at the intersection point with the target orbit at distance R is determined by Eq. 139. The resulting displacement is

$$\Delta\eta_{\circ}^{(\mathrm{rad})} = \frac{d\eta_{\circ}}{dv_1^*}\frac{\Delta v_1}{V_e}; \quad \Delta s = R\Delta\eta_{\circ}^{(\mathrm{rad})} \tag{140}$$

where Δs (Fig. 5) is the distance, along the orbit, between the correct and erroneous intersection.

Differentiating Eq. 137 yields

$$\frac{d\left(\frac{R}{R_1}\right)}{dv_1^*} = 2v_1 \frac{1+v_\infty^{*2}}{v_\infty^{*}} \frac{[1+\cos\eta_{\circ}\{1-(1+v_\infty^{*2})^2\}] + (1+v_\infty^{*2})\cos\eta_{\circ}}{[1+\cos\eta_{\circ}\{1-(1+v_\infty^{*2})^2\}]^2}. \tag{141}$$

This equation yields the change in R at a given true anomaly $\eta_{\circ}$ due to a planetocentric departure error dv_1^*. Replacing the differential by the difference term which is sufficiently accurate if the error $\Delta v_1/v_1 \ll 1$, the change in distance is given by

$$\Delta R = \frac{d\left(\frac{R}{R_1}\right)}{dv_1^*} R_1 \frac{\Delta v_1}{V_e}. \tag{142}$$

It will be remembered that the present discussion is based on the assumption that the heliocentric departure point represents very closely either the perihelion or the aphelion of the heliocentric transfer orbit. If one is not interested in the displacement dR at any arbitrary true anomaly $\eta_{\circ}$, but in the displacement of the opposite apsis, Eq. 141 simplifies to

$$\frac{d\left(\frac{R_{\mathrm{apsis}}}{R_1}\right)}{dv_1^*} = \frac{V_1^*}{V_1^* - 1}\frac{4v_1}{(2 - V_1^{*2})^2}. \tag{143}$$

Putting $R_1 = R_\oplus$, the mean distance of the Earth from the Sun, yields for the displacement of the opposite apsis

$$\Delta R_{\mathrm{apsis}} = \frac{d\left(\frac{R_{\mathrm{apsis}}}{R_\oplus}\right)}{dv_1^*} R_\oplus \frac{\Delta v_1}{V_e}. \tag{144}$$

It is $R_\oplus = 80.8184 \cdot 10^6$ n.mi., $V_e = 97{,}770$ ft./sec. Therewith for $\Delta v_1 = 1$ ft./sec. one obtains for the displacement of the opposite apsis

$$\Delta R_{\mathrm{apsis}} = 826.6179 \frac{d\left(\frac{R_{\mathrm{apsis}}}{R_\oplus}\right)}{dv_1^*}\left(\frac{\text{n.mi.}}{\text{ft./sec.}}\right). \tag{145}$$

The value of $d(R_{\mathrm{apsis}}/R_\oplus)/dv_1^*$ according to Eq. 143 is plotted in Fig. 12.

This plot is the two-force field counterpart of the single-force field error sensitivity of the elliptic orbit, plotted in Fig. 1. For $V_1^* = 1$ the differential (error sensitivity) becomes infinite. This is in agreement with the single-force-field results which show an infinite sensitivity at parabolic escape. Thus, the case $V_1^* = 1$ and $v_1 = 2K/r_1$ represent the contact points of the two analyses. It means that for small changes

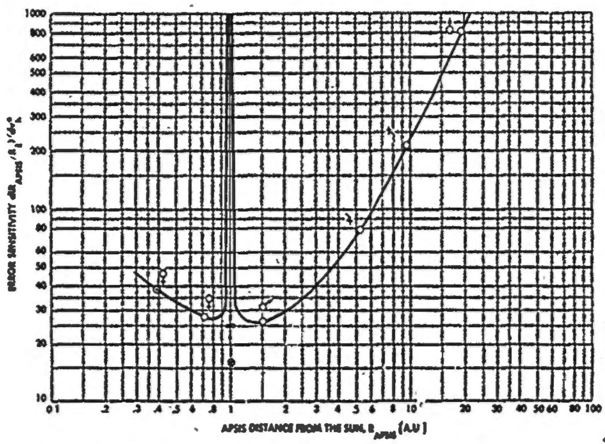


FIG. 12. Error sensitivity of a two-field transfer (Earth to planet).

of a vehicle orbit from the Earth's orbit the error sensitivity is extremely high. The sensitivity falls off quickly in both directions, closer and further away from the Sun. It is interesting to note that our two neighboring planets lie close to the ensuing minima which are followed by a renewed upswing of the two branches. The right hand branch reaches again infinity for $V_1^* = \sqrt{2}$, while the left hand branch passes through a maximum (not shown) and eventually is reduced to zero for vertical fall into the Sun, since Eq. 143 shows that for $V_1^* = 0$ the differential quotient becomes also zero. This result is of course correct only if $\beta = 0$. In general, Fig. 12 applies to the case $\beta = 0$ only.

8. DISCUSSION AND CONCLUSIONS

The preceding analysis has established the relations required to determine the effect of velocity vector errors on the orbital elements of the vehicle's flight path, both, if its motion is confined to a single central force field or if it escapes hyperbolically from one force field and enters another, such as in the case of interplanetary flight where the vehicle's path changes from that of a satellite (planetocentric motion) to that of a comet or a planetoid (heliocentric motion).

In the single force field motion the effect of radial errors decreases as the point of its occurrence approaches one of the apsides ($\sin \eta \to 0$) and approaches a maximum as the point approaches the end points of the latus rectum. In the case of azimuthal errors the trends are reversed. Exceptions are in both cases the effect on the position of the peri-apsis where radial errors have the greatest effect at the apsides and azimuthal errors the least effect. In terms of velocity, radial errors have the greatest effect on changing apsidal distances when the correct velocity is circular (that is, $v_r = 0$). Their effect lessens rapidly with increasing difference of path velocity from circular velocity. Azimuthal error sensitivity increases with increasing speed, being zero for $v = 0$ and infinite for $v = v_c\sqrt{2}$.

Flights involving two central force fields consecutively, show a more complicated pattern of error sensitivity, because errors in the final central force field are to a varying degree affected by errors in the hyperbolic escape motion from the original central force field. One significant difference from the single force field condition is the fact that whether or not the planetocentric cut-off velocity vector of the escaping vehicle is in error directionally, a directional error is incurred with respect to the heliocentric orbit if the planetocentric cut-off vector is in error as to its scalar value. The heliocentric radial error sensitivity with respect to planetocentric velocity errors in general is, however, quite low. The effect of planetocentric errors on the elements of the heliocentric orbit are a function of both, the true anomaly in the planetocentric as well as in the heliocentric path. The error sensitivity of heliocentric distances, especially aphelion and perihelion distances is very high. Based on Fig. 12, the sensitivity becomes infinitely large very near the Earth and again upon parabolic escape from the solar system.

Errors play a relatively minor role in the case of roving probes which do not attempt to meet another body in space, but rather measure environmental conditions along a flight path which leads them into a general area of interest, such as the solar probe. If another body, Moon or planet, is to be met at close distance, however, then not only the spatial displacements of the vehicle, but also the accompanying change in transfer time must be considered. Displacement of the vehicle and of the planet, both determine the effective miss-distance of the probe. Fortunately, the two errors always have a tendency to compensate each other. Indeed, if the probe has too high a departure velocity, the intersection point with the target orbit (that is, the probe's displacement) is counterclockwise (seen from the north pole of the orbit); but due to the resulting higher speed the transfer time is shortened and the planet therefore has not yet reached the original ("correct") intersection point. In other words, the planet's displacement is also counterclockwise. The same holds true in reverse if the departure velocity of the probe is too slow.

If the displacement of vehicle and planet is about the same, then velocity error and resulting time error about cancel each other and the miss-distance would remain small. For example, for near 180° transfer orbits to Mars an erroneous velocity excess of 1 ft./sec. would result in a counterclockwise displacement of the intersection point with the Mars orbit by the order of 400,000 to 600,000 n.mi. (depending on which part of the Martian ellipse is involved). The vehicle would arrive about 0.4 days earlier (roughly 10 hours). At a mean Martian velocity of about 48,000 n.mi./hr., the planet's counterclockwise displacement is about 480,000 n.mi., that is, of the same order as that of the vehicle. Thus, although both errors are large, they about cancel each other.

However, conditions are quite different in the case of fast transfer orbits. For example, for a transfer of about 160 days, the displacement of the intersection point is only about 18,000 n.mi. in counterclockwise direction for 1 ft./sec. heliocentric departure velocity excess, while the reduction in transfer time is about 3.6 hours, corresponding to a counterclockwise displacement of the planet by about 170,000 n.mi. per ft./sec. Here the time-induced miss-distance is approximately 9.5 times the intersection point displacement. It is therefore seen that the faster orbits show a greater error sensitivity than slower orbits, because the ratio of time-induced planet displacement to vehicle displacement is much greater.

In general, the error sensitivity of interplanetary transfer orbits is so high, that simple "ballistic" flights cannot be made accurate enough for a close encounter with other planets. Therefore a probe which is aimed at another planet must be equipped with a spaceborne navigation system and a small low-thrust (but not necessarily non-chemical) propulsion system for midcourse correction maneuvers (7).

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PROBING THE ATMOSPHERES OF VENUS AND MARS

BY

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I am not a professional astronaut. My interest began with a theoretical problem involving radiative heat transfer in the terrestrial atmosphere. It soon became apparent that certain aspects were susceptible to generalization. Without too much difficulty the terrestrial results could be transferred and applied to our neighboring planets Venus and Mars.

The talk will be divided into five parts: an introduction; the theory briefly described; the presentation of the data; an analysis; and prospects of things to come.

INTRODUCTION

There are numerous speculations that the first space cruise will not necessarily be to the moon, the most compelling practical reason being that of power. To land on the moon one will have to fight the lunar gravitational attraction, and to get off again the same problem in reverse applies. Quite recently an astronomer at Columbia University hinted that perhaps it would be more fruitful to attempt a landing on one of the tiny satellites of Mars.

Certainly I think that some form of planetary reconnaissance will be undertaken prior to an attempted landing on Mars or Venus. By reconnaissance one usually implies photo-reconnaissance. However, I wish to stress another as being of great importance: that of infra-red or thermal reconnaissance.

The radiation in the infra-red, since it arises from the thermal emission of the planet itself, gives us immediate information relating to the vertical temperature structure of the planetary atmosphere. So it becomes worthwhile to see what we know regarding the planetary atmospheres of Venus and Mars; second, to see whether we can't use what we already know to learn a little more; and, third, use what we know to tell us what we should look for.

THEORY

First, we must digress into an apparently unconnected subject, that of limb darkening of the sun. If one looks at the sun one will find that the disk is not uniformly illuminated, that there is a falling off of brightness as one approaches the edge or limb. This phenomenon is known as limb darkening.

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A moment's reflection provides the reason. In the solar atmosphere, the photosphere that we see, the temperature falls off with increasing height.

Now let us put ourselves back on Earth. If we look at the center of the solar disk, we see farther down into the solar atmosphere and our eye is intercepting radiation from the hotter solar depths. If, however, we look at the edge of the sun, our eye is intercepting photons arising from the cooler outer layers primarily; their temperature being lower, the intensity will be less.

The point to be made is that if one has a cooling of a spherical atmosphere with increasing radius this will be perceived by an observer elsewhere as an apparent limb darkening or falling off in intensity as one moves from the center to the edge of the disk.

That example alone suffices to tell us qualitatively the conclusion we are groping toward, namely, if I have a knowledge of how the intensity varies across the disk, then I can work backward to get some insight as to the vertical temperature structure of the atmosphere.

The connection between these two is qualitatively easy to grasp, but quantitatively much more difficult. At this point in the theory, the name of Chandrasekhar looms like a giant across the mathematical landscape.

Chandrasekhar, a distinguished service Professor of Astrophysics at the University of Chicago, was able to give a complete mathematical solution to the following problem: given the limb darkening, what is the vertical temperature distribution under the condition of radiative equilibrium?

This he did by using an invariance method first formulated by the Soviet-American astrophysicist Ambartsumian, and applying it to a stellar semi-infinite atmosphere. Chandrasekhar, being an astronomer, was primarily interested in interpreting the solar case. This has to be altered in two important ways before application to the planets. First, in the sun one finds that the absorption coefficient is roughly constant over frequency. This is called a gray atmosphere, an atmosphere in which the emission or absorption is independent of frequency. This condition maintains quite well in the solar case. However, it is patently out of step with what we know of planetary atmospheres.

The second fundamental way in which the theory of Chandrasekhar must be altered is by treating a finite instead of a semi-infinite atmosphere. By semi-infinite is meant an atmosphere in which one cannot see down to the bottom. The sun is a case of this par excellence, but certainly as we look at the Earth from outside or at Mars, we clearly see down to their surfaces. Therefore, we must consider a finite, rather than a semi-infinite atmosphere.

Figure 1 is a plot of this emerging radiation intensity as one scans across a planetary or stellar disk. The steepest line ($\tau_1 \rightarrow \infty$) represents the semi-infinite case of the sun. The ratio between the brightness at the center of the disk and the edge is about 2.9.

For a finite atmosphere one gets a shouldering effect. This is due to the fact that as the center of the disk is approached, the eye is intercepting more and more photons emitted from the isothermal surface. In the limiting case of zero atmosphere thickness, the intensity curve becomes a step function.

As mentioned earlier, the gray atmosphere assumption, valid for the sun, is totally inapplicable for planetary atmospheres. Their emission bands in the far infra-red spectral region partake of a discrete line character rather

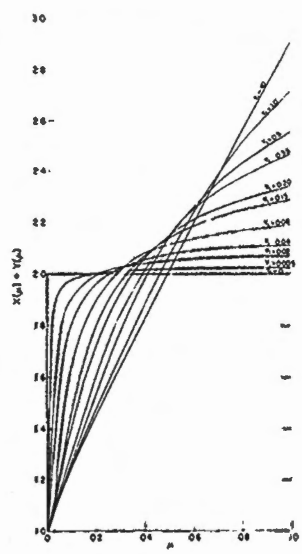

FIG 1.

than a smooth band continuum. Manifestly any theory which aspires to validity or completness must incorporate *ab initio* the wildly fluctuating or non-gray character of the absorption coefficient.

Figure 2 shows the effect of non-grayness. The parameter beta is a measure of this non-grayness. The limiting beta equals infinity case is that of a gray solar-type atmosphere. We see as our absorption coefficient becomes more and more jagged, that is, partakes more and more of a discrete line character, the limb darkening effect becomes more and more pronounced.

Thus, the effect of non-grayness is to enhance the contrast between the center of the disk and the edge, until in the limiting case, beta approaching zero, the center would be infinitely more intense than the edge. In that event, there is a continuous intensity fall-off towards the planet's edge. This gradual fading is in contrast to the sharp edge of the disk-like image that we see visually in the sun.

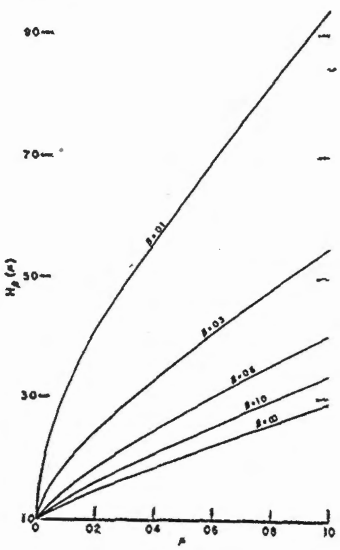

FIG. 2.

PRESENTATION OF DATA

Now let us leave theory for observational facts as far as we know them. Figure 3 is taken from Kuiper's book "The Atmosphere of the Earth and Planets" [2] and gives physical data for the planets. There are a few columns here that are particularly worthy of note for our purposes.

Notice the albedo, that is, the reflectivity of the planets. Venus is highly reflecting; the Earth less so; Mars lesser still.

It is interesting to observe that the Martian albedo is known to three significant figures. The Earth's is given as 0.39. However, the terres-

[2] Gerard Kuiper, "The Atmosphere of the Earth and Planets," 366 pages, plates, 6¾ × 9¾ in. Chicago, University of Chicago Press, 1949.

trial albedo is not even known to within 5 or 6 per cent. It is rather remarkable that we know the albedo of Mars to three places while that of the Earth is not known to within two. Here the advantage of looking at a system from the outside is apparent and constitutes a good argument for use of our own earth satellites for determining this inaccurately known terrestrial albedo.

Another item of passing interest is the color index. The sun is taken as 0 5, so a figure greater than one-half implies a reddish planet; less than one-half, a bluish light. Again, the color index of the Earth is not known. Venus is definitely not whitish, but tends more to the yellow. The high

PLANETS AND PRINCIPAL SATELLITES' PHYSICAL DATA

Body	Mass*	Pk. Mass (Per Cent)	Mean Radius†	Mean Density (cgs)	Albedo A	Color Index ⊙ = 0 50	g	V_e (Km/Sec)	V_e' (Km/Sec)	T_M °K	T_A °K
Mercury	0 0543	0.7	0 38	5 46	0 058	0 87	0 38	4 3	3.4	625	.
Venus	0 8136	0 05	0 961	5 06	.76	0 80	0 88	10 4	12 0	324	229
Earth	1.0000	. . .	1 000	5 52	.39	. . .	1 00	11.3	12 0	349	246
Mars	0 1069	0.25	0 523	4 12	.148	1.30	0 39	5.1	5 8	307	217
Jupiter	318 35	0 00	10 97	1.35	.51	0 88	2 65	61 0	100	145	102
Saturn	95.3	0.1	9 03	0.71	.50	1 07	1.17	36.7	70.	107	76
Uranus	14 54	0 03	3.72	1 56	.66	0 50	1.05	22 4	54.	69	49
Neptune	17.2	1.	3 38	2 47	.62	0 36	1.23	25 5	67.	56	40
Pluto	0 033?	.	0 45	2?	.16	0 63	0 16?	3.2	8?	60	42
Moon	0 0123	0 03	0 273	3 33	.072	0 84	0 16	2.4	2 4	387	274
I	0 0121	8	0.255	4 03	.57	1.06	0 19	2 5	4.1	140	99
II	0 0079	2	0 226	3 78	.60	0.77	0.16	2.1	3 6	137	97
III	0 0260	1	0 394	2.35	.34	0 64	0.17	2 9	4 6	156	110
IV.	0 0162	8	0 350	2 06	.15	0 68	0 13	2 4	3 7	166	117
Mimas	0 000006	3	0 04	0 5	.8:	. . .	0 00	0 2	0.4	85	60
Enceladus	0 000014	20	0 05	0.7	.8:	0 52	0 01	0.2	0 4	85	60
Tethys	0 000109	3	0.08	1 2	.8	. . .	0 02	0 4	0.9	85	60
Dione	0 000176	3	0 07	2.8	.8:	0 64	0 04	0 6	1.3	85	60
Rhea	0 00038	35	0.102	2 0	.8	0 66	0 04	0 7	1 5	85	60
Titan	0 0235	1	0 377	2.42	.27	1.17	0.17	2 8	5.2	118	83
Iapetus	0 00024	35	. . .	. .	var	0 59	. . .	. .	. . .	. . .	.
Triton	0 022	18	0 4?	2?	0 2?	0 72	0.14?	3?	6?	68	48

* Mass of the earth = 5 975 × 10[27] gm; of the sun, 332 488 (1 ± 0 00013)E = 1 987 × 10[33] gm, of the moon, 0 0122269 ± 0 000004)E = 7 343 × 10[25] gm.

† Mean diameter of the earth is 12,742 46 km, equatorial diameter, 12,756 78 km, polar diameter, 12,713 82 km.

[illegible]

FIG. 3.

color index of Mars yields its characteristic ruddy glow known to all of us.

However, the fact that Venus, even though it is less red than Mars, is still reddish, or more accurately, yellowish, is quite important because that very fact alone rules out, according to Kuiper, the speculation that the cloud cover or the obscuration that is characteristic of the Venus disk can be due to water evaporation. Water droplet clouds because of their reflectivity in the visible would result in an index closer to one-half.

The last column of interest to us is the gravitational acceleration for the planets: Venus about that of the Earth, Mars 0.4 that of our planet.

Figure 4 shows the atmospheric constituents. The unit here is the centimeters *NPT*, which is the thickness of gas if all the atmosphere in a uniform column were reduced to constant sea level pressure and temperature.

The aspect of Venus that strikes us immediately is the literally tremendous amount of carbon dioxide in the atmosphere, about five hundred times as much as that on the Earth.

Nothing else has been detected on the Venus atmosphere. The "less than" figures merely are the upper-limiting figures, set by the resolution of the observations.

PROVISIONAL TABLE OF ATMOSPHERIC COMPOSITIONS
(McDonald Spectra)

Object	Gas	Amount* (In Cm N.P.T.)	Band Used (μ)	p (Lab.) (Cm)
Venus	CO_2	100,000	Weak IR	9.
	CO	<100	2.35	10.0
	N_2O	<100	2.15	75
	CH_4	<20	1.16, 1.7	4.5
	C_2H_4	<3	1.68	2.6
	C_2H_6	<1	1.20, 1.73	7.0
	NH_3	<4	1.51	1.7
Mars	CO_2	440	1.57, 1.60	Atm
	SO_2	<0.003	0.30	8.
	O_3	<0.05	0.30	Atm
	N_2O	<200	1.52, 1.68	75.
	CH_4	<10	1.16, 1.7	4.5
	C_2H_4	<2.	1.68	2.6
	C_2H_6	<1.	1.20, 1.73	7.0
	NH_3	<2.	1.51	1.7
Jupiter	CH_4	15,000	0.6–0.9	74.
	NH_3	700	0.645	44
Saturn	CH_4	35,000	0.6–0.9	74.
	NH_3	<250	0.645	33
	O_3	<0.1	0.30	Atm
	SO_2	<0.01	0.30	8
Uranus	CH_4	220,000	0.5–0.9	76.
	O_3	<0.1	0.30	Atm
	SO_2	<0.01	0.30	8
Neptune	CH_4	370,000	0.5–0.9	76.
Moon	SO_2	<0.0003	0.30	8.
	O_3	<0.005	0.30	Atm
J I–IV	CH_4	<300	0.887, 0.726	4.5
	NH_3	<40	0.786, 0.792	3.5
Titan	CH_4	20,000	0.6–0.9	74.
	NH_3	<300	0.645	33
Earth	N_2	625,000	[illegible]	[illegible]
	O_2	166,000	[illegible]	[illegible]
	CO_2	220	[illegible]	[illegible]
	CH_4	1.2	[illegible]	[illegible]
	N_2O	0.4	[illegible]	[illegible]
	O_3	0.3	[illegible]	[illegible]

* Assumed average path length 2 air masses for Venus and the Jovian planets (extended disks), but [illegible] air masses for the satellites.

FIG. 4.

For Mars the only constituent which has been positively identified is that of carbon dioxide, having roughly about two times the amount of our terrestrial atmosphere. No oxygen, nor strangely enough, no water vapor has yet been detected spectroscopically, although it is quite definitely known that the polar caps which cover the Martian poles in their wintertime are of a water-frost character.

However, the surface pressure of Mars has been estimated from other techniques—spectral line broadening, polarization measurements, etc.—to be about 90 millibars, roughly 1/11 the atmosphere of the Earth. Therefore, a good part of the Martian atmosphere must consist of optically inactive gases, probably nitrogen, which is not detectable easily spectroscopically, and perhaps argon, if one argues by analogy with the terrestrial atmosphere

ANALYSIS

It would be desirable in studying Mars to make use of the theory to deduce the vertical thermal structure from the limb darkening scan. Unfortunately, the small size of the Martian disk and its small emission in the far infra-red render this difficult. Gerard de Vaucouleurs, perhaps the

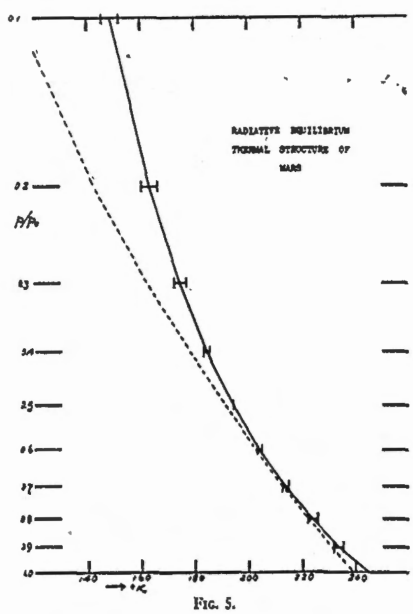

FIG. 5.

foremost authority on Martian atmosphere, has said that the main conclusion one comes to after years of visual study of the planet is that Mars is a mighty small object and therefore any detail is extremely hard to verify.

The vertical thermal structure of Mars, displayed in Fig. 5, is a theoretical calculation based on assuming the main absorbing constituent is the 15-micron band of carbon dioxide. This is a reasonable assumption and actually the theory should be more valid than the terrestrial case because of the fact that water vapor and ozone which are complicating factors in the terrestrial case, are not appreciably present.

The pressure of the Martian atmosphere is plotted against the temperature. Dashed in is the adiabatic lapse rate for the Mars atmosphere, which gives some measure of the vertical stability of the atmosphere.

For example, if the temperature slope under radiative equilibrium is steeper than adiabatic, as in the lower levels, one would expect an unstable configuration of cold air down over warm leading to vertical overturning. Up higher, where the lapse rate is less than the adiabatic, the atmosphere should be stable against convective overturning.

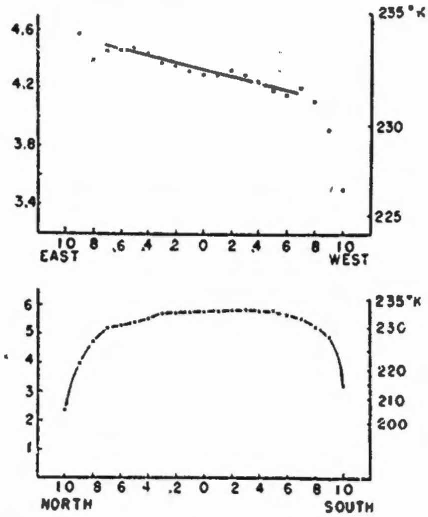

FIG. 6.

It is almost embarrassingly fortuitous that the height of this convective layer turns out to be about 4 kilometers. This accords very nicely with visual observations of yellow dust clouds which occur occasionally in the equatorial regions of Mars.

One automatically thinks by analogy that these are Sahara type dust storms. They are yellow and they are transient. They tend to subside with sundown on the Mars surface. So it is a tempting speculation to make that these yellowish clouds we see on the Martian landscape are due to dust storms quite similar in origin to what we find here on Earth.

Beyond that it is dangerous to go. Students of Martian literature know the troubles involving cloud layers. There are the blue layers and the violet haze layers, etc. These are very difficult to explain.

Quite recently Urey, the Nobel Prize winner, speculated that due to ultraviolet radiation of the sun one might get certain free radicals in the Martian atmosphere which leads to possible fluorescence, and that these upper violet haze layers are such a fluorescent phenomenon.

Venus gives us a little more hope. These measurements displayed in Fig. 6 were made by William Sinton using the 200-in. Hale Telescope. They are infra-red scans, the thermal emission of the Venus atmosphere itself. On the vertical you have the temperature of the atmosphere plotted against a disk scan from east to west. As one scans across the Venus disk toward sunset, one finds, exactly as expected, that the temperature is falling as the angle made with the sun, the zenith angle, increases.

Below, we have an equatorial scan from pole to pole. One finds that the temperature is lower at the pole, increases roughly symmetrically to some maximum, levels off actually for quite a wide region, and then falls off again as one approaches the south Venus pole.

In these scans the limb darkening effect has been removed and the limb darkening effect is precisely what we need to test our theory. Figure 7 shows precisely this, the experimental observations of the limb darkening scan across Venus.

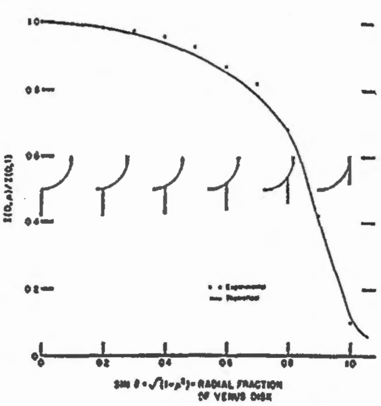

Fig. 7.

The dots are experimental points. The curve is the theoretical curve for a value of this parameter beta, equal to one-tenth. The first point to be made is that there is no shouldering effect. Recalling the theory, this means that we are not seeing down to any opaque level in the Venus atmosphere. We are still intercepting radiation arising from the gaseous atmosphere. There is no penetration down to a cloud level or a dust level or the Venus planetary surface itself.

The second point to be made is that these points are fitted quite well by assuming a line discreteness parameter of beta equal to one-tenth. The limb darkening here from center to edge is about a factor of 10. That means that we have a non-gray atmosphere, that we are "seeing" thermal radiation arising from discrete lines rather than a continuous type absorption coefficient characteristic of stellar gray atmospheres.

You might inquire of what importance is this—we knew it all the time. And it certainly is true that one would be very surprised at a contrary result since the Venus atmosphere emission bands must be very similar to those of the Earth itself.

FUTURE PROSPECTS

I think that the main point to be drawn from this is that a measurement of a single parameter, that is, the scan of intensity across a planetary disk, yields a great deal of information. One can determine the amount of grayness, the line structure of the atmosphere; and also the thickness of the atmosphere. You can also determine the actual temperatures of the atmosphere by matching it against black bodies of known thermal emission.

These are certainly cogent reasons for infra-red observations in any reconnaissance flight coursing about Venus or Mars.

Of course, we can turn the problem inside out and use this limb darkening scan to look back on ourselves; in other words, use it as a tool for probing the upper terrestrial atmosphere. This one does by attaching some infra-red device, such as a spectrometer, to a satellite.

The first attempts in this direction are being made by Professor John Strong at the Johns Hopkins University. He hopes eventually for satellite instrumentation. The instrument problems are quite large. At present, he is flying an infra-red spectrometer using large plastic balloons. He has a device which scans along the surface of a cone so that at one portion of the cycle it is looking within 30 degrees of nadir, directly down, and at the top it is looking within 30 degrees of zenith.

He made one flight last November. The data are not yet evaluated, but the method is very promising for determining the upper atmospheric temperatures in a quite direct way.

For instance, let us dream a bit and see what we could do if we had ideal instruments on satellites. An instrument which intercepted radiation in the 9.6 region alone would measure the temperature of the ozone region.

We could have a band pass filter at 15 microns to tell us the vertical distribution and temperature of carbon dioxide.

Similarly, by strategic placing of the filters we could do the same for the water vapor bands; and, finally, just by looking through a window of the atmosphere at 10 microns, where there is no absorption due to any atmospheric constituents, we would get the thermal radiation arising directly from the planetary surface itself.

One can visualize this sort of instrumentation on the satellite, and we could have then continuously monitored, world-wide temperature scanning of the atmosphere as well as the surface itself.

The first steps have been made in this direction of reasonable satellite instrumentation. A primitive prototype was made a couple of years ago by Strong. He simply converted an ordinary Weather Bureau radiosonde used for atmospheric sounding, by taking the humidity channel and replacing that channel with an infra-red signal. Unfortunately, even with miniaturization, the apparatus was still too weighty for the Vanguard But future improvements, and already there are some technical advances, show that this system has much promise.

I hope I have given you some feeling for what one can do with the rather exciting possibilities of infra-red reconnaissance of our neighboring planets in the not-too-far distant future.

SPACE MEDICINE—THE HUMAN BODY IN SPACE

BY

DAVID G. SIMONS[1]

INTRODUCTION

The problem of placing man into space is a subject of increasing interest to everyone. During the past ten years, I have been privileged to participate in research programs looking in that direction. As a research scientist primarily concerned with identifying and studying the problems man will face when he first ventures into space, it is apparent that a crucial question concerns the criteria for qualifying and selecting the individuals. One of the major challenges of General Don Flickinger's Air Research and Development Command Life Sciences research team is to determine what tests should be applied for selecting and training such a man. Let us examine some of the conditions and problems which our space men of the future must face.

In thinking of space, we must get in the "swing" and become "space" oriented. There is an interesting mental exercise that may help to put your minds into space where they should be for this lecture. Imagine you are in orbit circling the Earth, and your best friend is following one mile behind in the same orbit. If you want to toss him a ham sandwich so that he will catch it, how should you do it?

The simplest solution is simply to toss it straight forward in the opposite direction from where you want it to go eventually. It starts forward, rises upward, gradually drifts backward going overhead, and eventually lands in your friend's hand half an orbit later on the opposite side of the Earth!

This somewhat anomalous situation is further emphasized by what happens if you thoughtlessly decide to toss something down toward the ground out of your orbiting satellite. Its performance is equally strange. It starts downward, slowly drifts forward and then upward. Finally it returns to where is started, hitting you on the head.

In space you must consider carefully, then, how you dispose of what.

CONDITIONS IN SPACE

Medically, the high altitude situation becomes progressively equivalent to the conditions in space as one ascends. Thus there is no one altitude at which space begins for man. Rather, the situation becomes space equivalent in more and more ways as one goes higher and higher until there is essentially no difference.

For instance, very few who are acclimated to sea level pressure would

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survive above 20,000 feet without supplementary oxygen, or survive even while breathing 100 per cent oxygen above 45,000 feet without some pressurization technique. Medically, these altitudes already represent space equivalence with regard to oxygen.

Above 100,000 feet, nearly twenty miles, the atmosphere is so thin there is practically no conductive heat transfer between the atmosphere itself and an object in it. As a result the balance of the heat radiated by the object at high altitude and the heat received from radiation sources such as the sun and Earth determines its temperature almost exclusively. Thus, at 100,000 feet we are in space so far as temperature problems are concerned.

The biologically most effective cosmic rays, heavy cosmic ray primaries, upon entering the atmosphere of the Earth are converted to secondaries in the region between 15 to 25 miles and thereby change their nature. That is roughly between 80,000 and 130,000 feet. The higher one ascends through this transition region, the more nearly one receives the full exposure that would be experienced in space. The lighter primary particles penetrate more deeply into the atmosphere than the heavies.

One would not expect any hazard from collision with a meteorite below approximately 80 miles. At this level two other interesting phenomena occur. There is no longer any illumination from the sky. It will present the pure, lightless black of night, even in the daytime. At this altitude, also, there is no longer sufficient air to transmit sound. Biologically, therefore, above 80 miles you *are* in space.

There is one other characteristic of orbital space flight which requires more than altitude. Weightlessness requires velocity. Unless some special means are taken to provide a pseudo-gravity that is, spin, a stable satellite will produce a truly weightless environment.

An orbiting manned satellite must be high enough to be sufficiently free of atmospheric frictional drag to remain aloft, yet low enough to stay under the "van Allen" radiation. This will probably place early manned satellites in the region between 100 to 150 miles.

Let us now consider details of four specific medical problem areas which are of particular importance in terms of manned space flight: cosmic radiation, weightlessness, provision of a sealed cabin environment, and man's psychological reactions to the strange environment of space.

COSMIC RADIATION

Primary cosmic rays consist of atomic nuclei nearly all of which range in weight from hydrogen through iron. Most of the particles arrive in the vicinity of the Earth's orbit from interstellar space apparently accelerated by the magnetic fields within the spiral arm of the galaxy that encompasses our sun. Some of the low energy low weight primaries likely originate from our own sun. The arriving nuclei have been stripped of all electrons, leaving a strong positive charge that interacts with the Earth's magnetic field. Nuclei of the two lightest elements, hydrogen and helium, comprise approximately 96 per cent of the total, while the nuclei of medium and heavy elements comprise the remaining 4 per cent.

The least energetic cosmic ray primary nuclei arrive with nearly one billion electron volts energy per nucleon. Figure 1 relates the energy

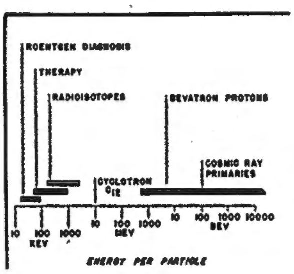


FIG. 1. Relative position of cosmic ray primaries with respect to other forms of familiar radiation in the energy spectrum. KEV is thousands of electron volts, MEV is millions of electron volts, and BEV is billions of electron volts per particle.

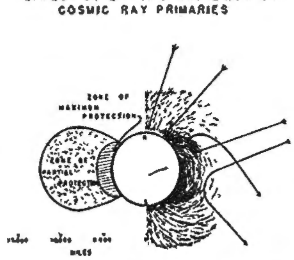
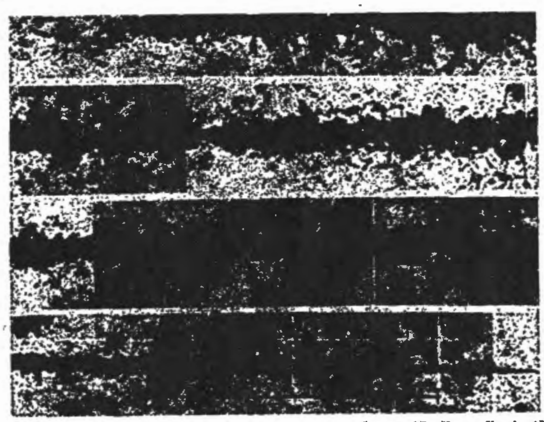


FIG. 2. The central circle represents the Earth. The shading at the right represents the magnetic field of the Earth, and the arrows show the paths of low energy heavy cosmic ray primaries as influenced by this magnetic field. The resultant zones of partial and complete protection are illustrated in terms of the scale of the Earth's diameter. The stippled area on the surface of the Earth represents the thin shell of the Earth's atmosphere.

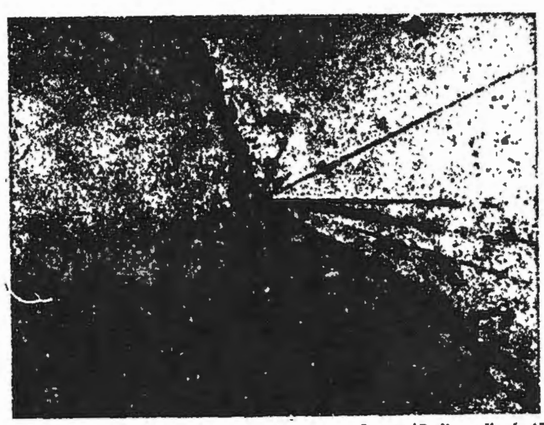
range of cosmic ray primaries to the energy spectra of familiar forms of radiation. The energy spectrum of cosmic radiation extends through eighteen decimal places. The distribution with respect to energy favors low energy particles strongly in the same manner that the distribution with respect to atomic weight strongly favors low atomic weight elements. Thus, the great bulk of incoming cosmic ray primary particles consists of low energy hydrogen and helium nuclei. Low energy heavy primaries are of greatest biological concern.

Since the path of low energy particles is strongly influenced by magnetic fields, magnetic considerations influence strongly the pattern of exposure with respect to particle energy. From the biological standpoint, the low energy heavy primaries are of particular interest. Figure 2 illustrates the fact that low energy primaries approaching the Earth in equatorial regions are frequently deflected back into space, whereas primaries approaching polar regions penetrate to the atmosphere.



Courtesy of Dr. Herman Yagoda, AFCRC

FIG. 3a. Heavy cosmic ray primary thindown recorded in nuclear emulsion by Dr. Herman Yagoda of AFCRC. The particle traveled from left to right and successively from top to bottom, terminating in the lower right hand corner as an atom of atomic weight close to iron.



Courtesy of Dr. Herman Yagoda, AFCRC

FIG. 3b. Collision of a relativistic particle approaching the weight of iron with a nucleus in a nuclear emulsion track plate. This photograph is on the same scale as Fig. 3a.

Thus, a zone of protection from exposure to low energy primaries exists in the equatorial region near the Earth and some distance into space.

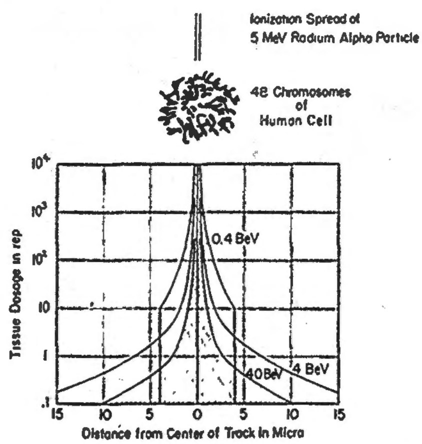
Those primaries which penetrate the Earth's atmosphere undergo ionization and collision producing the secondary cosmic radiation. This consists of familiar nuclear components such as neutrons, protons, electrons and other nuclear debris. All primaries are converted to secondaries at an altitude of approximately 75,000 feet. By the time primaries have reached this level, they have either lost their energy by ionization or collided with the nuclei of air atoms.

The low energy heavy primaries which penetrate a photographic emulsion plate and terminate by ionization, produce what has been termed a thindown illustrated in Fig. *3a*. When penetrating the atmosphere, tissue, or other absorbing material, heavy nuclei can also undergo a second type of collision in which they collide directly with nuclei of absorber atoms. If such a nuclear collision takes place in an atom of heavy atomic weight (silver or bromide) in a photographic emulsion, a stellate pattern is produced which is called a star. The spray of radiation characterizing stars adds up to less than one half the generally accepted permissible 0.3 roentgen equivalent physical (r.e.p.) dose per week, as illustrated in Fig. *3b*.

In order to establish a basis for estimating the biological effects associated with the thindown phenomenon, Schaefer calculated that whenever a particle produced more than 10,000 ion pairs per micron of path length in tissue (ip/μ_T), it would be biologically damaging. He therefore defined a traversal by such a particle a pre-thindown "hit." His calculations indicate a carbon atom can produce 10^4 ip/μ_T for a distance of 900 microns; an iron nucleus produces 10^4 ip/μ_T along nearly 10,000 microns (1 cm.) of its path length. Figure 4 illustrates the radial spread calculated by Dr. Schaefer based on this hit criterion. It should be noted that the maximum diameter of radial spread theoretically observable at a dose level of 100 (r.e.p.) is 5 microns.

In order to relate the significance of exposure to cosmic radiation within the atmosphere and within tissue, in terms of its ability to penetrate, the concept of grams per square centimeter (g/cm²) of absorber is extremely helpful. As a useful rule of thumb the most important factor is not so much what kind of material the radiation penetrates, but rather, the total mass penetrated. Thus, the mass of absorbing material can be conveniently expressed in terms of the number of grams of material traversed per square centimeter of area regardless of distance. For example, the absorption effect upon a primary traversing 2.7 centimeters of water at a specific gravity of 1 would be very nearly the absorption effect of the same primary traversing 1 centimeter of aluminum with a specific gravity of 2.7—2.7 g/cm² of absorber being traversed in each case. The air remaining above 135,000 feet also accounts for approximately 2.7 g/cm² of absorption, the numerical value of absorber to the top of the atmosphere in g/cm² being fortuitously almost exactly the same as the pressure expressed in millibars.

The calculations of Dr. Schaefer show an interesting phenomenon. Due to the high energy of primary particles and the fact that the region of greatest energy release is at the terminal portion of the tracks, at high altitudes in the vicinity of the Earth there is a region near the surface of the top of an exposed body that receives no thindown hits because the particles penetrate this region before slowing down enough to reach the ionization density of 10,000 ip/μ_T. Dr. Schaefer devised a standard sphere which is of equivalent weight to a man, for simplification of



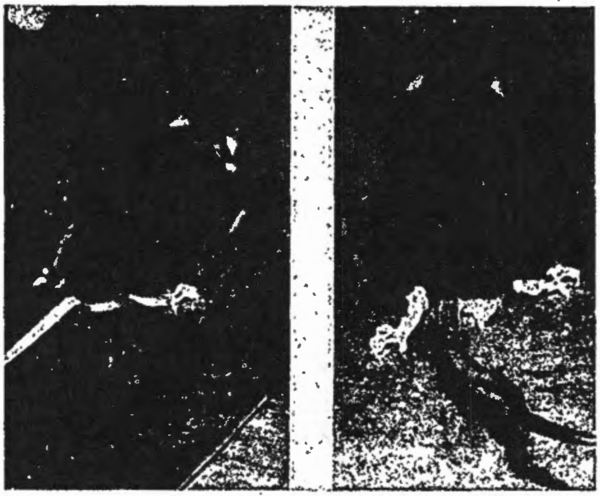
RADIAL SPREAD OF TISSUE IONIZATION DOSAGE ABOUT HEAVY NUCLEUS TRACK OF Z=20

Courtesy of Dr. Herman J. Schaefer, Naval School of Aviation Medicine

FIG. 4. Calculated values for the intensity of ionization produced by a calcium ion undergoing thindown. The intensity of radiation at various distances from the center of the track is compared to the size of a human cell.

geometry, and points out that even in the simple geometric configuration of a sphere there is a highly complex dose pattern throughout the sphere. It is hopelessly difficult and impractical to compute dosage accurately for a complicated and variable configuration such as the human body. We have approached this problem in terms of actually exposing animals to altitude and then determining the effects of primary cosmic radiation on them. To do this, we exposed black mice directly to heavy primary particles.

Dr. Herman Chase of Brown University had conducted ground experiments for several years on the effects of X-rays on graying of hair on black mice. He found that if a black mouse is exposed to anything less than 100 r.e.p. there will be no noticeable effect on hair graying. However, dosage levels between 100 and 1000 r.e.p. produce an increasing incidence of gray hairs in the hair coat, and at the 1000-r.e.p. levels, the entire exposed area of the animal hair coat turned gray. The figure on the left in Fig. 5 shows a normal C 57 strain black mouse with



Courtesy of Dr. Herman B. Chase, Brown University

FIG. 5. Normal C 57 black hair mouse on the left, and on the right, animal showing scattered clumps of gray hairs following exposure at 90,000 feet.

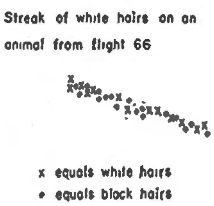
no graying typical of laboratory and equatorial-balloon flight controls. The figure on the right shows a mouse that has been exposed for 24 hours above 90,000 feet on a balloon flight at northern latitudes. This is a typical effect from such a flight. In addition, ground control mice were also flown at Holloman Air Force Base, New Mexico, and then examined under the direction of Dr. Chase. Because of the magnetic shielding effect of lower energy particles at this near-equatorial latitude, those particles which could terminate by thindowns producing the uniquely high rate of ionization are eliminated. Only higher energy particles which terminated by collision producing stars penetrate the Earth's magnetic barrier there. These control flights were conducted under similar conditions of temperature, pressure and duration, with the only exception that the animals were shielded from the low energy primary cosmic rays.

A previously unheard of phenomenon observed repeatedly but only in mice exposed at high altitude in northern regions is illustrated in Figs. 6 and 7. Streaks of gray hairs have been produced in the rodent hair coats where a heavy primary apparently grazed the skin of the animal and produced damage to a series of hair follicles.

Each hair follicle contains a group of sensitive pigment cells producing color at its root. Streaks of graying which have been observed by Dr. Chase and produced by heavy primaries measure up to 200 microns wide. The average distance between hair follicles in black mice is approximately 120 microns. The minimum radial spread that could involve a series of hair follicles to this width must correspond to a radiation intensity of 100 r.e.p. at a distance of 30 to 100 microns. At the 100-r.e.p. radiation intensity level the theoretical value of radial spread calculated by Dr. Schaefer is less than $2\frac{1}{2}$ microns, illustrated in Fig. 4. One is impressed by the marked discrepancy between theoretical values and those observed. To my knowledge there is no adequate explanation of this discrepancy.

Courtesy of Dr. Herman B. Chase, Brown University

FIG. 6 Gray streak observed in mouse flown above 90,000 feet. The streak runs parallel to the animal's backbone over its left flank.



Courtesy of Dr. Herman B. Chase Brown University

FIG. 7. Schematic of the gray streak illustrated in Fig. 6. The average distance between each hair follicle is approximately 120 microns.

Dr. Wilson Stone of the University of Texas has used another approach studying genetic effects of primary cosmic radiation. Neurospora of a mutant type that cannot produce adenine when placed on an andenine deficient medium, die in a relatively short period. He plated a group of this mutant variety on minimal (adenine-free) media as spores and exposed them to primary cosmic ray particles. When the heavy primaries irradiated the Neurospora they produced a back mutation in some of the exposed spores, producing the original wild type which could then produce its own adenine and survive on the minimal media. In this way one can pick out from many, many millions of spores just those which have been affected by primary cosmic radiation and have suffered genetic reversion to the original wild type. Preliminary results of his experiment are illustrated in Fig. 8. This indicates

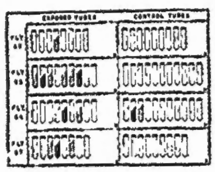


FIG. 8. Results of preliminary Neurospora experiments conducted by Dr. Wilson Stone of the University of Texas. Tubes with shaded area showed back mutations; tubes with black a a were strongly positive for back mutations. (Data for chart, courtesy of Dr. Stone.)

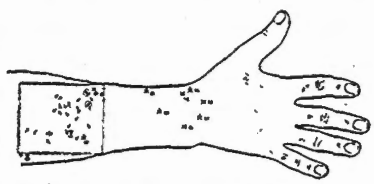


FIG. 9. Position of track plate and gray hairs on the right hand of the MANHIGH II Flight subject. The numbered X's indicate gray hairs. The circles with tails and crossed tails represent penetration of primary cosmic rays of charge corresponding to oxygen and greater. (Track plate data, courtesy of Dr. Herman Yagoda, AFCRC.)

that heavy primaries have a potent effect on genetic tissue but it requires further investigation to establish the relative significance compared to other forms of radiation.

In the MANHIGH experiment nuclear emulsion monitoring plates were placed on the forearms and chest of the subject to record the traversal of the heavy cosmic ray particles. The analyses of these plates by Dr. Herman Yagoda of AFCRC revealed that numerous heavy primaries penetrated these plates and therefore the arms of the subject. The position and appearance of medium and heavy primary nuclei on the plates are illustrated for the right arm in Fig. 9. The subject spent 16 hours above 90,000 feet, which is the minimum altitude for one to expect primaries of a weight greater than about nitrogen to be recorded. The atmosphere above 100,000 feet absorbs two thirds of the heavy primaries so that the exposure at this altitude is approximately one third that expected beyond the atmosphere at low orbital altitudes.[2]

Figure 9 illustrates the gray hairs observed on my right arm correlated with the medium and heavy thindowns of elements of a weight greater than oxygen recorded in the nuclear track plate. Seven gray hairs have been observed by this time in the wrist area and three in the monitored region under the track plate. It is likely that two of the three hairs observed in the monitored area have turned gray spontane-

[2] H. Yagoda, "Frequency of Thindown Hits by Heavy Primary Nuclei in Emulsion and Tissue," *J. Aviation Med.*, Vol. 27, pp. 522–532 (1956).

ously. There is a medium weight primary recorded in the general direction of number 9, and number 10 was incompletely monitored since it occurred along the edge of the track plate. It is not clearly established that any of the twenty-three medium heavy primary cosmic particles recorded penetrating the plate happened to strike a hair follicle. The skin of man has a much lower hair follicle density than that of mice, so this would not be surprising.

The greatly increased radial spread observed in the hair graying effect compared to that predicted by theory clearly indicates that heavy cosmic ray primaries produce unique ionization effects which need further investigation. Preliminary results with genetic materials and examination of monitored brains by Dr. Webb Haymaker at the Armed Forces Institute of Pathology raise very serious questions as to the sensitivity of these tissues to the unique radiation pattern of heavy primary thindowns. It is not surprising that heavy primaries may have missed hair follicles on the manned exposure study considering the relatively short duration of 16 hours above 90,000 feet and the considerable blanket of air absorber above the capsule at that altitude. It is important that the fundamental nature of the radiobiological mechanism producing the biological damage observed from heavy primaries be examined since previous experiments have shown that the effects observed are quite sensitive to the conditions of exposure such as oxygen concentration. The intensity of low energy heavy primaries in outer space is still an unresolved question.

WEIGHTLESSNESS

Another area of medical interest in space is weightlessness. A number of experiments have been conducted with animals in rockets starting in 1948 by placing monkeys in V-2's and finally in Aerobees. In these experiments physiological response in terms of blood pressure and respiratory rate was measured. Also, photographic records of the reaction of mice were obtained under various conditions in weightlessness.

As illustrated in Fig. 10, one mouse was permitted to run freely without any foothold that it could hang on to. The other mouse had a foothold on its paddle so it could ride around on this drum during weightlessness with a tactile reference. The first mouse, without the tactile reference, was quite disturbed particularly during the 15 seconds when there was no rotation of the nose cone exposing it to very nearly true weightlessness. This does suggest that there may be problems in terms of being completely unrestrained in the weightless state. This short exposure tells us nothing about any cumulative effects of exposure lasting for hours. These experiments show no significant impairment or disturbance of the circulatory system under weightlessness, although there was a slight decrease in the pulse rate which was no more than might occur when going from the resting to the active conditions.

Currently available aircraft such as F94's can produce near weightlessness for about 20 seconds disturbed by bumps produced from aerodynamic forces and turbulence. True weightlessness can be achieved during a rocket's ballistic trajectory which, of course, requires a lot of velocity as time is extended. Several minutes of weightlessness are theoretically obtainable using modern high performance rockets, but we can't expect to be sure of the true significance of weightlessness until man has achieved at least one daily cycle of approximately 24 hours in orbit.

There are a number of interesting little sidelights which have been discovered using aircraft. One is the result when trying to drink water from a cup during weightlessness. As you tilt the cup, the water doesn't pour out, of course, so you have to throw the water at your

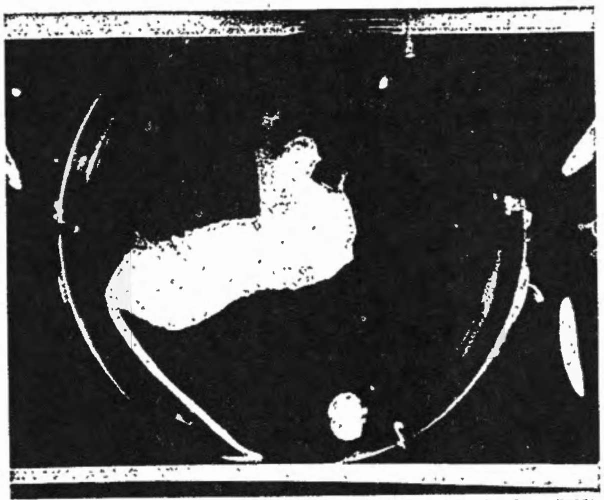


FIG. 10. Mice in the weightless state floating in an Aerobee rocket launched in April 1951.

face. Naturally some of the water gets into your windpipe, producing choking and difficult breathing which can lead to near panic.

The best solution found (this was first worked out at the School of Aviation Medicine) is to use a plastic squeeze bottle. In this way you can squirt the water out and by gradually moving the bottle away, you can keep the water up against the end so that it will continue to flow evenly.

Eating doesn't seem to be any problem unless you attempt to eat crunchy food that produces dust in your mouth. This is really no problem if you have the patience to keep chewing it until you get a nice damp bolus that will remain coherent as you swallow it. If you are in a hurry and try to swallow the dust, you quickly find dust floating in your pharynx and on down into your lungs instead of into your stomach. This is most distressing.

The body can be expected to react to prolonged weightlessness in one of two ways. First, it can acclimate much as the body does to low atmospheric pressure or low oxygen tension on top of high mountains. Over a period of time people become well adjusted to high altitudes of 18,000 and 20,000 feet without supplementary oxygen. Many of us living here at ground level if suddenly taken up to 18,000 to 20,000 feet would lose consciousness. Over a period of time the body may acclimate in a similar manner to weightlessness.

On the other hand, the body may react in the manner characteristic of sea sickness or motion sickness. Here it takes 10 to 20 minutes for symptoms to develop during an acute attack. In fact, if the stimulus is continued the situation tends to get worse with time. If this type of reaction occurred to weightlessness, it could become critically serious.

One hint as to what to expect concerns what happens when sensory input is reduced to the body. Sensory stimulation, particularly the ever-present touch sensations, can be minimized by placing a person in a swimming pool that has the same temperature as the body (as done by Dr. Lilly at the National Institutes of Health). Another approach has been to provide a monotonous tone and to place goggles over a person's eyes so that he has no organized auditory and visual sensory input from the outside world (done by Dr. Hebbs at McGill University). These and other experimenters have found that this type of situation eventually seriously disturbs the individual's equilibrium because he is not getting information with which to react. Persons in this situation characteristically generate their own sensations by experiencing illusions and hallucinations. This may occur within just a few hours.

The situation in weightlessness will not be this severe, but undoubtedly tactile sensations will be greatly reduced and visual and auditory sensations will be of a monotonous nature. The first 24 hours in orbit should be very revealing with respect to many aspects of weightlessness.

SEALED CABIN ENVIRONMENT AND EFFECTS OF ISOLATION

There are two more problems that are conveniently discussed together: the sealed cabin and the effects of isolation. I shall refer to the MANHIGH balloon flight in these areas since it provided considerable practical experience with them.[3]

The MANHIGH capsule operated as a truly sealed cabin, venting nothing of its atmosphere to the outside and receiving nothing from outside. Whatever inert gas was in the capsule initially remained there unless it leaked out. The oxygen was supplied by the 5-liter liquid oxygen converter and was metabolized to carbon dioxide and water

[3] "The MANHIGH II Balloon Flight," Holloman AFB Technical Report, in publication.

vapor. Carbon dioxide and water vapor were then removed by an outboard chemical absorption system. As the absorption unit removed these byproducts, total pressure dropped slightly, tripping a valve that released more oxygen to maintain a constant atmospheric pressure of 6 psi. as compared to the normal sea level pressure of 15 psi.

There were additional equipment and instrumentation: the emergency battery supply, emergency oxygen supply, the controls for the atmosphere in the capsule, electrical controls for the balloon flight operations, a panel for flight control instruments, and a camera to record the information on the panel.

The selection of the atmosphere in such a capsule presents interesting problems.[4] An adequate oxygen partial pressure of between 100 and 150 mm Hg had to be maintained. This was done by enriching the atmosphere to approximately 60 per cent oxygen even though the total pressure was only 6 psi. The remaining 40 per cent of inert gas required was divided half and half between nitrogen and helium. This was done on the basis that with regard to producing bends, both have undesirable characteristics, but for somewhat different reasons. From the information available, there may be a slight advantage in dividing the obnoxious qualities of each. There appears to be no disadvantage, and it is a relatively simple manner to provide this atmosphere.

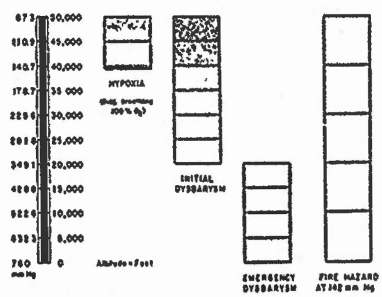


FIG. 11. Indication of relative hazards involved in sealed cabin atmosphere in terms of the pressure selected. The intensity of the shading indicates the degree of the hazard.

Figure 11 summarizes the problems that needed to be considered: First hypoxia, then the problem of bends or dysbarism, both during ascent and in the event of emergency decompression while floating at altitude, and finally the question of fire hazard. It illustrates clearly the advantage of selecting an intermediate cabin pressure, corresponding to approximately 20,000 feet or 350 mm Hg pressure.

[4] D. G. Simons and E. R. Archibald, "Selection of a Sealed Cabin Atmosphere," *J. Aviation Med.*, Vol. 29, pp. 350-357 (1958).

One of the problems of serious concern in space is that of temperature control. Figure 12 illustrates the surface temperatures and internal capsule temperatures of a 24-hour animal balloon flight in the 90,000 to 100,000-foot region. During ascent through the cold tropopause region, the surface became cold. Upon reaching ceiling during the day-

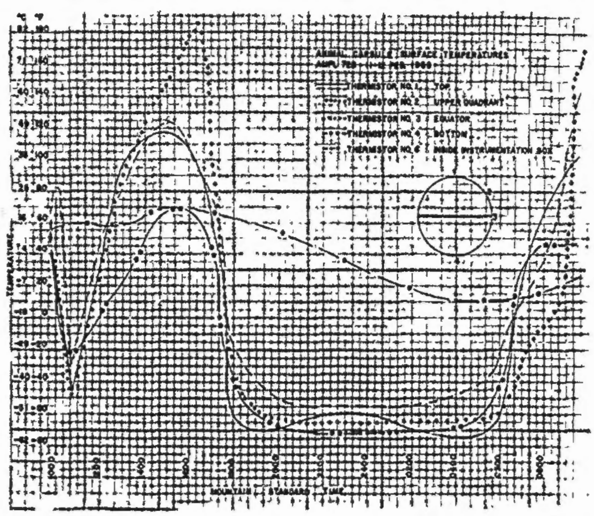


FIG. 12. Temperatures observed on the surface and in the center of a 30-inch diameter animal capsule floating above 90,000 feet on a balloon flight.

time, the top surfaces became hot, reaching a maximum of 182° F. At night all surface temperatures dropped to approximately −80° F. Recent experience has shown that painting the surface flat white reduces the heat absorbed during the day without significantly reducing its insulation qualities during the night.

On the MANHIGH II flight I wore a modified MC-3 pressure suit solely as an emergency garment. This was not a primary protection, but rather a secondary one, in the event of failure of capsule pressurization which fortunately did not occur. The combination of this suit which had poor ventilation and temperatures which sometimes reached 80° F. produced serious discomfort resulting in marked reduction of efficiency during the flight. In addition, the suit fit so snugly that it was physically uncomfortable much of the time. An important question is the relative importance of emergency protection of this type *versus* the loss of efficiency in discharging primary responsibilities. Considerable research must be done in this area.

The following is a brief review of the flight from the point of view of isolation in a sealed cabin. Figure 13 shows the flight preparation phase. The flight was launched at 0922 in the morning, went to altitude

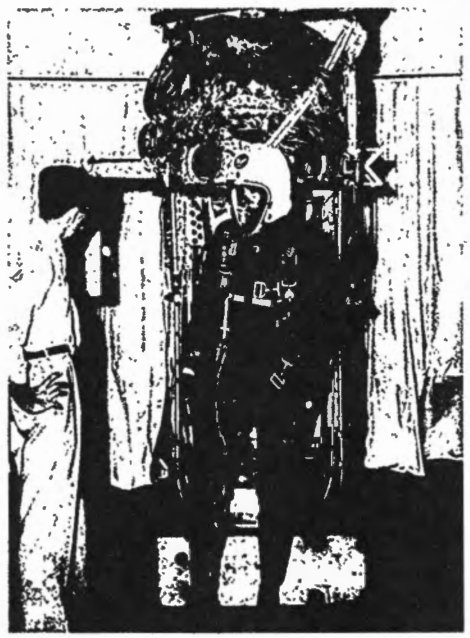


FIG. 13. Pilot of the MANHIGH II Flight of 19-20 August 1957 standing in front of the capsule preparatory to flight.

at about 1000 feet per minute, and then floated level at ceiling throughout most of the day. Close to sunset when the helium in the balloon cooled off due to the reduction of ambient radiation, the balloon descended to approximately 70,000 feet where it leveled off toward sunrise. It was necessary to drop approximately 200 pounds of ballast through the night to maintain this altitude.

The flight had been expected to float at 80,000 to 90,000 feet during the night, but the storm clouds below cut off the long-wave infrared radiation from the Earth, allowing the balloon to become much colder than it would have otherwise. In the morning after sunrise, the gas expanded as it was warmed by the sun. This, plus dropping a little more ballast, allowed the balloon to return to 95,000 feet.

By this time the weather situation had cleared sufficiently to permit descent. I released helium by opening an electric valve at the apex of the balloon. Slowly but surely it finally started to descend for landing.

The path of the balloon throughout the flight is quite interesting. On the Earth's surface the winds were blowing from west to east, as they generally do. Above 80,000 feet they were blowing from east to west as they normally do through the summertime at this latitude.

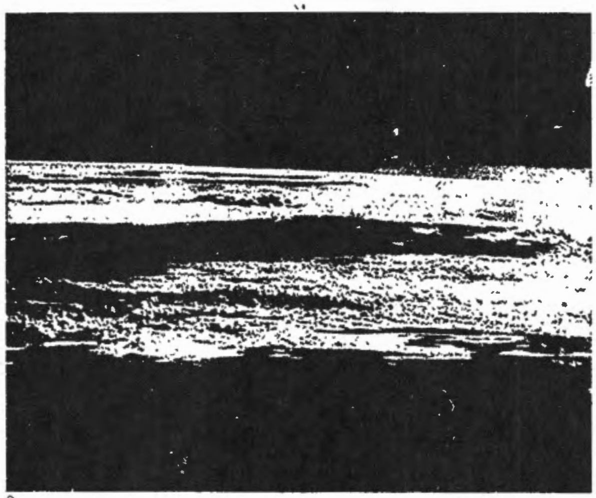


FIG. 14. Cloud systems and horizons as seen from the MANHIGH II Flight capsule on the afternoon of 19 August 1957. The curvature of the horizon is somewhat accentuated by the short focal length lens used to take this infrared picture.

Toward sunset the balloon dropped below 80,000 feet, so it encountered winds going to the south, later on during the night, it encountered winds going to the north. Then in the morning, as it ascended, it again encountered winds going west. Finally, on descent, westerly winds brought it back toward the east again. On this flight, at least, by selecting a particular altitude, it was possible to go in any direction—east, west, north or south. By selecting the right altitude, then, it is sometimes possible in balloons to go in whatever direction you wish.

The environment that one experiences in this situation realistically simulates conditions in space. There is a very real threat to existence in the event of equipment malfunction, so that one must be constantly alert to make sure that the life support mechanisms are functioning effectively. One just doesn't dare relax too much. In addition, there is the difference in physical appearance of the environment. Figure 14 is a view of the sky taken from 100,000 feet. In the foreground far, far below is the patchwork pattern of the western Minnesota farmlands. The clouds of the cold front system which the capsule crossed during the night stand out clearly in the background.

A comparison of the color of the sky 30° above the horizon with reference color charts held in direct sunlight (*Munsell* color charts kindly provided by Dr. Stakutis of AFCRC) showed the hue of 5.0B (blue) was too green and the other chart 5.0P (purple) was too red. Apparently a purple-blue hue would have been much closer. On both charts, value/chroma displays of 3/6 matched best. In the Munsell system the sky could then be approximately described as dark purple-blue of 5.0PB 3/6. Above the white band which measured approximately 1½° wide lying immediately above the horizon was a band of blue similar to the blue sky which one can see from the ground. Except at altitude, the range of blues usually seen looking from just above the horizon to the zenith is compressed to a band only 2° or 3° wide. This narrow blue-hued band faded into the narrow white band next to the horizon on its lower side and shaded quickly and evenly into dark blue-purple on its upper side. This sky color gave the impression of a spectral violet such as appears very close to the end of the spectrum. It was not an intensely lighted color because I had to observe it for a period of time to permit my eyes to become accustomed to its relatively dim intensity to see it at all.

The colors of the sky were concentrated in bands close to the horizon, and, when looking toward the horizon, my view generally included both white clouds below and dark sky above. The marked brilliance of the clouds below made the sky above seem black, or more accurately, completely absent. This gave me the impression that I was suspended above a large soup bowl with slightly inverted, upturned edges and without a top. Observing the faintly curved horizon, with very little imagination I felt as though I were on the very fringes of the atmosphere, closer to space than to Earth.

Generally my attention was concentrated outside the gondola in terms of what was there to be observed. The regular reports to the command post tracking van every half hour constituted an annoying interruption. The capsule seemed like a welcome window providing a fabulous view and precious opportunities rather than a prison and inclosure.

During the night as fatigue took its toll and external observations became prohibitively difficult because of frosting on the windows, problems and conditions within the capsule assumed increasing importance. The subject matter of the tape recorder clearly reflects this change in point of view. It was during this period that the capsule seemed cramped, yet a snug haven. It was also interesting to note that rather than being a source of annoyance, the half hour reports were a welcome contact with friends on Earth below.

After a very active period of observation during sunrise, a relaxed and leisurely breakfast seemed in order. It was during this period that the darkening of the sky and narrowing of the blue band above the horizon heralded a welcome return to the high altitude of the day before. While reflecting on his situation during breakfast, the pilot felt an dentification with space above rather than with the Earth below. It is also interesting to note that the bank of storm clouds below tended to establish a feeling of contact with Earth rather than separation from it. From this it appears that the tendency for "break-off" is reduced by a high degree of activity-saturation and enhanced by contemplative introspection.

Newest of the psychological factors of manned space travel and least explored is the area of astrochromatics: the study of the effects of environmental situations in terms of color, comfort, and the arrangement of living space. On the MANHIGH flight, a modest effort was made to break the monotony of the capsule interior by painting the walls a light blue, the ceiling flat white for maximum reflectance, and using instrument panels decorated with brightly colored knobs.

In this area, the nylon webbing seat designed by Lt. John Duddy of the Aero Medical Laboratory, Wright Air Development Center, represented a major achievement. Its nylon mesh stretched across a tubular, aluminum framework providing a very light-weight, strong, adjustable support which could be readily form-fitted to the individual using it. It proved extremely satisfactory throughout the 32-hour MANHIGH II flight.

In summary, much helpful information concerning man's reactions to flight under space equivalent conditions is becoming available. Man can certainly live under space equivalent conditions at 100,000 feet, and there is every indication that he should survive the physical rigors of 24 hours in orbit if properly protected. The critical question concerns his ability to observe new phenomena and think creatively in this situation, both to contribute to his own survival and to bring back new scientific knowledge. Man is uniquely blessed with a creative imagination, the keystone to progress in research. Machines and black boxes can provide facts to fill in recognized gaps of knowledge. Only man can recognize new problems and devise means of studying and measuring them. To fulfill his destiny of progress and discovery, man himself *must* broach the vastness of space.

SATELLITES AND TRAVEL IN THE FUTURE

BY

I. M. LEVITT[1]

When I speak of space travel in the futur: I am dealing in the realm of speculation, and I would like to say this about speculation. If any of these things should materialize as I say they will, nobody in the world will be more surprised than I. These are projections in the future and, as such, we can't afford the luxury of dogmatism. We have to strike out afresh.

We are certain that with the future will come a new technology and there probably will come a more comprehensive understanding of the fundamentals of the physics of nature, physical processes, the nuclear sciences and even of gravity.

Five years ago, if someone had discussed the possibility of harnessing the gravitational field or using it, the astronomers would have looked around for the man in a white coat to take him away. Today, five or six universities in this country and one abroad are undertaking research on gravity, in an effort to understand the nature of this force. Before we can use or do anything with gravity, it is necessary to comprehend fully its position in the scheme of nature.

CHEMICAL PROPELLANTS

First, I would like to speak about propulsion systems and fuels of the future. Today we have four satellites in the sky. For the placing of those four satellites in the sky, we have paid an enormous penalty in fuel and structure for the weight of payload. The Vanguard satellite weighs 21½ pounds. The all-up weight of the Vanguard vehicle is 22,000 pounds. Thus it means we use about one thousand pounds of fuel and hardware for each pound of payload. The Jupiter-C used 2200 pounds of fuel and hardware for a pound of payload. Because of the tremendous size of the Sputniks, American engineers believe this ratio to have been between 400 and 800 for the Russian rocket systems.

From these figures, it is apparent that chemically fueled rockets are frightfully expensive and terribly big in terms of the final result—and while the efficiency of these larger rockets will eventually become higher, there is not too much hope for improvement. I think it was Kurt Stehling who said that the optimum ratio you can expect for our present day rockets is about 400 to 1, that is, 400 pounds of fuel and hardware per pound of payload.

[1] Director of the Fels Planetarium of The Franklin Institute, Philadelphia, Pa.

If we expect to put really large payloads in the sky, the all-up weight of the chemically fueled rocket systems has to be tremendous and this country can't afford it. A full-scale space travel program with trips to the moon and probes to the planets may be undertaken, but if it is undertaken it will probably have to be on an international scale because it is difficult to foresee any single country footing the bill for a program of this type.

NUCLEAR POWER

While space travel may be realized through chemical propulsion, the salvation of regular and continued space travel on a grand scale will have to come through a solution to the problem of obtaining the necessary energy and power in ways other than by chemical fuels.

The most obvious way of getting more power is to use the nucleus of the atom, and at this moment there are several contracts in force to study the problem. The Rocketdyne branch of the North American Aviation Corporation has a contract with the AEC to try a feasibility study of a nuclear rocket. The AEC has also let another contract (for $10,500,000) called "Project Rover," a pure nuclear rocket system. North American Aviation is the contractee for this project.

There is a third proposal called "Project Snap," to be let by the AEC This will be tied to the "Pied Piper" project, which deals with a television camera circling the Earth and relaying pictures back to the Earth. We apparently need a power source for that and we also need power to get it up, so this will become part of Project Snap. It may very well be that Lockheed or Convair or both will have a part of this contract. From this, you see that there is considerable activity today, on the American scene at least, in the application of nuclear power to space travel.

I recall a long talk I had about four or five years ago with one of the most prominent men in the space travel picture, and I spoke about the impossibility of using nuclear energy as a power source. After five or ten minutes, this man looked at me and said "How do you know that this cannot happen in five or ten years? It is impossible now, but it may be possible in perhaps five or ten years."

Only today do I realize how intelligent an answer that was. I had made the statement that in order to use nuclear power efficiently you would have to use temperatures on the order of 10,000 degrees; and inasmuch as we have nothing on the Earth which will remain a solid at 10,000 degrees, obviously a motor couldn't be built to use nuclear power.

Apparently I was dead wrong on that. I was also dead wrong on the possibility of radioactive poisoning. I thought that if you had a nuclear power plant, some of the exhaust materials would be fission products and therefore there would be vicious radioactivity in the immediate vicinity of the launching point. Again he said, "How do you know?" I didn't at the time, and now I see that I had worried too much about it, because apparently radioactivity will be a minor factor and tremendous power can be produced using much lower temperatures.

Therefore, we will probably go to nuclear power in the near future, for purposes of getting off the Earth.

Fusion

In the nuclear field there are two approaches. You can, for instance, visualize a rocket using fusion for power. Figure 1 shows what a fusion rocket would be like. It would include a tank of heavy water, or even liquid

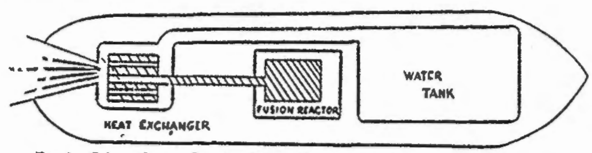


FIG. 1. Schematic drawing showing application of fusion power to space travel

hydrogen, and some mechanism for bringing the hydrogen atoms together to fuse them, thus creating a tremendous amount of energy. Water would be heated and vaporized under tremendous pressures in the pressure chamber, and finally would be exhausted through a regular nozzle. Or, it is also possible to use the energy to create electricity and ionize some of the alkali metals and use the ion stream in the exhaust.

This is a form of rocket which would use fusion power, but it is of academic interest only, for we have yet to prove that we have achieved true fusion on the Earth. The only evidence we have today is that in the so-called magnetic bottles some stray neutrons have been found which could have come from some other source. In spite of all we read, we still have to find definitive proof that fusion has occurred on the Earth. We must consider this strictly as an activity of the future.

In fusion, we use a working fluid. We exhaust at a very high temperature and a high speed, or we can create a high temperature to run a generator, create an electric current and discharge ions at tremendous speeds.

Apropos of this, you may have read quite recently about experiments performed at the General Electric Laboratory in which heat was converted directly into electricity. If this can be done, then you have an application for that type of device in which heat is converted into electricity, which in turn accelerates ions in an ion rocket.

But, as was indicated, so little is known about this that except for generalities we can't discuss this intelligently.

Fission

The other type of nuclear energy involves fission which today is a well understood process. There are two ways in which we can use fission. In one we create a high temperature to heat and exhaust a working fluid. The other is to generate electricity for use in an ion rocket.

Figure 2 shows a device using fission power in the first of these two ways. This utilizes a fission reactor with liquid hydrogen heated to high temperatures and high pressures. The hydrogen at a pressure of, say, 225 psi. is exhausted to zero pressure through a nozzle.

The other way of using fission is to create electricity using a fission reactor. The electricity will accelerate ions for an ion rocket.

These are the two ways in which we use fission. I am coming back to this later, incidentally, to give you a concrete example of this application

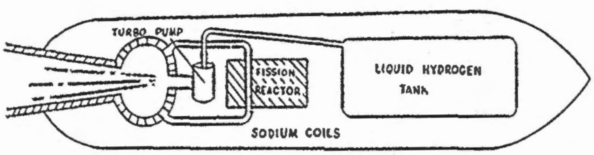


FIG. 2. Schematic drawing showing application of fission power to space travel

SOLAR ENERGY

Let us consider next what energies are available in space. In addition to nuclear energy already mentioned, there is solar energy and finally photonic energy. I will mention four different ways in which solar power can be used.

In the first way, a solar power plant uses sodium or mercury in a heating cycle. The power plant creates electricity which again can be used to accelerate ions for an ion drive. Figure 3 gives an example of this type of solar engine in which the receivers—the boiler—are mercury or sodium. The boiler drives the turbine which in turn drives a generator creating the current. The current goes to the cesium chamber, the cesium is ionized and the jets come out, giving thrust to move the rocket.

The second way is to heat a working fluid and exhaust it. The third way is to create electricity directly using a device similar to the Bell Telephone Laboratory silicon wafers. In this way, enough high voltage electricity for an ion drive may be created. Lastly, there is a thing called solar sailing. If that sounds strange to you, it sounded strange to me, too, just about a month ago, but it is no longer strange and I would like to discuss it later.

NUCLEAR VERSUS SOLAR ENERGY

To speak about these different energy systems intelligently, we have to find out just how to apply them. There are two ways in which they can be applied: one, to leave the Earth; and two, to travel in space.

To leave the Earth we need high thrust for short durations. The one thing we can't afford is to fight gravity for too long a time because gravity will always win. Therefore, satellite velocity or escape velocity must be achieved in the shortest possible time, which puts a high premium on high thrust.

Only the chemical and the nuclear fuels can fulfill this condition of high thrust, so immediately solar power is eliminated as a means of leaving the Earth. Chemicals are prohibitive in cost and, therefore, only nuclear fuels

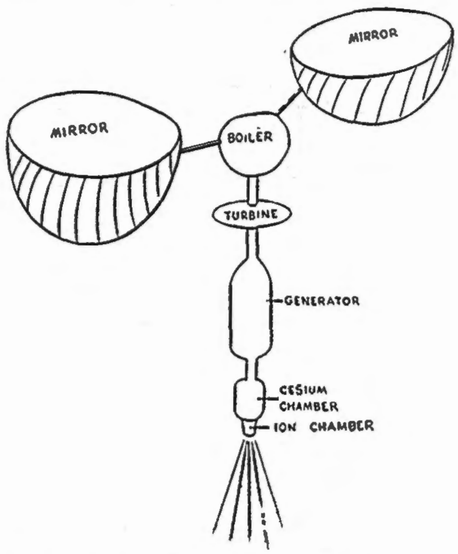


FIG. 3. Application of solar power to space travel using ion rocket.

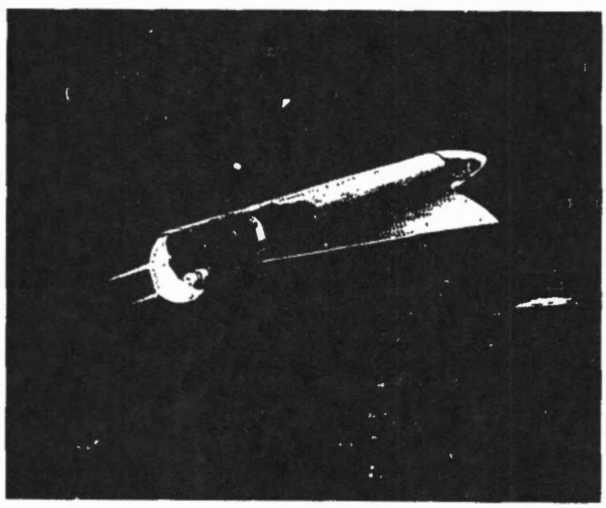
are left. Though nuclear rocket motors are unproven, untried, still I think this country and perhaps even Russia, is staking a great deal on the future of nuclear power—not just to put a 30-pound satellite or even a 3000-pound satellite in the sky, but for payloads on the order of many tons.

In traveling through space, there are three methods of obtaining energy: one is the chemical method again, which is prohibitively costly, so we immediately discard that; the second is the nuclear source; and the third is solar energy.

In the case of solar energy, certain reservations must be made, for, if you get far enough away from the sun the radiation of the sun will fall off (the intensity will decrease according to the inverse square law) and pretty soon you get far enough away from the sun so that its radiation ceases to be effective. For places close to the sun, let us say out to the orbit of Mars, it is possible to use solar energy to power the space ships.

Project Snooper

Now I would like to speak about one particular form of an ion rocket called "Project Snooper." I have looked through many papers on the subject of ion rockets, including those by Ernst Stuhlinger who established most of the fundamental ideas on ion propulsion. Of all, I think Project Snooper is the one that will catch your fancy and imagination.



Courtesy of Rocketdyne.

FIG. 4. Artist's concept of ion rocket in space. Nuclear power is used to get rocket off the Earth and then provide electrical energy for ion drive.

Figure 4 shows Project Snooper. It is the concept of two scientists, M. I. Wilinsky and E. C. Orr, who are with Rocketdyne. Project Snooper utilizes a double propulsion system. One system uses nuclear energy to get off the Earth into space. Once in space, the nuclear power plant is discarded as a propulsion device and is used to generate power for the ion rocket.

Let's discuss the details of Project Snooper. A fast or an intermediate nuclear reactor is built into this space ship, which, incidentally, is unmanned The ship has a total weight of about 3300 pounds, broken down in the following way: 1500 pounds of payload, which includes TV, radar, guidance systems, communications systems, etc.; 1000 pounds for the reactor; 220 pounds for the propellant (cesium is used); and 580 pounds, miscellaneous.

In this reactor, sodium is used as the coolant. The sodium is used to heat mercury which in turn drives a turbine. The turbine will drive a generator to generate enough current to create cesium ions and expel them. This gives 147 kilowatts of electrical energy. Most of this must be dissipated as heat and that is the reason for the wing-like surfaces. They are rolled up in a nice little package and are put into orbit. When in orbit they unroll. They are covered with soot to give them a high emissivity to get rid of unwanted heat.

In the propellant section cesium is used. As a matter of fact, all the alkalis have rather low ionization potentials. Cesium will ionize in a matter of microseconds by passing over hot platinum or hot tungsten. The cesium is stored in a liquid state at 100° F. It is forced through a check valve and sprayed on the walls of a vaporizing cylinder at a temperature of 1400°. This will create the ions which come shooting out of the two motors.

With 4 amperes of current, these two motors can generate a thrust of one-sixth of a pound each. This thrust will yield an acceleration of a tenth of a milli-g; in other words, a ten-thousandth of the force of gravity.

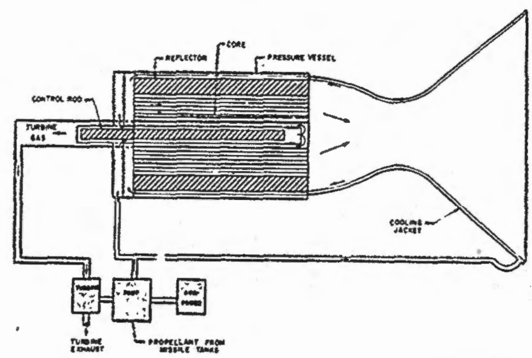
To prevent a charge from building up on the vehicle, electrons are also ejected, so the device is electrically neutral.

This is one way of using a nuclear ion rocket. This is a vehicle configuration which has been very well worked out and may eventually reach the hardware stage.

PURE NUCLEAR ROCKETS

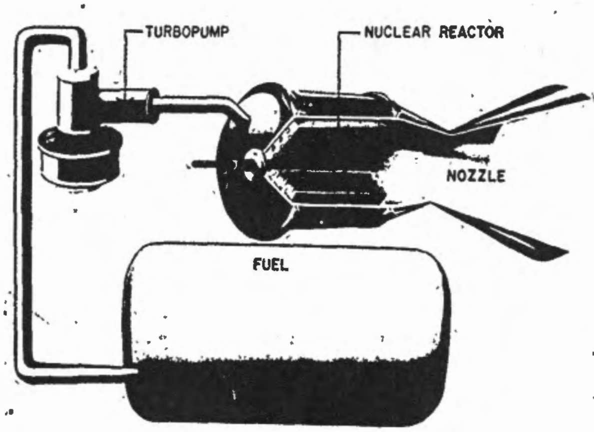
Now let us go to pure nuclear rockets without worrying about an ion propulsion, and for that I would like to give you a little bit of background. If we use the V-2 as an example, it has a specific impulse of about 220 seconds. To get a large enough rocket off the surface of the Earth with that type of impulse, 98.4 per cent of the rocket would have to be fuel, which is obviously impossible. That is why we have as many as three stages on our satellite rockets.

If the specific impulse were up from, say, 220 seconds to 400 seconds, then it would be possible to put a satellite in the sky with two stages. And if the specific impulse went up to 800 seconds, then the job could be done with a single rocket. Of course, that is the aim because in a single stage rocket a tremendous amount of weight is saved.



Courtesy of Rocketdyne.

FIG. 5. Schematic diagram of nuclear rocket.

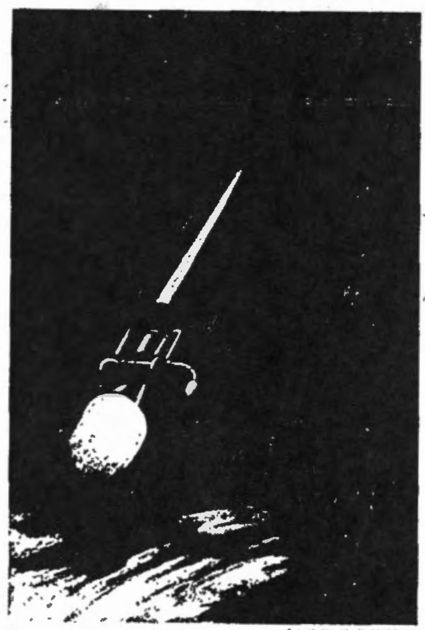


Courtesy of Rocketdyne.

FIG. 6. Artist's concept of a nuclear-powered reaction motor.

We know the specific impulse is proportional directly to the square root of the temperature and inversely to the square root of the molecular weight of the working fluid in the heated state in the chamber. This means the higher the temperature, the higher the specific impulse; and the lower the molecular weight, the higher the specific impulse.

We investigate materials for the working fluid and find that hydrogen is ideal. It has a molecular weight of 2. As a point of fact, it is the

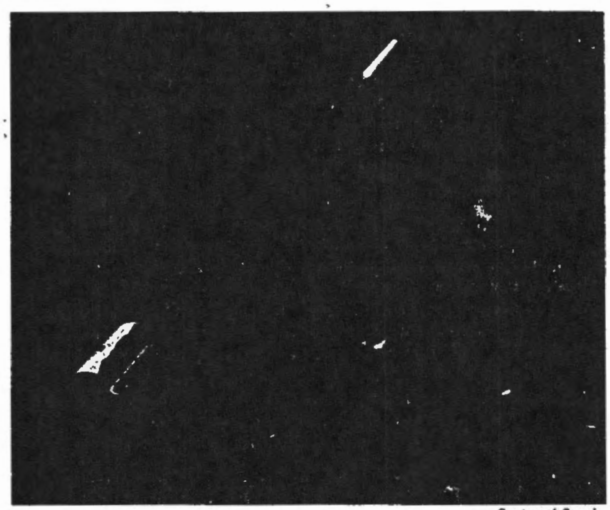


Courtesy of Rocketdyne.

FIG. 7. Artist's concept of a nuclear-powered rocket.

outstanding working fluid for nuclear rockets. Helium has twice the molecular weight of hydrogen, but logistically it is a most difficult material. The temperatures for liquid helium are very low (−268.9° C.). It is very difficult to make, very difficult to store and it isn't easily procured, curious as that may sound.

Then we go up to molecular nitrogen which has a mean molecular



Courtesy of Convair.

FIG. 8. Nuclear rocket in operation showing power plant in front with payload trailing by 2000 feet.*

weight of about 8.5. Finally we go to the compounds. Ammonia, for instance, even though it does have a much higher molecular weight than helium and, of course, is poor in performance, is still better than helium because of the shortcomings I have just mentioned.

The nuclear system will be or could be like a conventional chemical system in that there is a turbo-pump forcing the working fluid into a thrust chamber which contains a reactor. At the end of the reactor is a nozzle to expand the gases. The operation appears very simple.

Figure 5 is a schematic diagram of a nuclear rocket. It contains a propellant section. As the propellant flows in, it is divided into two portions—one flows in one direction and one in another direction as the cooling medium. These two portions finally come together and then the working fluid is forced down through plates. As it is forced down through the plates it picks up the heat from the plates. It comes out at a pressure of perhaps 250 psi. and expands to zero pressure, giving the necessary thrust. Figures 6 and 7 are artist's concepts of a nuclear-powered reaction motor and the rocket it propels.

* Figures 8, 9 and 10 are representations of concepts developed by Krafft Ehricke, of Convair.

Thus the operation of a nuclear rocket motor is not too different from the chemically fueled rockets. As to the elements of the reactor, there is the reactor core which is a matrix of graphite impregnated with U-235; there are reflectors which are also graphite; there is a control rod; and there is a structure system. All of these are units of this system.

Figures 8 and 9 show the nuclear rocket application to a moon trip.

Design of a Nuclear Rocket

Now I have a problem which I would like to discuss. This was worked out by the Rocketdyne people who have this study program. They said: "Suppose we wanted to build a nuclear motor which would give 100,000 pounds of thrust. How is it done?"

They limited themselves to a maximum temperature of 4000° F. The width of the plates and the gap between the plates were set at ⅛ of an inch. The hydrogen pressure in the reactor is 800 psi. and the temperature of the hydrogen entering the reactor is 134° K.

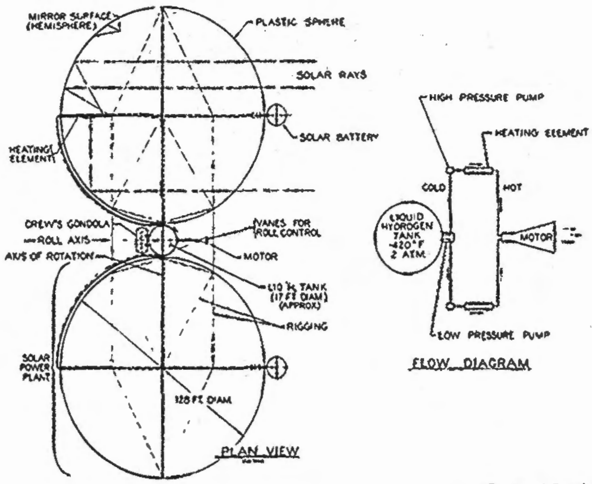
If the reactor raised the temperature of the hydrogen to 3600° F. (notice this is less than the 4000 I spoke about), the reactor would have to be 6 feet long. We could get a mass velocity of hydrogen passing through

Courtesy of Convair.

FIG. 9. Nuclear rocket setting down on the moon.

the gap at the rate of 40 psi., and a specific impulse of about 790 Dividing the 100,000 pounds of thrust by the 790 gives 127 pounds per second of hydrogen which must flow through. And if this is divided by 40 (pounds per second per square foot), it gives an area of 3.2 square feet which must be multiplied by 2 to take into account these plates. This shows that a reactor 6 feet long, with a cross-sectional area of about 6½ square feet will give 100,000 pounds of thrust to propel a rocket.

With that, I think we will close our discussion of nuclear reactors and go on to the solar power devices.



Courtesy of Convair.

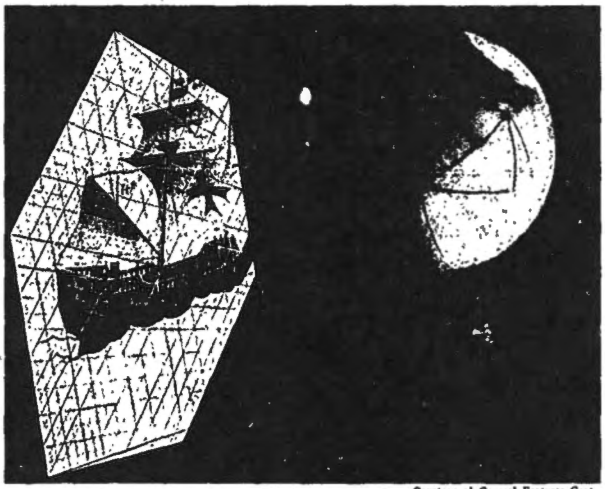
FIG. 10. Schematic drawing of solar powered space ship.

SOLAR POWER DEVICES

Figure 10 is Krafft Ehricke's concept of the solar space ship using a two-man crew. The ship itself consists of two giant shells 128 feet in diameter, made of a very thin plastic, coated on one side with aluminum to concentrate the radiation from the sun on a sort of cooker. This is a workable system which collects solar energy to operate on a working fluid, that is, heat it up, get it up to a high pressure and discharge it to zero pressure. It is the direct method of using solar energy.

It is also possible to use solar energy to create electricity, either by a generating plant or by direct conversion, for use in an ion drive.

Finally, there is this thing called solar sailing, shown in Fig. 11, which is about as unscientific a sketch as you will see. However, I don't know of anything I could have done which would have made this more apparent to you.



Courtesy of General Features Corp.

FIG. 11. Symbolic application of solar radiation to motion in space.

In the March 1958 issue of *Jet Propulsion* Dr. Richard L. Garwin, of Columbia University, said, in effect: In space we find radiation pressure. Astronomers have been familiar with radiation pressure for many years. They believe even today that that has a great deal to do with the formation of the stars. Dr. Garwin believes that radiation pressure can be used for solar sailing.

Suppose we have a satellite circling the Earth, and as that satellite is moving away from the sun we could put up a sail to intercept a certain amount of radiation. When the satellite moves around toward the sun, the sail is furled and radiation pressure doesn't affect you except on the surface of the space ship which has a comparatively small cross-sectional area.

It isn't too difficult to make a computation on this type of ship, because we know the solar constant. We get 2 calories per square centimeter per second on the surface of the Earth. We know the Earth's distance from the sun, so we can compute the size of the sail we need for, say, a unit mass of matter to have radiation pressure affect it.

Dr. Garwin did just that. He contemplates a 0.1-mil thick plastic aluminized on one side to double the effect. He computes the mass per unit area to be 5×10^{-4} g/cm^2. If a 21-pound Vanguard is circling the Earth and if a sail 70 meters in diameter were put on to that 21-pound Vanguard, that sail would accelerate the satellite at 5×10^{-2} cm/sec^2 which is approximately 1.6×10^{-4} times the force of gravity.

From this, Dr. Garwin shows that it is perfectly feasible to build a big sail, tie it to a space ship and use the radiant energy of the sun to move around in space. And if by chance our technology should improve so that instead of the sail having a thickness of a tenth of a mil we could cut that, say, by a factor of 5 or 6, then the acceleration will be increased still more, giving a significant push to the satellite.

This concept of sailing around in space and letting solar energy propel you is not nonsense. Apparently it has considerable merit and you may be hearing a great deal more of this in the future if man should get out into space. Of course, if you have this power all you have to do when you circle the Earth is to unfurl the sail when you are moving away from the sun to pick up the energy and then furl it when you are moving toward the sun. A small solar battery can be used for the energy required to furl and unfurl the sail, which can be done in a matter of 80 seconds. There will be no retardation because the sail is open on the side of the Earth when it is moving toward the sun. Thus we can move from satellite velocity to escape velocity given sufficient time.

PHOTON ROCKETS

Now I would like to speak for just a moment about the photon rockets. I think most of you know that a beam of light produces a thrust. On a very delicate micro-balance you can actually make the balance move by shining a beam of light on it. It is a rather delicate experiment, but it has been done.

It is conceivable that, given enough energy in these photon beams, they could be used to propel an object at a rather low acceleration. Essentially this means converting matter directly into radiation. At this time only in giant accelerators or in cosmic ray reactions do we see complete conversion of matter to energy. The mutual annihilation of the electron and positron is well known. Recently other evidence of the heavier particles entering into this type of a reaction has been noted. The 6 BEV Bevatron has given evidence of the annihilation of a proton and its negative counterpart, the antiproton. Thus the reactions for the complete conversion of mass to energy are being uncovered.

The energy from these reactions results in a stream of photons of various energy contents. If the stream could be directed, then we could use it to impart motion to the vehicle containing the energy generating process

In theory, an ordinary lamp could be used for this. There are two possibilities: (1) a lamp with a very high current going through the filament and a reflector in back of it so as to direct all the energy out in one direction; or (2) an electric discharge used to excite gas The glowing gas will emit photons to be directed in a given direction.

That is as much as I can tell you about the photon rocket, except to say that Professor Kurl Stanukovitch, a Russian scientist, has apparently been engaged in this work for the past four or five years The Russians may be pursuing this work more diligently than the West.

MAN IN SPACE

After we get these rockets with unlimited amounts of energy and nuclear reactors, what do we do then? The first thing I can visualize is that by the year 2000 a civilization will be established on the moon. The moon will then become a super-base for operations in space in general.

Astronomers believe that on the moon there are magnesium silicates. Some of these rocks contain 13 per cent of water by weight. For every 100 pounds of rock there are 13 pounds of water or crystallization locked up tightly in these rocks. With an unlimited amount of energy—and by the year 2000 I think the most plentiful and least costly thing (except for taxes) will be energy—I will assume that it is possible to break up the rocks, crush them and bake or electrolyze them to extract the water from them.

When that water is subjected to the ultra-violet radiation of the sun, it dissociates into oxygen and hydrogen. Thus the rocks on the moon will furnish an atmosphere and water. By the year 2000 our chemists will have advanced to the point where they will be able to synthesize most of the materials we have on the Earth. Thus there is no reason why a civilization cannot be started on the moon.

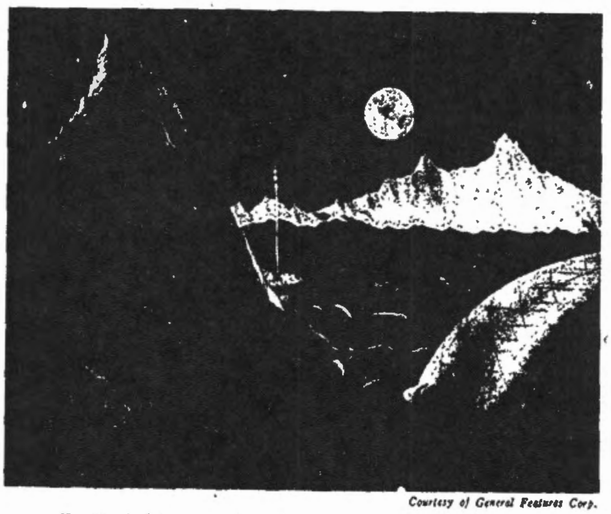
By a civilization I mean villages and towns, covered by plastic domes (see Fig. 12) which will trap an atmosphere at perhaps 6 psi. and filter out the lethal ultraviolet radiation. People will probably be sitting in auditoria on the moon (not you, but perhaps your children or grandchildren) the way you are seated here—without space suits, without fantastic helmets and radios to permit communications, etc.

Once we have conquered the moon, we can begin moving through space.

One of the most popular topics in the Planetarium is the demonstration on the possibility of life on other worlds. In this story of life on other worlds, astronomers find evidence that many planetary systems abound in space. We find stars such as 61 Cygni and 70 Ophiuchi, which do not move uniformly. These stars apparently waver in space with a sinusoidal motion. The only object that can make those stars move in that fashion is a dark body or planet near it, and from the amplitude of that motion the mass of this dark body can be determined in terms of the mass of the visible star.

If there are large dark bodies or planets surrounding the stars, certainly there must be small planets. Statistically we think that there are perhaps a billion planetary systems in our own galaxy—in our Milky Way system. In the observable universe there are perhaps a billion billion planetary systems.

With the tremendous number of planetary systems it is inconceivable that many of them won't be the size of the Earth, the same distance from their sun as we are from our sun, and because the elements are scattered through space rather uniformly (the same elements in the same proportions), life could very well have begun there the way it began here some four or five billion years ago.



Courtesy of General Features Corp.

FIG. 12. Artist's concept of a settlement on the moon after the year 2000.

There may be life out there and that is one of the things we would like to know. In the future I could visualize a giant celestial Noah's Ark leaving the moon, weighing perhaps a million tons and fully compartmented. In this celestial Noah's Ark there would be a complete civilization—schools, libraries, universities, factories, nurseries, hospitals, places to grow plants and food and to raise animals. Agriculture would be practiced, along with rigid birth control.

This civilization will start out from the Earth at some date in the future and it may travel for 700 years with a speed of, let us say, 6000 miles per second. At the end of 350 years that space ship may have gone to one of the nearest stars such as 61 Cygni or 70 Ophiuchi to see whether there is a planetary system around that star. And if a planetary system is there, is there life on one of those planets, perhaps similar to ours or perhaps vastly superior to ours?

PROBLEMS OF SPACE TRAVEL

I have blithely spoken about speeds on the order of 6000 miles per second and yet there may be particles in space which will preclude our getting to these stars even at this speed.

If, in traveling through space, we should hit anything, it makes no difference who possesses the motion—the fact that there is a certain velocity differential between the two bodies can create a hazard of extreme magnitude.

We know many particles abound in space in the vicinity of the sun. At night, if you watch the sky assiduously you may see three or four streaks of light cutting rapidly across the sky. Astronomers call these things meteors —you call them "shooting stars." Most of these particles are small, usually no larger than a grain of sand, but they enter the Earth's atmosphere with speeds up to 45 miles a second. It is this speed which gives them their great energy. A particle the size of a grain of sand moving with a speed of 25 miles per second, which is parabolic velocity at the Earth's distance from the sun, has about as much energy as a high power rifle bullet. A larger particle has a greater amount of energy in proportion to its mass. Thus in the vicinity of the Earth we find tiny particles which are potential hazards to any extended space trips in which we will be moving at high velocities.

Some have suggested that although this is true in the vincinity of the Earth, these particles thin out or are non-existent when we get out beyond the solar system.

Dr. Fred L. Whipple, Director of the Smithsonian Astrophysical Observatory, has advanced evidence that the space between the stars is the deep freeze which houses the comets that move around the sun. As comets are composed of these tiny particles in a matrix of hydrogen ices, we must accept the fact that out beyond the solar system we should find similar particles. These particles present a severe problem.

How will these particles affect our celestial Noah's Ark?

In order to view this in its proper perspective, let's do some arithmetic. Let's begin with a particle of matter about the size of a grain of sugar weighing about 0.004 grams.

The formula for the kinetic energy of a body is:

$$K.E. = 1/2mv^2.$$

For a speed let us choose 10 miles a second which is a reasonable speed in traveling between the planets. Let us convert miles into feet and the grams into pounds to get our answer in foot-pounds.

Putting these figures into the above equation we get a K.E. of 11,000 foot-pounds. Remember, this is a tiny grain of matter moving at 10 miles a second.

However, we were concerned a moment ago with speeding our space ship along at 6000 miles per second. As this is greater by a factor of 600, the energy content (which goes up as the square of the speed) is increased by 360,000 times. Thus that tiny grain of matter now possesses an energy of about 4,000,000,000 foot-pounds! And even with this speed we will take about 700 years for a round trip to a star. Some day we might be able to accelerate to still greater speeds to perhaps consummate this trip in a lifetime. Let us imagine that it were possible to speed up to 62,000 miles a second—a third the velocity of light. A round trip under these conditions may be made in 70 years.

Now let us look at the energies involved in the tiny particles which we run into at this tremendous speed. We have speeded up by a factor of 10 so the energy generated is increased by a factor of 100. Thus that tiny grain now possesses an energy of 400,000,000,000 foot-pounds. This is an incredible amount of energy and I would like to translate that for you in terms which may be more readily understood.

A rifle bullet travels with a speed of about 3000 feet a second—slightly more than a half mile a second. Let us assume that the bullet weighs about an ounce. That bullet possesses an energy of about 300,000 foot-pounds. The ratio of energies in the grain of matter in space and the bullet is about 1.3×10^6. To get some idea as to what this means let us imagine a block of stone 17 feet on each edge. If the Celestial Noah's Ark collides with a tiny grain in space the effect on the Ark will be the equivalent of having a block of stone 17 feet on an edge strike the Ark with a speed of 3000 feet per second!

Under these conditions it is highly unlikely that any Celestial Noah's Ark ever contemplated could withstand an encounter of this kind. As it is highly probable these particles exist in space, encounters of this type must be expected, though we cannot deduce the frequency of hits.

From this it is apparent that it is highly unlikely that the fastest trips could be made with the fabricated Ark unless we discover how to utilize "force fields" to vaporize the impeding matter.

However, there is another way out of the dilemma.

ASTEROIDS AS SPACE SHIPS

In that part of space between Mars and Jupiter we find the asteroids. Astronomers believe that a long time ago there was another planet in the solar system which for some unknown reason was shattered.

Thus we find a belt in which are smaller bodies ranging in size from over 400 miles in diameter to less than a mile in diameter. One of these asteroids would make a perfect Celestial Noah's Ark.

We could get to that asteroid with huge space ships fabricated on the moon. Once on the asteroid we will assume that we still have unlimited sources of energy, so we could begin hollowing out the asteroid. This could then be compartmented and again a civilization could be taken aboard. Rocket motors using nuclear fuels could be installed on all faces of the irregular chunk of rock (see Fig. 13). Plastics would be synthesized to provide covers for small settlements on the asteroid and it may even be possible to find water-bearing rocks on the asteroid.

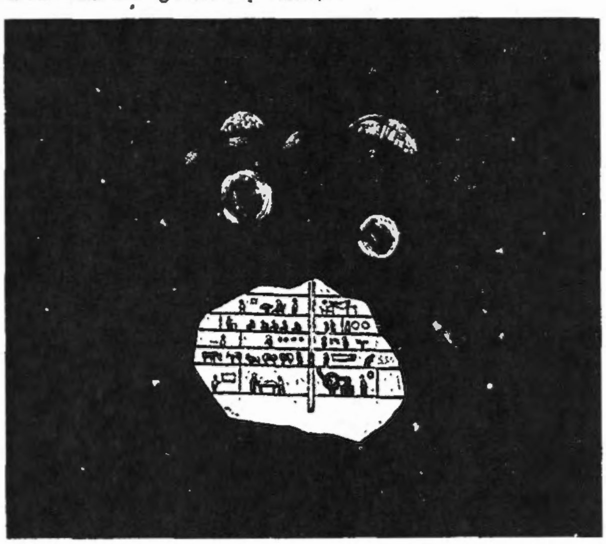


FIG. 13. Hollowed out asteroid for use as a celestial Noah's Ark to go to the nearest erratically moving stars.

After the asteroid has been hollowed out we will leave a thin skin perhaps several hundred feet thick. Now when these fast moving particles hit the surface of the asteroid they will not devastate it.

There are two ways in which we can get away from the sun's gravitational field and get to the distant stars. Both of the methods involve the use of a "gravitational whip."

One way which has been discussed for a long time has been the moving in toward the sun and letting the sun whip the Ark off into space in a hyperbolic path. The other method, the work of Krafft Ehricke, has been disclosed rather recently. He has shown that when a space ship approaches the planet Jupiter, the gravitational field of the planet is so strong as to attract the space ship to its surface. If, however, the approach is precisely right, then the gravitational field of the giant planet can act as a sling shot to project the Ark into a hyperbolic path to the distant stars. Thus we have the means whereby power need not be used continuously. All that is needed is the power to get the Ark to the right spot at the right time and the gravitational fields can do the rest.

In moving through space it is still possible that the plastic shells will be hit and they will be damaged but the odds are all against all of them being hit at the same time. Thus we will have opportunities of seeing where we are headed. The captured asteroid as a space ship has definit merit.

Using this type of an Ark we will be able to get away from the Earth to the other stars to answer one of the most profound questions we can pose: Is there life on other worlds?

TIME DILATION

The last question I want to discuss is time dilation.

More than 50 years ago Einstein enunciated his Special Theory of Relativity. This law has opened the floodgates of speculation and has led to dramatic and intriguing conjectures of the effects on a human being moving through space at extraordinarily high speeds.

One aspect of the theory relates to changes in mass, length and time with increase in speed. As a consequence of this theory, it can be shown mathematically that as we approach the speed of light—186,300 miles a second—the length of a body will shrink to zero, the mass of the body will become infinitely large and time will slow down and finally stop.

This slowdown or time dilation is normally imperceptible, because most motions on the Earth or in the solar system are so slow. Only at speeds of tens of thousands of miles per second can even the most sensitive of instruments be expected to show this effect in short time periods. As this will be a cumulative effect, in a significant time interval this slowdown may be apparent.

Any doubts as to the validity of these laws have been partially dissipated, for some facets have been verified in our laboratories. For example, when cyclotrons were used to accelerate particles to bombard the nuclei of atoms to transmute them into new isotopes or elements, the scientist found that when the accelerated particle reached a large fraction of the speed of light there was a substantial increase in mass. In fact, so pronounced was this increase that the cyclotrons had to compensate for this relativisitic increase in mass so that the electric "kick" could be synchronized with the particle. Only then could the cyclotron operate satisfactorily.

Laboratory proof showed that this law was valid and understood.

Experimental proof also has been furnished by the behavior of mumesons passing through the atmosphere of the Earth. Because of their extremely short life, comparatively few should survive and reach surface. The only explanation for the significant number reaching the surface is that the time-dilation effect has expanded their life spans.

For many years there has been general acceptance of the time-dilation part of the theory. Only recently has any opposition to these concepts been advanced. At this moment an acrimonious wrangle has been triggered among distinguished scientists of outstanding ability. The imminence of the proposed earth satellite has lifted the question from the purely academic realm and has brought it to realistic appraisal.

In the staid and proper *Nature*, a ranking magazine of science published in Great Britain, a series of provocative articles has appeared in which two articulate scientists and astronomers, Professors W. M. McCrae and Herbert Dingle, have debated the question of remaining young while traveling at high speeds. While both agree that with increased speed the clocks of the space traveler will slow down, the interpretation of the slowdown is the basis of the controversy.

Professor McCrae considers the heart as a form of clock. When the speed reaches a certain value there will be a perceptible slowing down of the heart so that the body metabolism will also slow down and thus we will live more slowly and longer.

Professor Dingle contends this interpretation is a violation of what he calls "common sense."

Why anyone should use a term like "common sense" in light of the phenomenal progress made in the field of electronics, atomic bombs and other nuclear weapons, is difficult to comprehend. Today, it must be admitted that common sense is the usual, but the unusual also has an insidious way of becoming reality.

Scientists like Professor Dingle, who do not believe that youth can be maintained with fast motions, argue that the only thing that counts in this theory is velocities. They point out that the relative motions of the space traveler and the Earth are the same. They contend that the space ship is moving away from the Earth at precisely the same speed with which the earth is receding from the space traveler.

Therefore if the clocks on the space ship are slowing down with respect to those on the Earth, by the same token the clocks on the Earth are slowing down with respect to the space traveler.

Thus if the space traveler left the Earth at great speed and returned at some future date, the clocks on both the space ship and the Earth would precisely coincide and there would be no aging of the "Earth people" with respect to those who went on the trip.

They say that the relative velocities make for perfectly symmetrical conditions and thus it is impossible to differentiate between those who left and those who stayed on the Earth. Therefore, they contend, it is ridiculous to assume that one has a special property which the other does not possess Thus there can be no difference in ages.

To this, Prof. McCrae and his fellow scientists say "nonsense!" They hold that the space ship is being accelerated away from the Earth. The space travelers are the ones who are doing the moving by virtue of this acceleration. Therefore, if they are accelerated, then their clocks will slow down and the people on the accelerated vehicle will live longer.

They contend that the frame of reference is centered on the Earth and it is the space ship which moves in this frame of reference. The Earth is stationary in this system so that symmetry of motions does not exist. The consequence of this symmetry is that the aging process of the space travelers will be retarded.

Thus the controversy rages.

Medical men also have become involved in this controversy. A team of doctors from Harvard University, as reported in a paper given before the American Rocket Society, have investigated this from the metabolic point of view and conclude there would be no difference in ages between the space traveler and the Earth people.

Rather passionately, the doctors dispute the findings of one of Dr. Einstein's colleagues, Philip Frank. Of Dr. Frank's conclusions, they say:

"He compares this slowing of life's processes, due to increased uniform motion, to the slowing which occurs at a reduced body temperature. So the conclusion given is that the motion of the space ship inflicts a kind of hibernation on its passengers and they arrive home awakening, like the Sleeping Beauty and her household, to find a new generation on Earth while they themselves are but little older."

The medical men probe the ability of the body to withstand conditions which are alien and perhaps lethal to a human being. This slowing down of time to them means hiberation and hibernation means a dropping of the body temperature. They claim that a drop of 11 or 12 degrees brings unconsciousness and a possibility of death. Certainly the performance of the individual will be critically affected if there should be hibernation.

These doctors, like Prof. Dingle, reject the idea partially on the grounds of "common sense," because common sense insists that a sort of hibernation is the only logical consequence of this situation. Therefore, in their paper, as in the case of Professor Dingle, they insist on the symmetry of velocities of the space ship and the Earth. And like Professor Dingle, the medical men prefer to ignore the acceleration of the space ship to get it up to the high speeds. It is the opinion of many scientists, strongly shared by this writer, that accelerations cannot be ignored. Only by permitting it to enter the picture can a definitive answer be found.

If the special theory of relativity is completely valid, and at this time there is a strong feeling that it is, then at some future date man will come face to face with the possibility of a long life span. In this speculation we are not just juggling time sequences. The human aging process would be slowed. It can be explained in the following manner;

Imagine the human body as a clock of sorts. The balance wheel may have its counterpart in the heart. Thus the beating of the heart would measure time. As the space traveler moves at a high speed, the slowing down of the clocks on the space ship would be simultaneously accompanied by the slowing down of the heart in its beating. The normal heart beats 72 times a minute. A sufficiently high speed would make the heart beat 72 times an hour!

Run the speed up still higher and the heart may beat but once a day. But, remember, the unit of time we are using is the unit of time as measured on the Earth. The crew on the rocket ship would be completely unaware of the slowing down of their clocks. If they took their pulse using the clocks on the space ship, the rate would still be 72 a minute.

The metabolism of the body would be retarded until the functioning of the body practically ceased. In that way the body could survive for inordinately long travel periods.

This time dilation can be easily computed. We simply begin with a number, 186,300 miles a second—the speed of light. Now if a space ship can be accelerated until it attains a speed of 167,700 miles a second, approximately 90 per cent of the speed of light, the traveler would age but 10 years for a trip which the stay-at-homes on Earth said had taken 23 years.

If the speed of the traveler goes up to 184,000 miles a second, 99 per cent of the speed of light, a 100-year trip would take 5 Earth-years. With a speed of 186,200 miles a second, 99.99 per cent of the speed of light, a 1000-year trip would take but 14 Earth-years. The closer we come to the limiting speed of light, 186,300 miles a second, the more time is slowed until at the speed of light time stops.

A good bit of stimulating speculation has centered around the behavior of a space traveler moving with a large fraction of the speed of light.

In my office I can walk for a distance of 60 feet and without exertion I can cover this in about 10 seconds. Now imagine me in a space ship travelling at 186,200 miles per second. Instead of covering this distance in 10 seconds, an observer on the Earth, if he could see, would clock me over this distance in about 170 Earth-seconds and to him it would appear as though I were performing in a very slow, slow-motion picture.

Similarly, all my other movements would be slowed down. Eating, which might normally take a half hour on the Earth, would, in the fast moving space ship, seem to take the best part of a day. Even my reflexes and automatic actions would be expanded in this weird fashion.

If I see something out of the corner of my eye, it normally takes about one-tenth of a second for me to perceive it. On the space ship it would take almost a second for the message to get to my brain. After my brain received the message, instead of taking the normal one second to recognize what I saw, it would take almost a quarter minute for me to recognize it. All my functions would be decelerated, but to me on the space ship they would appear quite normal. Only to the stationary observer on the Earth would this time scale appear dilated.

In essence, those who speculate on what the future will bring are, in bold imagination, embarking on a trip through time. Let us assume that in 1928 a 28-year old couple departed on a 30-year trip in which they traveled at 99.2 per cent of the speed of light. Let us further assume that they left behind a one-year old daughter. While the clocks on Earth were ticking away the passage of 30 years, their clocks would indicate but three years of time had elapsed. The 28-year old couple would return and be 31 years old.

Every woman loves to hear the flattering remark when she is with her daughter that they look almost like sisters. If our people went out to dinner in 1958 with their daughter, then truly could the remark be made to the mother:

"Why you look like sisters, not like mother and daughter." Mother would be 31 years old, just the age of her daughter. This is one time the flattery would not be empty. If the women are really serious about staying young, and what woman isn't, here may be the solution to this longing.

If the application of this theory does slow down the aging process, will we become aware of it soon? Can we look forward to a definitive solution to this question in the immediate future? The answer is no.

While within perhaps 20 years man may travel to a space station and thus attain a speed of 5 miles a second, this speed is so slow that in a life span of 100 years the inhabitants of the space station would gain but one second over those remaining on earth. Only in the distant future when trips to the outer reaches of the solar system or even to the stars are contemplated, can the effects become appreciable enough to answer the question.

And if the answer should demonstrate the complete validity of this theory, then perhaps this Earth is due for a staggering social upheaval in which people would demand space travel as a sort of elixir of life.