

15 October 1965

NASA PATTERN

RELEVANCE GUIDE

Volume III

TECHNOLOGY DOCUMENT

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This document has been prepared as a part of NASA Contract 8-20207 and is intended to contain summary information to be used in assigning relevance to elements on the PATTERN relevance tree, levels seven through ten.

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FOREWORD

Some preliminary comments regarding the organization and intended use of the technology volume of the Relevance Guide are in order here.

SYSTEMS (LEVEL 6)

In the NASA PATTERN Study, those hardware entities which would be required to make a Concept (Level 5) operationally possible have been termed "Systems" (Level 6). Examples of "Systems" are: Booster, Spacecraft, GSE, Lander, Probe, Rover, etc.

SUBSYSTEMS (LEVEL 7)

The hardware devices necessary to make a "System" operational have been grouped into "Subsystems" (Level 7). Examples are: Auxiliary Power, Data Processing, Life Support, etc.

Some arbitrary assignments were necessary. The aim here was not to produce an exhaustive operational analysis of each concept, but rather to establish the relevances of several well-known technological deficiencies. These numbers will in turn provide a "yardstick" against which the desirability of various Apollo technology experiments can be estimated.

FUNCTIONAL ELEMENTS (LEVEL 8)

Within each subsystem there may be several "Functional Elements" (Level 8). For example, an auxiliary power subsystem may require an energy source, an energy conversion method, an energy storage method, and a power conditioning method. No vote is necessary at this level unless the group of technology deficiencies is large. Where a choice clearly was of significance it was included in the tree.

CONFIGURATIONS (LEVEL 9)

In some cases several different subsystem configurations are applicable as alternates for a given functional element; in such cases a choice node for

"Configuration" (Level 9) has been introduced. As an example, the auxiliary power of a certain spacecraft might be supplied by either solar cells plus a storage battery or by an isotope thermoelectric generator. The relevance of such a choice is assigned by the voters. In many other cases alternate configurations were clearly not competitive. In such cases no choice is possible at the node and none is offered.

TECHNOLOGY DEFICIENCIES (LEVEL 10 or *)

Associated with each subsystem configuration are certain technology deficiencies or TDs (Level 10). These are problem areas which require further work to make the subsystem perform as desired. In some cases the TD represents a critical problem area, but in others it merely represents a need (or desirability) for upgrading present technology in order to better meet the estimated subsystem requirements by the operational date considered. No claim of completeness is made for the TD list of the present study; more TDs will be added as expert criticism is obtained, and as deeper study of missions reveals them.

PURPOSE OF VOLUME III

The purpose of this document is to provide minimal background information on the state-of-the-art, and explanatory comments to more clearly define the sometimes cryptic "keyword" abbreviations used to describe TDs on the ballots. The presentation is pitched to the level of an already well-informed individual and is to provide him with adequate detailed information upon which to base his judgements of relevance. It is not intended to be a textbook in general space technology. Neither does it completely cover all of the space systems of future interest since it is confined to those selected for PATTERN concepts.

ORDER OF SECTIONS

The order of sections in this document is the same as the listed order of subsystems on the concept sheets of Volume II. Each section begins with a general discussion of the state-of-the-art. This is followed by a discussion of the configurations and their associated technology deficiencies in about the order in which one would encounter them while leafing through Volume II, except that repetition has been eliminated wherever possible.

ORGANIZATION OF SECTIONS

Configurations and technology deficiencies are discussed in this volume under paragraph headings which are the same keyword groups used to refer to them in Volume II. For example, in Volume II under "Propulsion" subsystems one will find "Cryogenic Liquid Rocket" as a configuration. A typical TD is "cryogenic storage". A discussion of this TD may be found in Section 9 of this document under these paragraph subheadings. The appropriate related introductory figures or paragraphs are referenced by number under the keyword headings.

SOURCES AND REFERENCES

Various authorities in the open literature are referred to and directly quoted wherever possible, in preference to merely stating the opinions, or limited knowledge, of the study group. Throughout this we have diligently tried to quote accurately and to give credit where it is due. If mistakes have crept into this process we apologize to those concerned. In some cases it has been impossible within the time allowed to determine the ultimate source; nevertheless, we have presented such information whenever it seemed to be the best available.

A reference list is provided for each section and sources are credited (and thanked) by reference number.

The inclusion of classified information has been avoided in order to permit easier use, storage and disposal; because of this (as well as normal human error) some of the "state-of-the-art information" will conflict with the individual's personal knowledge. He should be guided by his own knowledge in such conflicts.

Section 1

AUXILIARY POWER

1.0 GENERAL COMMENTS

Quoting from Ref. 1.1 (Space/Aeronautics): "Propulsion aside, it is in the area of auxiliary power that the need for intensive development is most clear. Unmanned missions of the early 1970s that proceed no further from the Sun than Mars, such as Boyager (which may need about 1kw), are likely to use solar cells. Manned missions through the early 1970s are likely to depend at least partly on improved fuel cells with their water by-product. But extended lunar expeditions, manned Mars fly-bys and large manned space stations will require power levels approaching 50 to 100 kw, and lunar bases and manned planetary landings will require over 100 kw. For these, the development of nuclear reactor powerplants is essential. One such program, Snap-8 with a goal of 35 kw, appears to be on the verge of extinction. . . The still-more-ambitious Snap-50, a longer-range experiment also cut back . . . would furnish 300kw; something of this order would be needed in a nuclear-electric propulsion system." (End of quotation from Ref. 1.1)

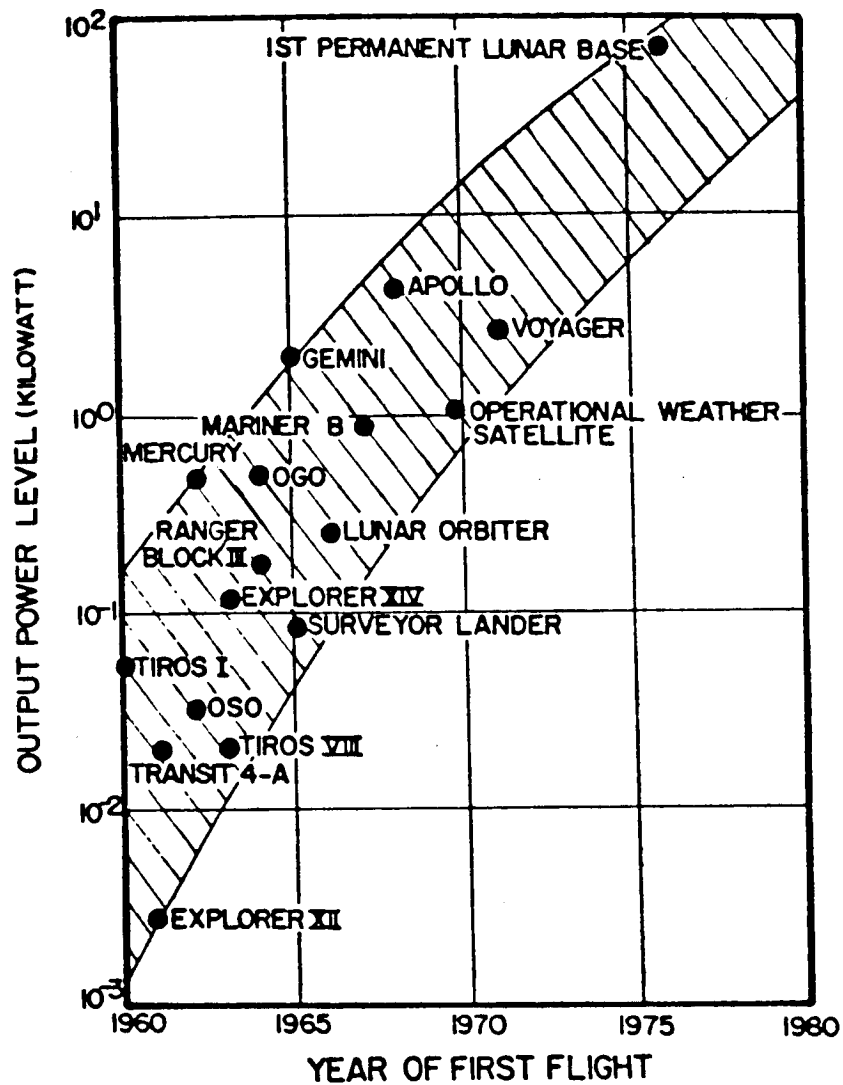
Figure 1-1 shows the anticipated growth of power requirements over the next 15 years. As can be seen from this graph, the general trend is toward higher and higher power levels. The performance of the subsystems in use today is simply not adequate to obtain the projected power requirements. Thus, new concepts must be evolved which can adequately meet the performance objectives.

Figure 1-2 shows an extrapolation of the 1960 Aerospace Industries Association forecasted estimates of the power capacity of the indicated systems.

Figures 1-3 through 1-5 show regions of power level versus mission time wherein the indicated subsystem is the preferred compromise.

Table 1-1 presents a summary of several power subsystems which have been used.

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FIGURE 1-1. PROJECTED SPACE ELECTRIC POWER REQUIREMENTS (Ref. 1.2)

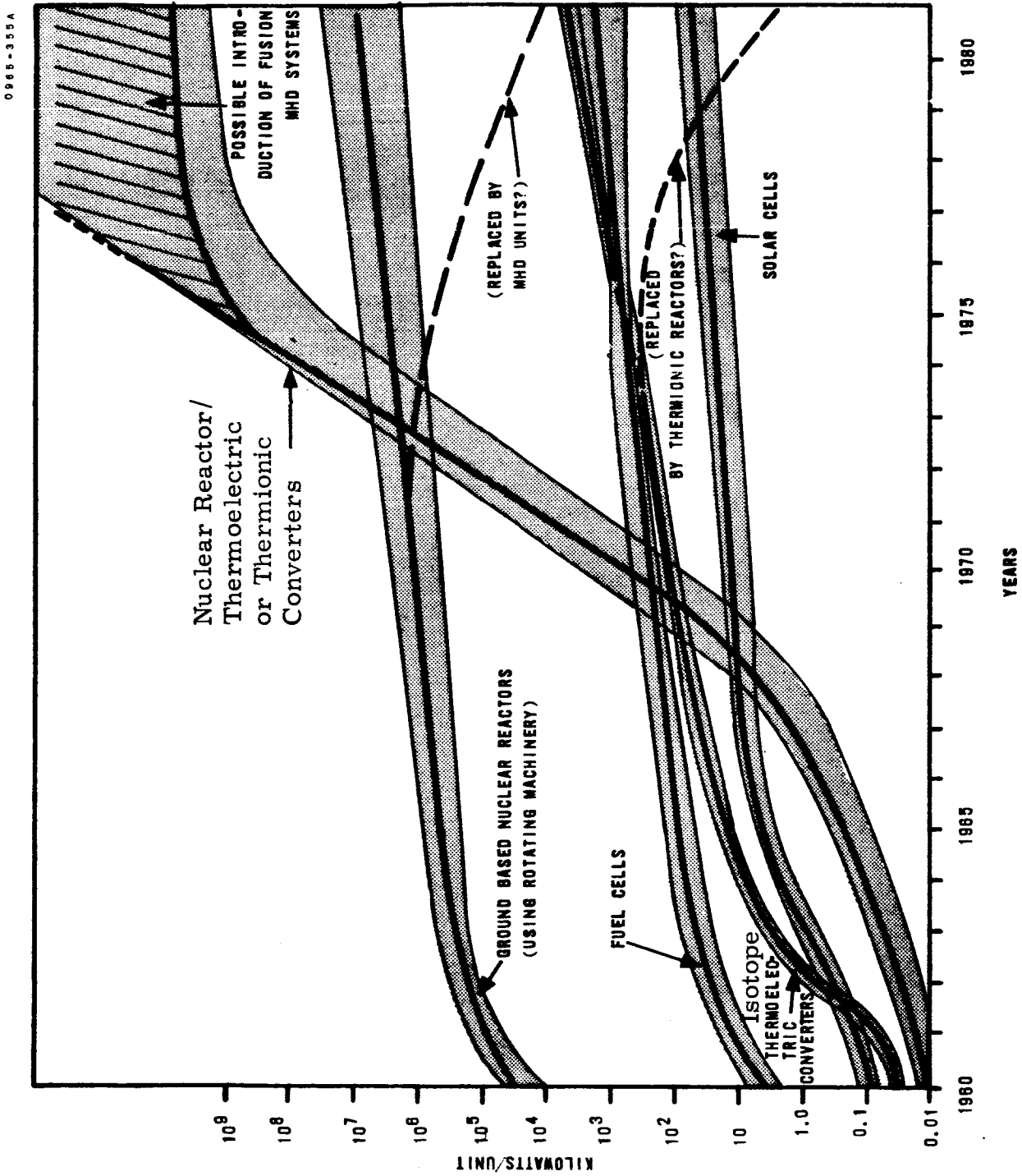
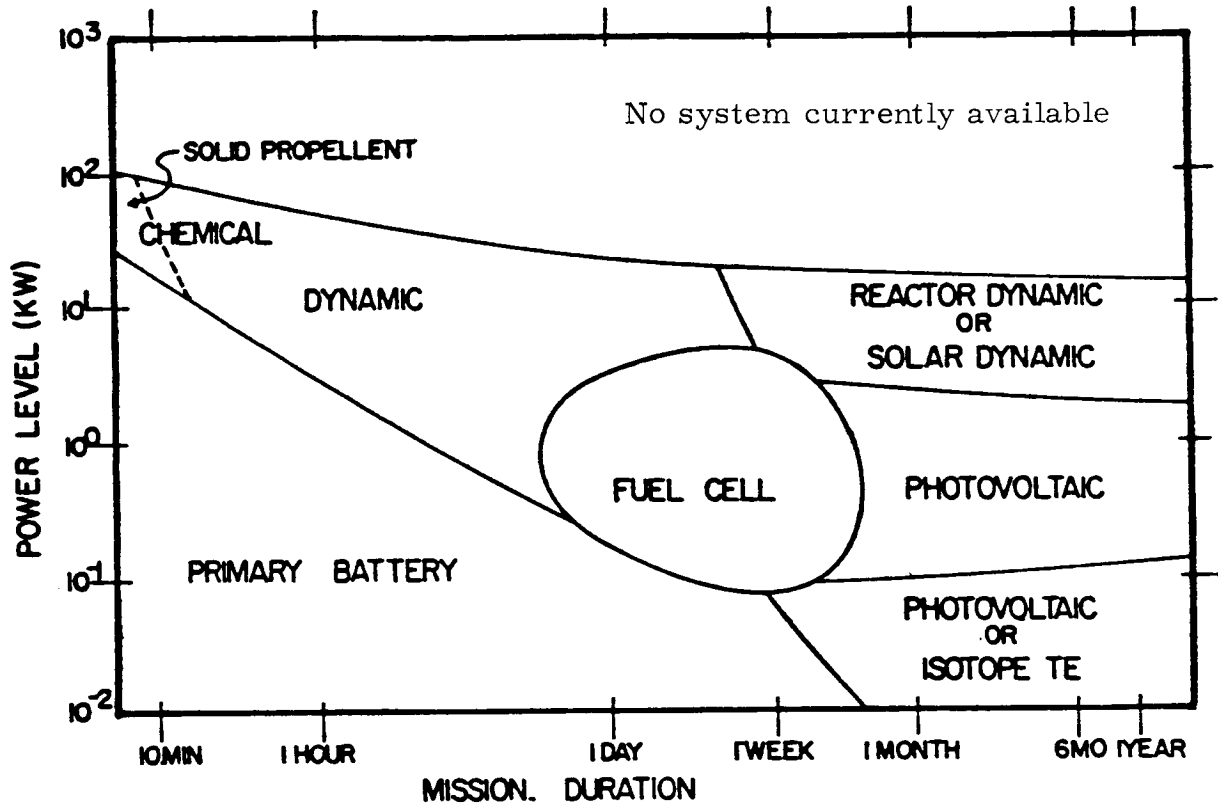


FIGURE 1-2. UTILIZATION OF NEW ENERGY SOURCES
(Numbers from many authorities were used to extend chart from Ref. 1.3)

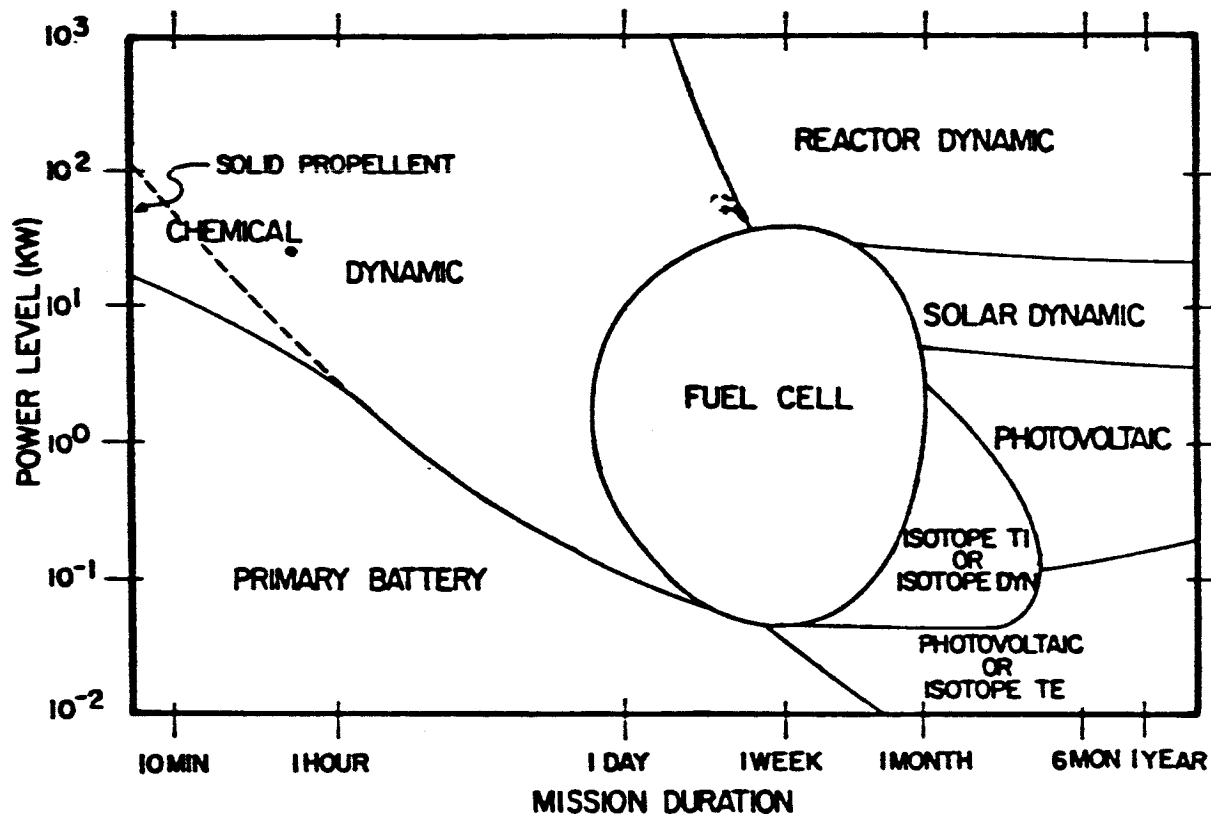
0965-357A



Gene B. Stafford

FIGURE 1-3. PRESENT APPLICATION AREAS
(Ref. 1.2)

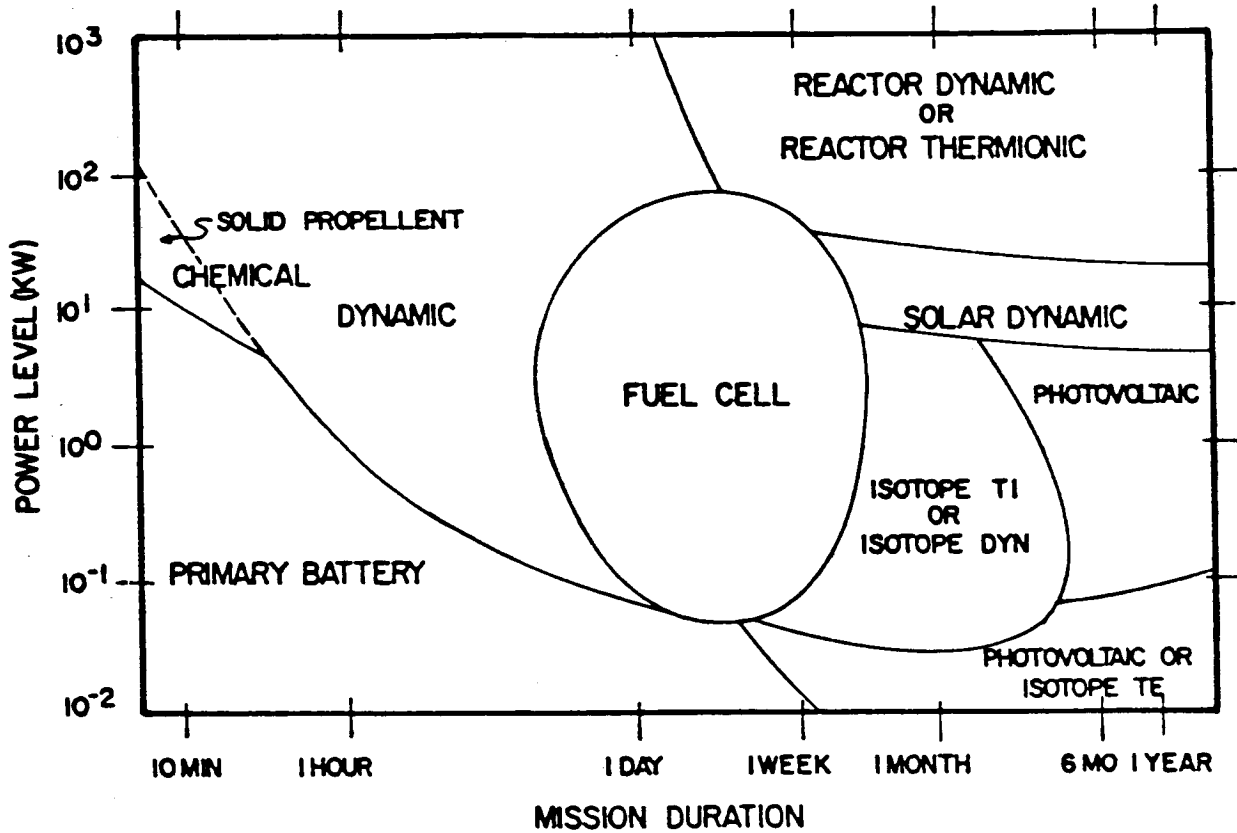
0965-358A



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FIGURE 1-4. 1968 APPLICATION AREAS
(Ref. 1.2)

0965-359A



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FIGURE 1-5. 1972 APPLICATION AREAS
(Ref. 1.2)

TABLE 1-1. SOME POWER SUBSYSTEMS IN PREVIOUS USE
SOURCE: Gene B. Stafford (Ref. 1.2)

<u>Space System</u>	<u>Power Subsystem</u>	<u>Power Level (watt)</u>	<u>Subsystem Weight (pounds)</u>	<u>Remarks</u>	<u>Subsystem Weight (kilograms)</u>
DOD SAT.	Isotope TE	2.7	4.6	Snap 3. 2 Launch 1961	2.086
DOD SAT.	Isotope TE	25.	27.	Snap 9-A, 2 Launch 1963	12.24
NIMBUS-B	Solar Cell	470.	191.	Panel, 1st Launch 1965	86.64
OGO	Solar Cell	535.	173.	Panel, 1st Launch 1964	78.47
OSO	Solar Cell	31.	35.9	Panel, 1st Launch 1962	16.28
RANGER	Solar Cell	180.	815.	Panel, 1st Launch 1964	369.68
MARINER	Solar Cell	250.	570.	Paddle, Launch 1964	258.55
UK-1	Solar Cell	11.7	22.8	Paddle, 1st Launch 1962	10.34
EXPL. XII	Solar Cell	20.4	17.3	Paddle, 1st Launch 1961	7.85
EXPL. XIV	Solar Cell	34.7	19.1	Paddle, 1st Launch 1963	8.66
TIROS (45 deg)	Solar Cell	51.	64.5	Body, 1st Launch 1960	29.26
RELAY I	Solar Cell	35.	53.8	Body, 1st Launch 1962	24.4

1.1 SOLAR CELLS

Quoting from Ref. 1.2 (Gene B. Stafford): "Solar cells have been used more extensively than any other power subsystem for space application.

"Although a number of materials are being investigated, silicon must be considered as the primary candidate due to its demonstrated performance and advanced stage of development. The cells can be mounted on panels, on paddles, or on the skin of the satellite, depending on the mission requirements and constraints." (While paddles are more efficient in terms of weight and area, they require orientation toward the sun; this affects design of stability and control. - Ed.)

"The present level (1965) of solar cell technology is 10 to 11 percent efficiency, power to weight ratios of 10 watt/lb (22 w/kg), and weight to area ratios of 0.90 lb/ft² (4.35 kg/m²) for panel mounted cells 0.80 lb/ft² (3.87 kg/m²) for paddle mounted cells, and 0.70 lb/ft² (3.38 kg/m²) for body mounted cells. The cost of present silicon cells is about \$300/watt.

Future improvements in solar cell performance will come through improved production (quality control) techniques, the manufacture of thinner and more radiation resistant cells, and better array fabrication techniques. The development of lighter weight substrates and integral cover glasses will further enhance the performance increases that can be expected.....

With continued effort future silicon solar cells should have efficiencies of 15 to 16 percent, power:weight ratios of 20 watt/lb (44 w/kg) and weight to area ratios of 0.40 lb/ft² (1.95 kg/m²) for panel mounted cells, 0.33 lb/ft² (1.61 kg/m²) for paddle mounted cells and 0.26 lb/ft² (1.27 kg/m²) for body mounted cells. The cost for these advanced cells will be on the order of \$100/watt for panel mounted cells." (1965 equivalent money) (end of quotation Ref. 1.2)

Quoting from Ref. 1.4 (George C. Szego): "Typical properties of spacecraft solar cell systems are shown in Table" 1-2. "Efficiencies stated thereon are based upon solar array output under the most optimum condition of the spacecraft aspect to sun angle." The table "does not take into account losses incurred by the power system's primary or secondary" conditioners.

1.1.1 Radiation Damage

"Solar cell power systems can now be designed to provide electrical power for at least several years in the radiation belts. This may be accomplished by shielding, overdesigning, or a combination of both. There is appreciable variation in the radiation resistance of one lot of cells to another

TABLE 1-2. SPACECRAFT PHOTOVOLTAIC POWER
SYSTEM CHARACTERISTICS*

	NIMBUS "B"	EGO	OSO	OAD	UK-1	Expl. XII	Expl. XIV	RELAY	TIROS	TIROS	IMP
Array ^a	O	O	O	NO	NO	NO	NO	NO	NO	NO	NO
No. Solar Cells	11,000 ^b	32,500	1,872	53,000	4,256	5,600	6,144	8,400	9,120	9,120	11,520
Cell Mounting	Panels	Panels	Panels	Panels	Panels	Panels	Panels	Body	Body	Body	Panels
Weight of Array, lbs.	64	127	5.2	222	8.8	11.0	12.8	25.8	24.5	24.5	26.5
Weight of Storage, lbs.	113	74	30.7	177	14	6.3	6.3	28	40	40	6.7
Weight of Power System, lbs.	177	201	35.9	399	22.8	17.3	19.1	53.8	64.5	64.5	33.2
Max. Power Dev., watts	410	560 ^g	31 ^c	772 ^d	11.7	20.4	34.7	35	51 ^e	25 ^f	74.3
Array, watts/lb.	6.4	4.4	6.0	3.65	1.3	1.85	2.7	1.4	2.1	1.1	2.8
System, watts/lb.	2.3	2.8	0.86	1.94	0.52	1.2	1.8	0.65	0.79	0.39	2.2
Array, Area, ft. ²	43	78	4	187	11	15.3	15.3	17.6	17.7	17.7	28.7
Array, watts/ft. ²	9.5	7.2	7.7	4.14	1.1	1.3	2.3	2.0	2.9	1.4	2.6
Array, Efficiency, percent	7.3	5.5	5.9	3.2	0.85	1.0	1.8	1.5	2.2	1.1	2.0

a - "O" Oriented, "NO" Not Oriented

e - At 45 degrees incl. to Sun

b - 2 cm x 2 cm cells

f - At 90 degrees incl. to Sun

c - At 70°C

g - At 60°C

d - Ave. Power from cold to hot panel

*From(Ref. 1.4)G. Z. Szego, AIAA paper 64-325, "Space Power Systems State of the Art", 29 June 1964.

and a better understanding of the effect of trace impurities, base resistivity, and impurity profile will lead to even more radiation resistant devices. The heavily protected Vanguard I has operated continuously for over 5-1/2 years"(by 1963 - Ed.).

Table 1-3"portrays the approximate dosages (ψ_c) necessary to reduce the sunlight maximum power to 75 percent of its initial value for N/P, P/N silicon, and P/N gallium arsenide solar cells with various monochromatic energetic particles."

"Sufficient satellite data have been transmitted back to Earth firmly to establish the superiority of N/P versus P/N silicon solar cells to radiation belt environment.

"While six-mil shielded gallium arsenide cells are more radiation resistant and have better temperature properties than silicon cells, the current state of the art does not warrant their use on today's spacecraft missions. Certain future missions involving high temperature and high radiation field environments might require this device (examples, mercury and solar probes).

1.1.2 Abrasion Damage

"It is clearly recognized that bare solar cells of any type cannot be flown on long-life missions due to meteoritic abrasion damage. Six mils of cover glass appears to be a practical minimum from the standpoint of protection of energetic particles and the handling of the cover slips during array construction." (This, of course, increases overall weight and slightly reduces efficiency / end of quote from Ref. 1.4).

1.1.3 Specific Power

The power per unit weight (kw/kg) of solar cells is limited chiefly by the mass of inert material which must be used behind the exposed junction.

One approach to improving this is the thin film cell . Quoting from Ref. 1.2: "Although the efficiency of these cells is low at this time (three percent), the extreme light weight (0.15 lb/ft^2)"(0.731 kg/m^2) "makes them attractive for space power generation. The operation of the cells is not completely understood at the present time but research on the basic physics and on production methods should yield six to seven percent efficient cells for 50 to 75 dollars/watt." (End quote from Ref. 1.2)

TABLE 1-3. AVERAGE CRITICAL DOSAGES OF
GaAs AND Si SOLAR CELLS
(SUNLIGHT MEASUREMENTS) From Ref 1.4:

Bombarding Particle and Energy	θ_c (Particles/cm ²)		
	GaAs	Si N/P	Si P/N
0.8 Mev electrons	1.1×10^{15}	1.3×10^{15}	4×10^{13}
5.6 Mev electrons	2.7×10^{14}	3.0×10^{13}	$2 \times 10^{12*}$
0.1 Mev protons	$\sim 10^{12}$	$\sim 10^{13}$	$\sim 10^{13}$
0.4 Mev protons	$< 10^{11}$	$\sim 10^{11}$	$\sim 10^{11}$
1.8 Mev protons	2.4×10^{12}	1.3×10^{11}	4×10^{10}
17.6 Mev protons	5.7×10^{12}	4×10^{11}	7×10^{10}
95.5 Mev protons	$> 10^{12}$	7×10^{11}	2×10^{11}

* Private communication from D. Brown, NASA to G. C. Szego

(Quoting from Ref 1.4):

"Thin film solar cells made by evaporation of cadmium sulphide and cadmium telluride have shown sunlight efficiencies of four to five percent on small areas (cm²) but only one to two percent in large (6 by 6 inches) areas. All other materials, such as GaAs, InP, Se, GaP, are still in a primitive state of development. However, the way to extremely lightweight, low cost solar cells appears to be in this direction." (Flexible arrays have been made experimentally under USAF ASD sponsorship. - Ed./ End quote from Ref 1.4).

1.1.4 Specific Cost

The cost of solar cell arrays in terms of dollars/kw tends to become a pacing factor for systems requiring high power. For example, a lunar colony may require 600kw of total power. At \$300,000/kw, this would amount to 180 million dollars for the "lunar day" power. Since an equivalent amount must be stored for the lunar night, \$360 million or more would be required just for the solar cells, without the rest of the subsystem.

Such a consideration generally limits solar cells to relatively low power applications below 2 kw.

According to Szego (Ref 1. 4):

The 1963 "cost of five to six dollars per 1 by 2 cm cell corresponds to \$250 to \$300 per electrical watt for unassembled silicon cells. Laboratory-built gallium arsenide solar cells cost around \$150 to \$300 per 1 by 2 cm unit. Including the cost of substrates, cover glasses, and the cost of array assembly, cost per watt for silicon cells range from \$1,000 for oriented arrays to as much as \$8,000 for spin stabilized spacecraft. Automation techniques in material preparation and cell fabrication as well as cell standardization could reduce these costs substantially," (and some progress has occurred during the past two years--Ed)

1.1.5 Specific Area

The solar cell can deliver no more electrical energy than the incident illumination energy reduced by the efficiency factor. Since incident energy is limited, high power systems require large area arrays. At about 85 watts per square meter, a one-kilowatt subsystem requires about 118 square meters of area (about 1,294 square feet--equivalent to the roof area of a small suburban house) at Earth's distance from the Sun.

The required power often calls for an area in excess of the vehicle's total area, and panel arrays are needed. Since less than half of the vehicle's area can be illuminated effectively, non-oriented arrays will require over twice the area of oriented panels for the same power. As mission time extends to 5 years about 50% extra must be allowed for cell degradation.

1.1.6 Deployment

Only the projected area normal to the incident light is effective in producing electricity, therefore it is desirable to make the panels track the Sun. This often requires that they be torqued with respect to the vehicle; doing so places a load upon the stabilization subsystem. In long term satellites the consumption of propellant can be a serious factor. In gravity gradient systems the disturbance can be intolerable.

Panels must be folded away for exit from the atmosphere, then deployed in space. This requires an electromechanical device. Sometimes centrifugal or pyrotechnic actuators are used, also.

The efficiency or even operability of the power system is contingent upon reliability of the deployment mechanism. Some failures have occurred.

1.1.7 Bearings, Solar Cell Arrays

In satellites orbiting the Earth and stabilized to local vertical, the solar panels must make one revolution per orbit with respect to the vehicle. Required motion is always in the same sense. Bearing friction and bearing seizure could result in sacrifice either of power, or stability, or both. Friction must be compensated at the cost of control moments. The alternative is a heavier, non-oriented array.

1.1.8 Degradation of Materials in Space

Lubricants tend to harden in vacuum due to loss of volatiles. Similarly, plastics lose plasticizer and become brittle. Glasses often darken due to radiation damage. Meteoritic dust tends to abrade any exposed surface (reducing transparency of cover glasses). Thermal cycling tends to harden many materials and can result in brittleness of metals.

Such degradation usually must be compensated by overdesign with consequent weight, volume, and cost penalties. In general, the longer the mission, the greater the factor of safety. A five-year mission would approximately double a solar cell system's required size.

1.1.9 High Temperature Operation

Solar probes and Mercury flybys will involve an additional payload penalty due to the requirement for a separate high-temperature auxiliary power supply designed for survival at perihelion temperatures. If new solar cell types (or better cooling methods) can be devised, it might be feasible to use a single power subsystem during the entire trip. Present silicon solar cells seem to deteriorate at temperatures much above 150°C. Unshaded perihelion passage may involve skin temperatures up to 600°C or more.

1.2 SOLAR DYNAMIC SYSTEMS

Figure 1-6 under nuclear reactors shows a typical solar dynamic system schematically. Solar dynamic power systems (summarized in Table 1-4) (Quoting from G. C. Szego (Ref 1.4): "have a unique advantage in that they need no shielding... The solar collector is not independent of the mission path... as is a reactor. Therefore, solar plants utilize both orientation for the collector and heat storage for periods when the vehicle is out of direct sun. The heat storage requirements currently limit solar power systems to a few peak cycle temperatures. For example, the fusion temperatures of lithium hydride and lithium fluoride are 1,250 and 1,550°F. These temperatures are consistent with hydride moderated and uranium carbide reactors so that conversion equipment senses about the same inlet conditions in both systems. The fluoride heat storage is much more attractive for Stirling and Brayton cycle use, while Rankine systems can use both, but not with the same working fluid. Lithium fluoride, however, has less than half the heat of fusion of lithium hydride so that system weight advantages of the higher temperature are substantially offset by the increased heat storage weight. It has been demonstrated that hydrogen disassociation from the LiH due to temperature and cycling is less than one pound per year in Sunflower tests. There has been no hydrogen embrittlement of materials at these high temperatures. The practicality of lithium hydride has been demonstrated while work with lithium fluoride is in process. Due to container-heat storage material combined weights, the trade-off cross-over point appears to be about 2.7 percent efficiency of the conversion device. Below this figure, electrical storage is lighter; above this figure, thermal storage is lighter.

"The power conversion portion of a solar power plant is similar to that of a nuclear plant... Equipment-wise, this is an advantage, for the same conversion equipment may be used for either solar or nuclear applications."

The Stirling Cycle (Refer to Figure 1-6) is of especial interest to the solar plant. "This system utilizes an inert gas, as in the Brayton cycle, in a closed loop with a piston engine instead of a turbine to do the work. Principal advantage of the Stirling engine is high efficiency at low power levels. A conversion system efficiency of 30 percent and over 1,000 hours life at 4 kw has been demonstrated. Another advantage is that most moving parts operate in a temperature of about 250°F, which is conducive to long life and utilizes previous experience." (End quote from Ref 1.4.)

TABLE 1-4. SOLAR DYNAMIC SYSTEMS
SOURCE: G. C. Szego (Ref 1. 4)

1983

STATE OF ARY

[illegible]

1.2.1 Waste Heat Rejection (At Low Temperature)

Quoting from G. C. Szego (Ref 1.4):

"Principal problems of the Stirling engine are heat rejection at low temperatures (150°F to 200°F) and heat transfer in the regenerator, especially as complicated by lubrication and seal problems. As in the Brayton system, high component efficiencies are needed to have a competitive system." (End quote)

The problem of maintaining an adequately low temperature in the condenser portion of any thermal cycle is made difficult in space since all waste heat must be radiated away. This requires large area condenser/radiator panel heat exchangers. Furthermore, the heat radiators must be directed away from the Sun which implies a configuration constraint. Since the concentrator mirror must face the Sun, however, the problem is less severe in solar concentrator systems since the shade area behind the mirror is available for rejection panels.

Since a working fluid moving within a thin metallic shell is the preferred heat exchanger method for dynamic systems, the danger of fluid loss due to meteoritic puncture represents a risk. Increasing metal thickness can reduce the risk, but also increases the necessary radiator area due to lowered efficiency.

1.2.2 High Temperature Materials

Operation at higher inlet temperatures could increase overall efficiency and permit rejection of waste heat at a higher radiator (outlet) temperature. This implies use of a higher temperature working fluid (such as a sodium potassium eutectic alloy) and requires refractory, durable containers and turbine materials.

The operating temperature, and hence the efficiency, of both nuclear and solar dynamic systems is limited by the available refractory materials. Materials R&D hopefully will permit eventual operation in the 1,800 to 2,000°F region without a serious weight penalty.

Table 1-5 summarizes the various dynamic cycles considered.

1.2.3 Wearout, Rotating Machinery

Dynamic cycle systems usually depend upon either sliding pistons or rotating turbines to drive a rotary generator. Lubrication of bearings for these elements must be at the working fluid temperature. The bearings will have the entire working pressure head across them, unless the entire rotating machinery unit is pressure sealed.

TABLE 1-5. DYNAMIC ENERGY CONVERSION SUMMARY

SOURCE: Gene B. Stafford (Ref 1.2)

Subsystem	Status	Cycle	Working Fluid	Power Level (kw)	Efficiency (percent)	Specific Weight (lb/kw)	Specific Area (ft ² /kw)	T _{in} /T _{rej} (°F)
SNAP 2	System Improvement	Rankine	Mercury	3.0	6.0	320.	40.	1150/600
SNAP 8	Development	Rankine	Mercury	35.	8.0	142.	53.	1250/580
SNAP 50/SPUR	Development	Rankine	Potassium	350.	---	16.	2.5	----
SUNFLOWER	Terminated	Rankine	Mercury	3.0	9.0	117.	16.6	1250/607
ASTEC	Program Definition	Rankine	Water	20.	15.4	67.	32.2	1480/235
ASTEC	Program Definition	Rankine	Mercury	20.	19.6	54.	7.1	1250/504
ASTEC	Program Definition	Rankine	Biphenyl	20.	15.4	85.	14.2	700/365
ASTEC	Program Definition	Brayton	Argon	20.	23.4	119.	14.8	1390/490
-----	Study	Brayton	Argon	5.0	13.7	120.	66.	1660/620
-----	Study	Brayton	Argon	35.	16.8	70.	42.	1760/650
-----	Study	Brayton	Argon	100.	13.2	50.	28.	1960/770
-----	Study	Rankine	Biphenyl	1.5	14.-19.	150.	60.	----
-----	Study	Rankine	Mercury	1.7	13.-15.	120.	22.	----
-----	Study	Brayton	Argon	1.5	16.	300.	60.	2000/545

Like other rotating machinery, these power systems have a life time limited by bearing wearout and turbine blade erosion. They have operated for over 1,000 hours without failure, but their use in long term missions is open to question.

Research in bearings, lubrication and turbine blade materials, along with development of the thermal cycle machinery is expected to reduce this doubt during the coming decade.

1.3 STORAGE BATTERIES

One drawback to the utilization of solar energy in space is the fact that for lunar and Earth orbit missions, energy storage must be available for dark time operation if continuous power is required. There are two methods of energy storage which are most frequently considered: electrochemical and thermochemical energy storage (see discussion of thermochemical under "Solar Dynamic").

Electrochemical storage is usually accomplished with secondary batteries. A secondary battery differs from a primary battery in that the secondary battery operates on a charge-discharge cycle. Although batteries have been used for years and are often considered to be well developed, a number of deficiencies and limitations have appeared when the batteries are used in space. Secondary battery performance can be increased substantially by work in the areas of increasing cycle life, better separator materials, higher allowable depths of discharge, and extending the temperature of operation.

1.3.1 Ni Cad Batteries

Specific Energy. Nickel cadmium batteries have been utilized extensively in space vehicles due to their long life but the device has a relatively low specific energy (KWH/Kg) compared with AgZn batteries (Ni Cad ~ 0.044 KWH/Kg, AgZn ~ 0.088).

Recycle Limited. Ni Cad batteries are usually specified for applications where long term missions are involved. Despite the fact that their chief virtue is in their outstanding capability to discharge and recharge many times, they are recycle limited (about 10,000 maximum) and even better batteries would be desirable. Their long life is purchased at a weight penalty. Silver Cadmium batteries with a third electrode appear to have recycle capability almost as good as that of NiCad, but also have better energy density.

Ni Cad batteries are not recommended for use in long day/night cycles due to the low specific energy. (They are replaced with AgCad for synchronous orbits, usually.)

High Load Efficiency. Where extreme loads for a short time are involved, it is necessary to overdesign a Ni Cad battery, if the mission duration will not permit use of AgZn batteries.

Reliability. MTBF of Ni Cad batteries is usually more than one year.

Specific Energy (KWH/Kg or WH/Gram). Table 1-6 shows comparative data on space storage batteries in English units under various conditions. Ni Cad batteries yield about 0.044 KWH/Kg as compared with 0.088 for AgZn. (There is some disagreement between the sources of information on storage batteries-- Ed.)

1.3.2 AgCad (Silver Cadmium) Batteries

According to Szego (Ref 1.4):

Silver-cadmium secondary batteries have now been flown on several satellites and given quite satisfactory service. Testing has advanced sufficiently to indicate that these batteries, with their considerably higher energy density, have cycle lives at least equal to the nickel cadmium system. The most significant recent development in the field of secondary batteries is the introduction of a third electrode-- usually connected to the cadmium electrode-- which permits a much higher rate of charging than has been possible heretofore. The auxiliary electrode, essentially a fuel-cell oxygen cathode, can be used either for sharply controlling the cut-off for charging or simply as a gas recombination device that avoids the danger of excess pressures being built up. Third electrodes will probably become available soon in both nickel cadmium and silver cadmium systems" (End quote from Ref 1.4).

Specific Cost (\$/KWH). The chief difficulty arises from the cost of silver metal; however, due to the higher energy density this is largely offset on a KWH/Kg basis, so that cost is roughly only 50 to 100 percent higher than for Ni Cad or AgZn batteries (\$2,000 to \$6,000/KWH versus \$3,100 to \$5,600/KWH for Ni Cad and \$2,700 to \$6,150/KWH for AgZn depending upon source quoting).

TABLE 1-6. SECONDARY BATTERY PERFORMANCE
 SOURCE: Gene B. Stafford (Ref 1. 2)

Type	Sil-Cad*			Ni-Cad			Sil-Zinc		
Temp. Range (° F)									
Upper	40.			40.			35.		
Lower	-10.			-10			15.		
Date of Availability	<u>Pres</u>	<u>1968</u>	<u>1972</u>	<u>Pres</u>	<u>1968</u>	<u>1972</u>	<u>Pres</u>	<u>1968</u>	<u>1972</u>
Rated Capacity (W-H/lb)	12.	13.5	15.	20.	22.	24.	40.	43.	47.
Effective W-H/lb									
At 25 percent DOD	3.0	3.4	3.8	5.0	5.5	6.0	8.0	9.5	11.5
At 35 percent DOD	6.0	6.8	7.5	7.0	7.7	8.4	14.	15.	16.8
At 45 percent DOD	9.0	10.1	11.2	9.0	9.9	10.8	18.	19.4	21.6
Effective W-H/in ³									
At 25 percent DOD	0.30	0.34	0.38	0.64	0.70	0.77	0.69	0.82	0.99
At 35 percent DOD	0.60	0.68	0.75	0.90	1.00	1.08	1.20	1.29	1.44
At 45 percent DOD	0.90	1.01	1.12	1.15	1.27	1.38	1.54	1.66	1.85
Specific Cost (\$/W-H)									
At 25 percent DOD	6.00	5.30	4.80	5.60	5.10	4.70	6.15	5.20	4.30
At 35 percent DOD	3.00	2.65	2.40	4.00	3.65	3.35	3.50	2.95	2.45
At 45 percent DOD	2.00	1.75	1.60	3.10	2.80	2.60	2.70	2.20	1.90
Cycle Life (Cycles x 10 ⁻³)									
At 25 percent DOD	10.	50.	100.	3.0	5.0	10.	0.3	0.75	1.5
At 35 percent DOD	4.0	15.	30.	2.5	3.5	7.0	0.2	0.5	1.0
At 45 percent DOD	2.0	5.0	10.	2.0	2.5	4.0	0.1	0.3	0.5

*Performance measured at 25, 50, and 75 percent DOD

Specific Energy (KWH/Kg). Although AgCad batteries excel in specific energy, they are far from ideal and easily lose out to fuel cells for large power requirements. Weight of required batteries becomes an important consideration in many missions.

1.3.3 Silver Zinc Battery

The highest energy density of the alkaline batteries is afforded by the silver zinc battery; however, this type of battery is limited to a low number of charge-discharge cycles. It does have the remarkable capability to operate effectively at very high discharge rates. This makes it attractive for short term missions or for use where extreme loads are delivered only occasionally.

Specific Cost (\$/KWH). AgZn batteries are presently the most expensive of those considered. However, with increasing use the cost is expected to decrease (see Table 1-6). This factor must be considered where very large power requirements exist, but is hardly a pacing factor for small loads.

Specific Energy (KWH/Kg). The present level of 0.044 to 0.088 KWH/Kg for AgZn batteries (depending upon temperature and discharge rate) leaves much to be desired. This limitation exists with all known secondary batteries and makes many otherwise feasible mobile power systems (for example: the electric automobile) somewhat impractical due to excess weight.

1.4 FUEL CELLS

Quoting from Gene B. Stafford (Ref 1.2):

"The fuel cell is one of the most actively pursued approaches to power being investigated today. In fact, the Gemini, Apollo, and possibly MOL will utilize this energy conversion process." (The chief advantage is efficiency of 45 to 65 percent in chemical to electrical energy conversion. This compares with about 35 percent for turbine power stations).

"As is the case for chemical dynamic units, there are any number of chemical reactants which can be utilized in a fuel cell. However, the utilization of the reaction of hydrogen and oxygen is by far the best." (The consequent production of water as a by-product is a valuable consideration. - Ed.)

Although a number of factors must be considered in the design of a fuel cell, perhaps the most influential is the current density at which the cell operates. Low current densities imply large and heavy cells but low reactant consumption while high current densities imply low weight and small volume but higher reactant consumption rates. The (service)"life desired from the fuel cell is also related to the current density in that at low current densities, longer durations are possible.

Future improvements in fuel cell performance will come through increased (service)"life capabilities at high power densities and through decreasing the reactant consumption rate at the high current densities. Table'1-7 "summarizes the present and projected performance of fuel cells.

The data in Table'1-7'represents only the hardware portion of the sub-system, that is, the fuel cell stack. To determine overall subsystem performance, the values for the reactant and reactant tank parameters must be added to those of the hardware." (End quote from Ref. 1.2)

1.4.1 Specific Power (Kw/Kg)

Present Gemini fuel cells deliver 0.0128 (Kw/Kg), that is about 5.8 watts per pound. It requires roughly one pound of reactants to produce one kilowatt hour of energy.

As Table 1-7 shows, by 1972 it should be possible to get 0.0735 Kw/Kg from fuel cells, that is about 33 watts per pound.

Compared with other subsystems, (0.6 to 0.9 W/pound for isotope TE, 0.22 to 3.0 W/pound for solar cells, 4.5 to 6.0 W/pound for solar dynamic and about 0.8 W/pound for nuclear reactor/Rankine cycle plants) the fuel cell is extremely attractive. However, as the cell is designed to operate at its most efficient current density and temperature, the weight increases considerably. That is, higher specific power can be achieved, but efficiency is decreased so that gains are offset by lower specific energy of the overall system.

1.4.2 Service Life

Fuel cells have been operated for several hundred hours, but contamination of the cell can cause severe deterioration and "poisoning" of the electrodes. Service lives much beyond one month should not be planned. Longer missions require use of some other system, until more is known about fuel cell reliability.

TABLE 1-7. FUEL CELL PERFORMANCE
SOURCE: Gene B. Stafford (Ref 1.2)

<u>Type</u>	<u>Year of Availability</u>	<u>Current Density (amp/ft²)</u>	<u>Specific</u>		<u>Specific Weight (lb/kw)</u>	<u>Specific Volume (ft³/kw)</u>
			<u>Reactant Consumption (lb/kw-hr)</u>	<u>Consumption (lb/kw-hr)</u>		
Gemini*	Present	40	0.95	172	4,560	
Gemini*, **	Present	100	1.22	87	2,320	
Apollo*	Present	85	0.95	225	16,100	
Apollo*, **	Present	195	1.07	112.5	8,050	
_____*	Present	70	0.95	68	2,930	
_____*, **	Present	150	1.05	40	1,710	
_____*	1968	150	1.00	35	1,500	
_____*	1972	150	0.85	30	1,300	

* Hydrogen oxygen ratio 1:8

** Applicable to missions of less than three days

Theoretically, a perfect fuel cell should last indefinitely, but considerable development remains before it can be used as the universal battery for all missions.

1.4.3 Efficiency and High Load Efficiency

Fuel cells are the most efficient chemical generator when operated within their design limits; however as current density (and temperature) increase, specific fuel consumption increases disproportionately. A secondary battery (such as AgZn) may be used to relieve intermittent operation at high loads in some systems. Research is expected to solve this problem eventually.

1.4.4 Reliability

So far, fuel cells have performed well in flight. However, failure could be catastrophic and more experience is needed with fuel cells to assess reliability. MTBFs are probably less than 1,000 hours. Theoretically, MTBF could extend to years with development.

1.4.5 Cryogenic Storage

Storage of LOX and LH₂ requires that cryogenic temperatures be maintained. This is accomplished by boil-off refrigeration with consequent loss of reactants as time increases. This factor alone limits service life of primary fuel cell systems. (Gemini experience indicates that the low temperatures sometimes limit fuel input to the cell, also.) Better insulation for tanks and achievement of optimum cryogenic storage can extend the service life of fuel cell systems by months.

This problem is also critical to lunar base fuel storage, and presently limits stay time without re-supply.

1.4.6 Reliability of Shutdown

The number of on-off cycles before failure is not available (to the writer), but is thought to present a problem. Residual reactants in the cell after shutdown may cause temperature rise through leakage currents with consequent damage to electrodes. The problem referred to here is that of reliability of start-up after extended shutdown.

1.4.7 Specific Cost

While no firm cost figures are available at present, Ref 1.2 states the specific cost at between \$30,000 and \$87,000 per kilowatt plus fuel. Widespread application of fuel cells would require reducing this by about a 10^{-2} factor.

1.5 NUCLEAR REACTOR

Wherever high power is required over long periods of time, the limitations of cost, weight and inefficiency of nuclear reactor systems begin to be less significant when compared with those of other systems.

Quoting Szego (Ref 1.4): "The absolute crossover between solar, chemical, and nuclear power depends upon a detailed evaluation of the specific mission. In this evaluation, the nuclear system offers definite advantages of ruggedness, high power per unit area, no collector deployment, no orientation, continuous power, minimum power storage requirement, etc. In many cases the added power availability of a nuclear system should offer significant operational flexibility and improved reliability through application of more conventional circuitry and instrumentation and through redundancy.

Despite the lack of specific plans for near term application of nuclear power, the SNAP (Systems for Nuclear Auxiliary Power) program has initiated the development and demonstration of a spectrum of units to fill future needs. The currently identified reactor systems under (1963) development are described in Tables" 1-8 and 1-9.

Quoting Stafford (Ref. 1.2): "The selection of a nuclear reactor is chiefly a function of application; but, operating temperature, fuel type, fission energy level, reactor geometry, and control mechanisms must...be considered.

Quoting Szego (Ref 1.4): "The basic cost of a SNAP 10A unit is estimated at less than 1 million dollars and the SNAP 2 unit should cost between 1 and 1.5 million dollars. An estimate of 3 to 5 million dollars for a SNAP 8 unit seems reasonable." (End quote from Ref 1.4)

1.5.1 Dynamic Conversion

Figure 1-6 and Table 1-10 (from Tonelli, et al Ref 1.5) compare the principle thermodynamic conversion cycles and heat sources for dynamic space power systems. Because it permits the lightest weight system and requires the least radiator area, the Rankine cycle is preferred for nuclear dynamic systems.

Quoting Szego (Ref 1.4): "The performance of the Hg Rankine cycles is limited by the practical working pressure limit and the thermodynamic properties of Hg. The major improvement in Rankine cycles occurs with the change in working fluid which unfortunately involves a discrete temperature step of about 315°C (600°F) to 426°C (800°F) to the SNAP 50 conditions. In the future, Hg systems will probably be used beyond their region of optimum size.

TABLE 1-8. REACTOR PERFORMANCE SUMMARY

	<u>SNAP 10A</u>	<u>SNAP 2</u>	<u>NASA SNAP 8</u>	<u>SPUR SNAP 50</u>
Electric - Power, Kwe	0.5	3	35*	350
Thermal - Reactor Power, Kwt*	30	50	600*	2,500
Efficiency, Percent	1.6	6	8	---
Inlet Temp (°F)*	900	1,000	1,100	---
Reactor Outlet Temp, °F *	1,000	1,200	1,300* (1,350)	2,000
Reactor/Moderator	U-ZrH _x Thermal	U-ZrH _x Thermal	U-ZrH _x Thermal	UC Fast
Primary Coolant	NaX-78	NaX-78	NaX-78	Lithium
Power Conversion	GeSi Thermoelectric	Hg Rankine	Hg Rankine	X Rankine
Boiling Temp, °F	---	930	1,070	---
Turbine Inlet Temp, °F	---	1,150	1,250	1,950
Condensing Temp, °F	---	600	650	---
Hot Junction Temp, °F	930	---	---	---
Cold Junction Temp, °F	615	---	---	---
Radiator Temp, °F	615	600	580*	---
Radiator Area, Ft ²	62.5	120	1400-1800*	
Ft ² /Kwe	125	40	45	
Reactor Wt (lbs) *	250	250	600	10 lb/Kw
System Unshielded Weight (lb)	650	1,200	4,200-4,950	5,000
U ²³⁵ Inventory (KG) *	4.3	4.3	7.0	---
Core Vol (Ft ³) *	0.3	0.3	0.6	---
Lb/Kwe	1,300	400	120-140	15
Available	1964	1966	1970-72	1975/80
Development Agency	AEC	AEC	AEC/NASA	AEC/AF
Flight Test Agency	AF	AF	NASA	AF
System Contractor	Atomics International	Atomics International	Aerojet General	Pratt & Whitney
Power Conversion Contractor	Radio Corp. of America	Thompson Ramo Wooldridge	Aerojet General	AiResearch
Reactor Contractor	Atomics International	Atomics International	Atomics International	Pratt & Whitney
Flight Test Contractor	Lockheed	Lockheed	---	---

* Nucleonics, Vol 21, No. 7, July 1963, p 79

* Ref 1.2

TABLE 1-9. SUMMARY CHART - NUCLEAR SYSTEMS
SOURCE: G. C. Szego (Ref 1.4)

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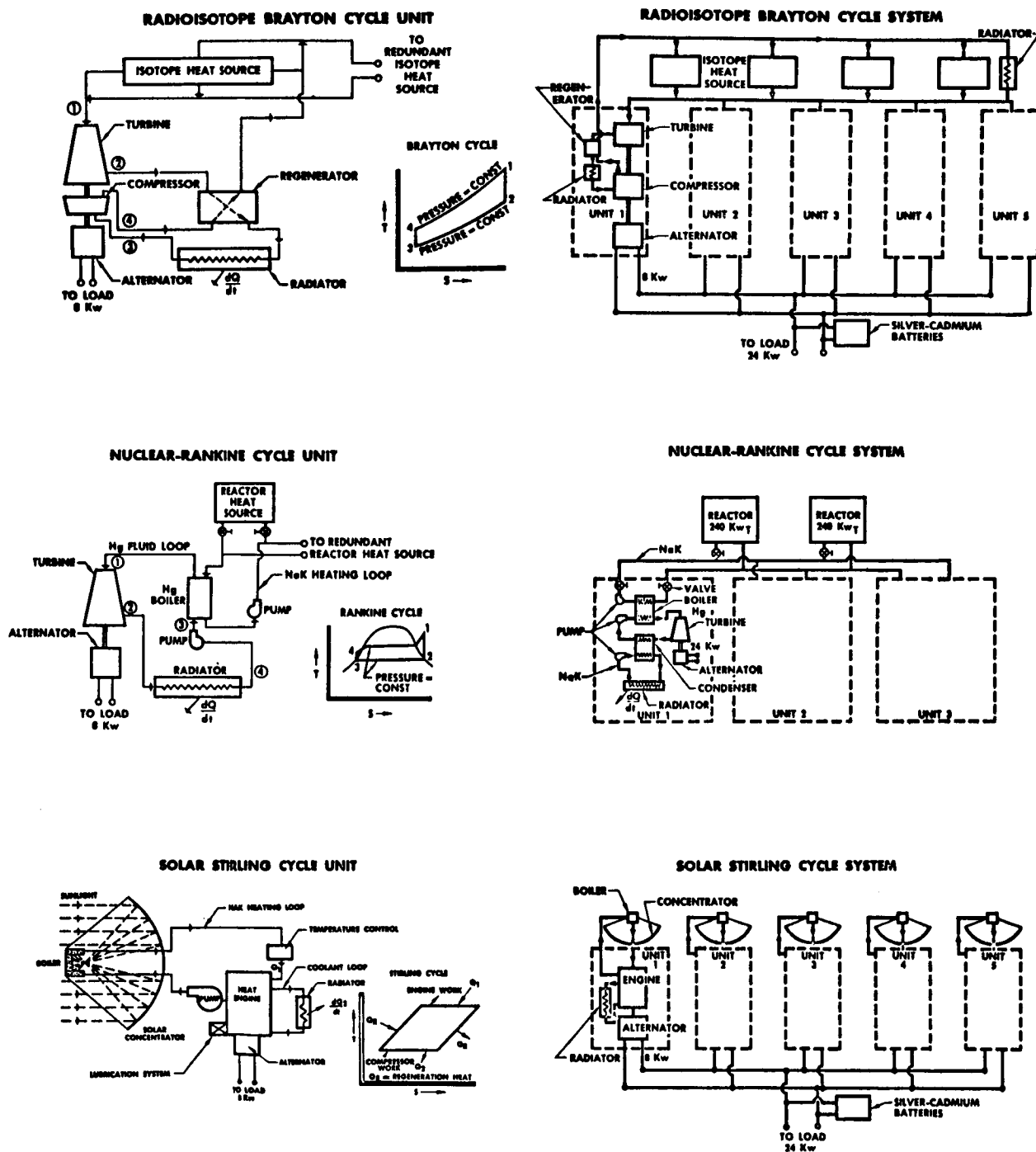


FIGURE 1-6. THERMODYNAMIC CONVERSION CYCLES
SOURCE: Tonelli, et al (Ref 1.5)

TABLE 1-10. DYNAMIC POWER CONVERSION CYCLE COMPARISON

Cycles And Heat Sources	Major Advantages	Major Disadvantages	Problem Area	Remarks
Brayton Cycle	Reduced mass transfer problems. Uses single phase fluid.	High radiator area.	Turbine-alternator gas bearing development.	Analysis comprehensive tests and close tolerance manufacturing required.
Rankine Cycle	Least radiator area. Allows lightest system weight	Requirement for boiling and condensing in "Zero G".	Two phase flow in "Zero G" environment. Erosion of materials. Liquid metal (Hg) lubricated bearings or rotating shaft mercury seal.	Boiling, condensing and liquid vapor separation in "Zero G" must be demonstrated. Materials research required for use of liquid metals at elevated temperatures. SNAP 8 mercury bearings replaced by lubricated ball bearing. Organic lubricant must be isolated from mercury.
Stirling Cycle	Simplicity. High cycle efficiency.	High radiator area.	Piston seal leakage. Engine lubrication in "Zero G" environment. Regenerator clogging due to contamination of working fluid with lubricant.	Research and development required to perfect dynamic seals. Lubrication for long durations must be demonstrated.
Isotopic	Does not require orientation system Operates in darkness	High cost, limited availability information.	Redundant isotopes require additional radiator area.	Limited research and development required.
Nuclear	Does not require orientation system. Operates in darkness. Lightest weight source.	Biological shielding required. Mirror susceptible to the space environment.	Development of compact high temperature reactors.	Parallel operation of reactors must be demonstrated.
Solar			Orientation control system requires a high degree of accuracy. Mirror storage volume is high and deployment in space must be demonstrated.	Thermal distortion, meteoroid penetration, film deterioration and "G" loads during mid-course corrections result in weight penalties

Source: Tonelli, A. D.; Blankenship, S. G. and Shinbrot, C. H. "Auxiliary Power Generating System for a Manned Mars Mission" presented to AIAA September 30 - October 1, 1963 at Palo Alto, California., (Ref 1.5)

because of the more immediate availability of the lower temperature technology. Thus, large Hg Rankine systems or multiple smaller systems may well be used in the several hundred kilowatt power range...The Hg Rankine cycle turbomachinery development for SNAP 2 has overcome the structural and thermal distortion problems that may have been limiting the reliability of the Hg lubricated bearings. Recent rotating machinery endurance accomplishments on SNAP 2 and Sunflower have demonstrated the potential engineering feasibility of a hermetic machine with working fluid lubrication.. In comparison to SNAP 2, which employs direct condensation of the Hg in a combination condenser-radiator, SNAP 8 now includes a compact condenser with a liquid metal heat transfer loop coupling the condenser to the radiator. In addition, SNAP 8 now includes an organic loop and radiator to cool the lubricant. The immediate SNAP 8 turbomachinery development will reveal the practicability of attempting to apply conventional conversion machinery technology to space power. The power conversion for SNAP 50 should be considered to be in the research phase. Considerable research has been initiated by several agencies in the problems relating to alkali metal power conversion systems. The effort must certainly be considered as dominated and paced by high temperature (refractory metals) materials and fabrication technology." (End quote from Ref 1. 4)

Table 1-11 (from Ref 1. 5) shows the comparative characteristics of seven different 24 kw dynamic systems hypothesized for a manned Mars mission (1980-85).

Research and Development Required. The present status of nuclear reactor/Rankine systems for space must be regarded as in early development. Major expenditures are required to bring these devices to the status of operational hardware. About 6 to 8 years of lead time are needed between a definitive requirement and hardware delivery. A definite need exists for a 100 KWe nuclear reactor system capable of being landed on the moon by a single Saturn V shot. This will be needed in the 1974 to 1976 period, hence serious development should start in 1964 to 1966.

High Temperature Materials. The key to improved performance (watts/Kg or watts/cubic meter) of all space power systems including the SNAP units is heat source temperature. In general, the performance advancements will be paced by the availability of the materials technology required by a given operating temperature. In the reactor, temperature and fuel material selection will determine the useful power output of the reactor before failure due to fission product induced fuel swelling. In the remainder of the system, temperature and material selection will limit life due to corrosion, creep, sublimation, etc. It is clear that temperature is the key to improved performance but is detrimental to system life and reliability.

TABLE 1-11
(Estimated 1980 Technology)

PARAMETER	NUCLEAR RANKINE	ISOTOPE RANKINE	ISOTOPE BRAYTON	ISOTOPE STIRLING	SOLAR RANKINE	SOLAR BRAYTON	SOLAR STIRLING
HEAT SOURCE WEIGHT, LBS	530	390	448	380	291A	315A	239A
POWER CONVERSION SYSTEM WEIGHT, LBS	560	560	294	555	560	313	555
RADIATOR WEIGHT, LBS	149	90D 149	90D 458	90D 374	149	442	375
TOTAL WEIGHT PER 8 KW MODULE, LBS	1239	1099	1227	1309	1000	1070	1169
ELECTRICAL TRANSMISSION WEIGHT (LBS)	120	120	120	120	25	25	25
BATTERY WEIGHT, LBS	735	735	735	735	335B 530C	335B 530C	335B 530C
TOTAL POWER SYSTEM WEIGHT, LBS	5460	6050	6637	7150	5370B 6555C	5710B 6975C	6205B 7569C
TOTAL RADIATOR AREA, FT ²	930	960	3230	2705	930	2946	2680
CONCENTRATOR AREA PER 8 KW MODULE, FT ²	-	-	-	-	1240	1400	1000
TOTAL CONCEN- TRATOR AREA, FT ²	-	-	-	-	6200B	7000B	5000B

NOTES: A INCLUDES 15 LB FOR ORIENTATION. C MARS EQUATORIAL ORBIT.
B MARS POLAR ORBIT. D SECONDARY RADIATOR TO COOL REDUNDANT ISOTOPE HEAT SOURCE.

Estimated characteristics for seven different 24 kw dynamic systems for a hypothetical manned Mars mission. (Copied from Tonelli, et al, page 81 of AIAA "Engineering Problems of Manned Interplanetary Exploration", October 1, 1963, Palo Alto, California. (Ref 1.5)

Operation in the 1600 to 1700°R region is desirable. Suitable refractory materials are available, but higher temperature operation requires increasing the weight of material to compensate for reduced strength. Still needed are extremely light materials capable of strengths around 60, 000 to 70, 000 psi at temperatures in the region from 1600 to 2000°F (red hot to white hot). Most metals become quite plastic and tend to anneal at these temperatures; many of those which do not soften at these temperatures (tungsten, platinum-iridium, osmium) are quite expensive, or are unsuitable for other reasons. It is conceivable that alloys, ceramics, or even plastics may be discovered having high strength at high temperatures, while meeting other requirements for turbine blades, pipes, heat exchangers, etc. However, it also is conceivable that such materials are impossible (or too costly) and that high temperature operation will always imply a weight penalty.

Shielding Weight. Quoting Szego (Ref 1.4): The shield weight is a stronger function of the mission and allowable integration configuration than it is of the power unit. In the case of payloads comprised of semi-conductor devices, dose levels below 10^{11} nvt and 10^6 R probably require very minor restrictions on component selection. Payload hardening for 10^{12} nvt and 10^7 R can be accommodated if properly considered from the outset of payload design. The shield weight for simple conical shadow shield geometry can be in the region of 200-500 pounds for SNAPs 10A, 2, and 8. In the case of manned applications, the shield weight varies from 4 to 7, 000 pounds for a simple conical shadow shield configuration of a small (10-foot diameter) space station to 15 to 20, 000 pounds for a large (150-foot diameter) toroidal station" (End quote from Ref 1.4).

Waste Heat Rejection. At the higher power levels required for future space missions it becomes increasingly important to reject waste heat at high temperature to minimize radiator weight. (See also paragraph 1.2.1, "Solar Dynamic".)

Wearout, Rotating Machinery. Refer to paragraph 1.2.3, "Solar Dynamic".

Reliability, Nuclear Reactor. Quoting Szego (Ref 1.4): "The high energy density advantage of nuclear heat sources directly implies that long life is a necessary requirement to achieve the full advantage of nuclear power systems. The only unique, self-imposed environment that could influence reliability is radiation. The more important influences of high temperature, corrosion, creep, high vacuum, micrometeors, etc., are shared by other approaches to high performance space power. The major problem becomes apparent when one considers the unreasonable time and cost associated with a statistical demonstration of reliability or with the corollary identification of failure modes with confidence. This basic dilemma is shared by

many other aspects of the space program. The ultimate solution must rely upon simplicity, basic phenomenological understanding, and sound engineering. Basic system development must progress to a level which allows a valid judgment of inherent reliability and considerable experience will have to come from interim usage." (SNAP 10A failed after 43 days in orbit due to unknown causes:) probably automatic shut down was initiated by electrical noise--Ed.).

1.5.2 Thermionic Conversion

"The thermionic converter is an electron tube which converts heat directly into electrical power by thermionic emission. The simple cesium vapor diode, with elemental metal electrodes, continues to be the most efficient and powerful form of converter at all practical temperatures. Cylindrical electrodes provide the most efficient configuration for internally heated converters in a nuclear reactor, but planar electrodes have been used predominantly in other applications with external heat sources. A typical configuration is shown schematically in Figure" 1-7....

"Thermionic converters can presently be purchased in quantity, meeting ± 10 percent output specification, with powers, efficiencies and lifetime suitable for many practical applications. (See Table 1-12.) Research continues to play a major role in defining the material and processing variables which might lead to still greater performance and stability." (End quote from Ref 1.4) Table 1-13 recaps some current thermionic system developments.

Quoting Stafford (Ref 1.2): "Thermionic energy conversion appears attractive for space power generation because of light weight, lack of moving parts, good efficiency and because it is a high temperature device, area requirements for waste heat disposal are small. Another feature of thermionic diodes is the applicability of the diode to any of the three" heat "sources considered".

Quoting G. C. Szego (Ref 1.4): "The maximum performance system of the future (2.3 to 4.6 Kg/kw) is the reactor thermionic system. Even though this concept has been called SNAP 70, no system development program has been established. Two system concepts are being advanced. One system employs diodes outside the reactor, perhaps in the radiator. This concept forces the entire reactor structure, control, coolant pumps, etc., to operate in the 1,090 to 1,370°C (2,000 to 2,500°F) region. The other approach places the diodes directly in the core as an integral subassembly of the fuel element. The nuclear material operates at the cathode temperature of the space charge neutralized diode of about 1,650°C (3,000°F), while the remaining

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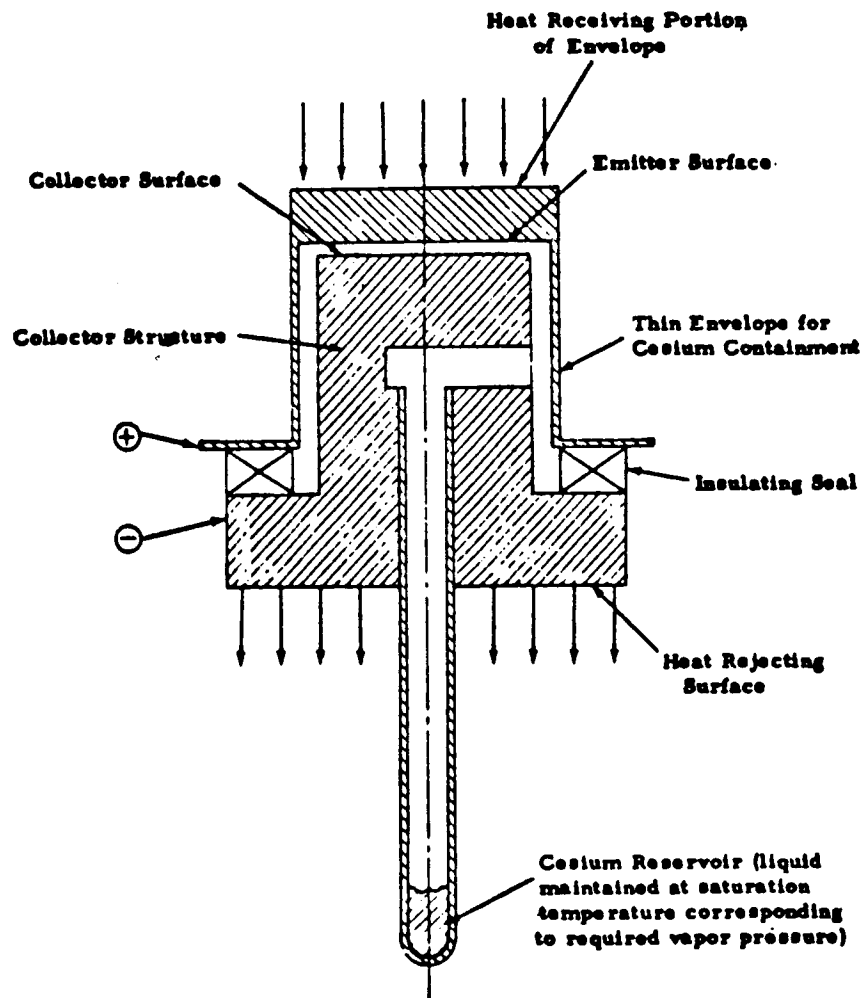


FIGURE 1-7. BASIC THERMIONIC CONVERTER STRUCTURE

SOURCE: G. C. Szego (Ref 1.4)

TABLE 4-12. PRESENT THERMIONIC DIODE PERFORMANCE

SOURCE: Gene B. Stafford (Ref 1.2)

Converter	Electrodes	Emitter Area (CM ²)	Spacing (MILS)	Emitter Temp (°C)	Power Density (W/CM ²)	Efficiency (%)	Life (Hour)	Remarks
Martin 1-K	Mo-NB	---	20.	1500	3.0	---	9600	(5/65)
RCA A-1272	Mo-NI	40.	5.0	1800	10.0	19.8	2600	Life based on MTBF for 12 Convts. (10/64)
RCA A-1274	Mo-NI	59.3	10.	1450	7.0	---	---	3 Convts. Mod. (1/65)
RCA A-1274	Mo-NI	59.3	10.	1300	1.5	---	800+	3 Convts. Mod. (3/65)
RCA A-1274	Mo-NI	59.3	10.	1200	1.0	6.0	---	3 Convts. Mod. (3/65)
TEECO 402-RL-1	RE-Mo	2.0	20.	1700	13.7	---	3200	Solar Diode (5/65)
RCA A-1198B	Mo-NI	60.	20.	1300	1.5	6.0	4462	Insulated Emitter (7/64)
GE (IN-PILE)	W-SS	8.6	10.	1730	6.0	10.3	224	Last Unclass'd data on In-pile tests (6/63)
SNAP-13 88-1*	---	---	---	---	6.0**	---	6400	Low power due to low collector temp.
SNAP-13 88-1*	---	---	---	1370	12.5**	6.5	8560	New Heater
SNAP-13 88-1*	---	---	---	1460	20.**	---	11000	Still operating (5/65)
TEECO	RE-Mo	2.0	2.15	1700	213	11.	---	Solar diodes (10/64)
Martin	W-Mo	---	40	1730	8.5	---	---	BA-CS Convts. (10/64)
Martin IDDM-1	M-N	40.8	10	1550	6.0	10	2000	2 Convts UO ₂ Fueled (4/65)

* Single Test

** Total Power In Watts

TABLE 1-13. THERMIONIC SPACE POWER SYSTEMS DEVELOPMENT
SOURCE: G. C. Szego (Ref 1.4)

Type of System	Organization	Sponsor	Design Objectives	Status	Availability
ISOTOPE SNAP-13	Thermo Electron Martin-Marietta	E	Meet Surveyor require- ments: 25 watts at 1 volt for 130 days	Power objectives met. Generators are 3.3 watts/lb. including fuel Those on electrically heated life test still operable (other data classified. Martin Co. reports MND-P-2890- 1,2)	Fueled within 1 year. Available within 2 years.
NUCLEAR In-pile Concept (SNAP-70)	Atomics Interna- tional Babcock & Wilcox General Atomic General Electric Los Alamos Martin Marietta Pratt & Whitney RCA Thermo Electron	A, E, G, J, K	0.3 to 10 MW for 1 year. Estimates in- dicate unshielded system weights from 4 to 10 lb/kw, for fuel temperature of 1500 to 1900°C	Basic feasibility still uncertain from fuel & other materials stand- points. Various elec- trical & inpile tests of fueled converters have but short lifetimes (<300 hrs.)	Unknown, but variously esti- mated at 5 to 10 years.
Pile-Surface Concept (STAR-R)	General Electric	--	10-40 kw. Estimates in- dicate unshielded system weights from 20 to 65 lb/kw for fuel temper- atures of 1900 to 2300°C	Feasibility still uncer- tain. No fueled conver- ter tests under assumed conditions.	Requirement uncertain. Dup- licates power range of SNAP-8
Thermionic Radiator Con- cept	Aerojet Nuclear RCA	--	0.3 to 10 MW. Esti- mates indicate un- shielded specific weights from 10 to 20/lb kw for fuel and coolant temperatures of 1200 to 1500°C	Feasibility still uncer- tain from coolant, fuel, and other materials standpoints. Lithium heated converter dem- onstrated for short time, (AGN-RCA). P&W also planning converter tests.	Unknown (3-7 years)
SOLAR TEE-100 Watt	Thermo Electron TRW	A	100 watts at 1700°C using 5-foot concen- trator	5 diode generator. Elec- trically tested; 120 watts at 1700°C. Solar tested: 19 watts at 1380°C.	Unknown
STEPS II	General Electric	A	Demonstrate sys- tem feasibility 5- foot concentrator	3 diode generator. Solar tested: 27.9 watts at 1670°C	Unknown
SET I	Thermo Electron EOS	H	135 watts at 1700°C using 5 foot concen- trator; 3000 hrs.	SET Ia (5 diode gener- ator) Solar tested: 41 watts at 1700°C approx. SET Ib Electrically tested: 82.5 watts at 1700°C JPL will solar test Specific power of gen- erator = 24 watts/lb Specific power of com- plete system = 7.8 watts/lb.	Flight test within 2 years (est.) Available within 4 years

reactor structure, control, coolant, pumps, etc., operate at the anode temperature of about 815°C (1,500°F).

R&D Required. There have been many independent and Government-sponsored research efforts in the field of thermionic conversion. The basic phenomenon is reasonably well understood as evidenced by demonstrated conversion efficiency and power density accomplishments. The problems of materials selection for the environment and for useful diode life are less well understood. The technology necessary for the selection of a reactor fuel material with appropriate physical properties and high energy output capability is currently far from the status required to support serious system design and development." (End of quote from Ref 1.4)

It should be mentioned that many of the diodes are experimental since prior emphasis has been on developing the diodes to a point where flight hardware could be designed and built.

The continuing research is being directed toward increased life and better definition of the material and process variables. Work is also continuing on the supporting areas of insulation materials and integration of the diodes into the hostile environments to which they will be subjected, that is, in contact with an isotope or in the core of a reactor.

The results of the work on the thermionics should yield diodes of approximately 20 percent efficiency, 50 to 60 watt/cm² flux densities and lives of more than 10,000 hours for about 1/2 pound per diode.

Reliability. No system reliability estimates are yet meaningful, however MTBF's in excess of 5,000 hours are anticipated for nuclear reactor thermionic systems. Elimination of rotating machinery is a very significant consideration.

Shielding Weight. Refer to paragraph 1.5.1.

Waste Heat Rejection. Refer to paragraphs 1.5.1 and 1.2.1.

Thermionic systems have an advantage over dynamic systems in that waste heat can be rejected at a much higher temperature, with consequent saving in radiator area. Outlet temperatures may go as high as 2,500°F eventually.

Electrical Noise. In common with other vacuum tube devices, thermionic diodes will generate thermal noise along with dc. This requires fairly heavy, low-pass filter networks in the power conditioning section to prevent interference with load devices. This noise problem is less severe than that with faulty generator brushes.

High Temperature Materials. Refer to 4th paragraph under 1.5.2

1.5.3 Thermoelectric Conversion. The SNAP 10A system employs silicon-germanium alloy materials for thermoelectric direct power conversion. The lower figure of merit of the SiGe alloys was accepted in order to achieve better fabricability and higher temperature capability than the more familiar PbTe. While any dissimilar conductive materials might produce a potential difference as a result of a thermal gradient across their junction, the semiconductors have so far proved to be the most efficient. Efficiencies between 4.5 and 8 percent compare favorably with those from Rankine cycle conversion, and have the additional advantage of eliminating rotating machinery. It is evident that one salient feature of the energy conversion process is its adaptability to any thermal energy source.

Quoting Stafford (Ref 1.2): "A great number of materials have been investigated for thermoelectric energy conversion but in general the materials may be categorized as low temperature, medium temperature, or high temperature. The selection of the material for a particular application is a function of the operating temperature which can be obtained from the energy source. Table 1-13 presents the present and projected performance of the three classes of thermoelectric materials.

"It is possible to obtain better subsystem performance from operation of thermoelectric devices over a greater temperature difference but materials tend to decrease in performance as the cold junction temperature decreases beyond some characteristic value. Consequently, investigations have been undertaken to consider segmenting the couples. Essentially the technique consists of two or more materials bonded together in such a way that a higher temperature material rejects heat directly to a lower temperature material. In this manner a greater overall temperature differential can be obtained without utilizing the the thermoelectric materials out of their best temperature regime."

R&D Required.

Thermoelectric Materials. "To achieve the projected performance shown in Table 1-14, effort must be expended in the areas of improved thermoelectric materials, particularly the high temperature materials, better bonding of the elements with the hot and cold junction, and better overall fabrication techniques." (End quote from Ref 1.2).

TABLE 1-14. THERMOELECTRIC CONVERTER PERFORMANCE
SOURCE: Gene B. Stafford (Ref 1.2)

Availability	Present				Projected (Eventual)			
	Bi-Te	Pb-Te	Si-Ge	MCC50	Bi-Te	Pb-Te	Si-Ge	MCC50
Material								
Hot Junction Temp (°C)	250	450	850	1,200	250	450	950	1,200
Cold Junction Temp (°C)	80	180	350	700	80	180	350	700
Module Efficiency (%)	4.5	5.0	6.5	4.0	5.0	7.0	8.0	8
Mod. Specific Power (w/lb)	15-20	15-20	15-20	10-15	15-20	30	40	40
Output Power Flux (w/ft ²)	2.6-4.	--	--	--	6.5	--	--	--
Mod. Specific Cost (\$/watt)	70	60	100	100	15-20	20-30	10-20	10-20

Reliability. In reviewing the state of the art of auxiliary power systems for space applications, it is most significant that the radioisotope fueled thermoelectric generator on Transit 4A, launched on 29 June 1961, was still operating satisfactorily after over two years in orbit. No other direct thermal energy conversion technique has matched this record for sustained performance in space. Reliability of the reactor is essentially the same, whatever the conversion process, and is largely dependent upon the reliability of the reactor control element servo-systems.

Waste Heat Rejection. See paragraphs 1.5.1 and 1.2.1.

Specific Power. Nuclear reactor thermoelectric systems are quite heavy for the power available. SNAP 10A yielded 0.8 watt per pound (0.00178 Kw/Kg) unshielded. This is largely due to its low efficiency (about 1.7 percent). R&D in high temperature materials and junction semiconductors may be able to improve the specific power by a factor of five or more by the mid-1970s.

Shielding Weight. Refer to paragraph 1.5.1.

1.6 ISOTOPE/THERMOELECTRIC

These systems use thermoelectric junction conversion and a radioactive isotope as a heat source.

Quoting Stafford (Ref 1. 2): "Although selection of an isotope as an energy source for a power subsystem is dependent upon many considerations, among the most important are the isotope half-life, power density, operation temperature limitations, availability (resources), and cost. With the aforementioned criteria, relatively few isotopes can be considered. Table 1-15 summarizes the performance characteristics of the isotopes of interest...

"The power densities of the isotopes shown in Table 1-16 are for the raw isotope. By the time the isotope is compounded and encapsulated, the usable power densities may be down by as much as an order of magnitude.

Improvements in isotopic heat sources will not come about through improving the isotopes themselves since the physical properties on the nuclides are fixed. Rather, the improvements will come through improved compounding, increased production, and better containment techniques." (End quote from Ref 1. 2).

Table 1-15 recaps some of the earlier isotope/thermoelectric systems. Snap 10A is listed for comparison.

Quoting Szego (Ref 1. 4): "Because of the improvement in power density of radioisotopic thermoelectric generators (as compared with solar cells) for some future space missions, the Radio Corporation of America is currently developing inhouse a prototype radioisotope generator in the 50 to 75 watt output range. The unit was scheduled for completion in late 1963 or early 1964 and is proposed to have all the characteristics of a flight system except for containment and safety demonstration of the flight capsule.

"Several programs are under way to advance the technology of radioisotope thermoelectric power; for example: (1) Sr-90 fuel space power system for communication satellites, (2) 200 to 500 watt systems for such missions as Voyager, Nimbus, etc., and (3) 3 to 5 kw systems for use in manned space stations.

"These studies have advanced to the point where feasibility of such concepts has been proven. In addition, advanced thermionic systems (such as SNAP 13) are being examined to show the feasibility of power supplies up to 500 watts for space application."

TABLE 1-15. RADIOISOTOPE HEAT SOURCE MATERIALS SUMMARY
(Compiled from various sources - CWB)

Isotopes	Strontium-90	Polonium-210	Cerium-144	Promethium-147	Curium-244	Cesium-137	Plutonium-238	Curium-242	Cobalt-60
Half-Life, Years	28	0.38	0.78	2.5	18	30	86.4	0.45	5.24
Compound Form	Sr TiO ₃	Metal	CeO ₂	Pm ₂ O ₃	Cm ₂ O ₃	Glass			
Compound Density gm/cc	4.6	9.3			11.75		10	11.75	8.9
Power Density Watts/cc	1.05-1.16*	1,210	24.5	2.22	27	0.215	3.9	1,150	15.5
Compound									
Decay Product	Beta, X	Alpha	Beta, Gamma	Beta	Alpha, Neutron	Beta, Gamma	Alpha	Alpha, Neutron	Beta, Gamma
Shielding Requirements	Heavy	Minor	Heavy	Minor	Minor	Heavy	Minor	Minor	Heavy
Useful Mission Life (Yr)	10	0.5		2.5	10	10			
Estimated Cost, \$/gm (pure)	43	90,000	32	68	4,300	26			
Estimated Cost, \$/watt	45 to 80*	640	1.50	168	980 to 1,600*	63	3,700	17	100
Estimated Availability Considering Sufficient Lead Time for Plant Facilities by 1980 Thermal Kilowatts/years	200 to 693*	14	4,900	181	204 to 300*	458	50	120	100
Estimated Power Capability by 1980 in Electrical kw/year based on 20% Conversion Efficiency	138	2.8	980	36	40	92			
Maximum Temperature °C	2,400	2,000			2,200		2,600	2,200	1,450

* Sources of information disagree.

TABLE 1-16. SNAP THERMOELECTRIC SYSTEMS

SOURCE: G. C. Szego (Ref 1.4)

<u>Ident.</u>	<u>Electrical Output (Watts)</u>	<u>Design Life (years)</u>	<u>Fuel</u>	<u>Amount of Fuel (K-Curies)</u>	<u>Total System Weight (lbs)</u>		<u>Temperature TH(°F) TC(°F)</u>		<u>Efficiency (percent)</u>	<u>Remarks</u>
SNAP 3	4.		Po-210	2.1	4.	4.	620	115	5.8	Transit
SNAP 3	2.7		Pu-238	1.6	4.6	4.6	--	--	5.2	Transit
SNAP 9A	25.	5	Pu-238	--	27.	27.	--	--	--	Navy Navig. Satellite
SNAP 11	25.	1/3	Cm-242	25.	30.	30.	--	--	4.8	Surveyor
SNAP 10A	500.	--	Reactor	--	--	--	930	630	1.7	--

"Development effort to date shows that high reliability, compact, isotopic power supplies in the 1 to 4 watt/pound range are feasible. These units can take the rigors of launch and continuing operation for extended periods of time in near and deep space environments. By virtue of more than one million hours of SNAP generator operation using Po-210, Pu-238, and Sr-90 isotopic heat sources, it has been shown that such system can provide reliable power for suitable space missions. Of particular applicability are:

1. Earth orbiting missions of a year or more duration.
2. Lunar surface exploration probes where dust and terrain uncertainties or lack of solar energy would dictate an unoriented, self-energized power supply.
3. Deep space probes toward or away from the sun, where temperature levels or lack of solar energy favor use of isotopic power units.
4. Higher power levels in the range of 1 to 6 kw where lower launching weights are achievable with isotopic power supplies." (End of quotes from Ref 1.4).

1.6.1 Isotope Resources (Availability)

Note on Table 1-14 that one of the better isotopes, Plutonium 238, has a projected availability of only 50 kw per year at an estimated cost of 3.7 million dollars per kw. Those yielding high power density (Po210 and Curium 242) have half lives less than one year. Curium 242 is the lowest in cost (and highest in availability), but its half life makes systems based upon it hardly competitive with fuel cells. Sr 90, the isotope with the lowest power density, has a reasonable half life and cost, but much more of it is required and heavier shielding is necessary.

Whichever isotope is chosen, the limited quantity available from known resources must be kept in mind.

It is conceivable that space exploration will disclose vast new resources of this type, but most likely they will have to come from transmutation reactions as a by-product of power generation.

1.6.2 Radiation Damage, Semiconductors

Referring to Table 1-14, note that some of the isotopes will emit high energy photons (Cobalt 60 emits gamma rays) and some emit neutrons. Shielding against these emissions involves a weight penalty, while

unshielded operation involves a risk of destructive damage by unwanted ionization (and transmutation doping) in semiconductor electronics located near the heat source. Numerical data is not available at this writing to illustrate the degree of deterioration, but in general designs to minimize radiation damage will involve a payload weight penalty. Plutonium 238 sources require little shielding, but availability of the isotope becomes a problem.

1.6.3 Temperature Limited (Efficiency)

Higher temperature operation is desirable from the viewpoint of efficiency, as well as heat rejection. Isotope sources can provide maximum temperatures ranging from 1450°C to 2600 °C without serious fuel element damage. The need is for development of conversion electronics to take advantage of this available energy.

1.6.4 Waste Heat Rejection

(See paragraphs 1.5.1 and 1.2.1.)

The problem is similar to that for solar, dynamic, and nuclear reactor/rankine systems.

1.6.5 Specific Energy (kwh/kg)

When long half-life isotopes are used, this parameter becomes so large that the system must be regarded as power limited rather than energy limited. However, with half lives in the 5 to 8 month region, specific energies in the region of 3,600 to 6,000 watt hours per pound (around 80 kwh/kg) begin to point up the value of isotope sources versus batteries and fuel cells.

In general, this consideration is similar to that of isotope resources availability (paragraph 1.6.1).

1.6.6 Specific Power (kw/kg)

SNAP 3 (Po-210) yields about one watt per pound of system weight. SNAP 3 (Pu-238) yields about 1.7 w/lb (Table 1-15). SNAP 11 also yields about 1 w/lb. In metric units, this is about 2.2 w/kg or 0.0022 kw/kg to 0.0038 kw/kg. This compares favorably with nuclear reactor/thermoelectric systems, but not with thermionic systems.

1.6.7 Shielding Weight

Refer to paragraph 1.5.1.

1.6.8 R&D Required, TE Conversion

Refer to paragraph 1.6.3. Improved conversion efficiency, higher temperature operation, higher reliability and lighter weight are needed to take better advantage of the available thermal energy.

1.6.9 Reliability

MTBFs in excess of 5 years are desired. Life tests upon many units is impractical, cost and timewise.

1.7 ELECTRONIC POWER CONDITIONING (EPC)(VOLTAGE/FREQUENCY/ PHASE CONVERSION)

The term power conditioning is self-defining; however, in practice the power conditioning system performs more functions than the name implies. Some of the key functions are listed as follows:

Accept prime power with all its characteristics and transform, convert, invert, rectify, regulate, filter, limit, etc., in order to satisfy all user subsystem requirements.

Respond to external commands by applying power, removing power, changing characteristics, etc.

Operate in an open or closed loop control system in response to external control functions.

Take corrective or protective action in the event of malfunction which occurs internally or in associate subsystems.

Table 1-17 indicates some of the probable characteristics of initial power conditioning systems for space vehicles.

1.7.1 Reliability

Power conditioning equipment must necessarily interface with all other subsystems. The power conditioning system is thus constrained by interface requirements with almost every other subsystem and hence figures most strongly in system evaluations from a standpoint of reliability, efficiency, weight, and general probability of mission success.

In order to minimize unnecessary power conditioning equipment there is need for early consideration in the design cycle. The problem is not one of designing power conditioning to match interfaces that are already established, but to interject power conditioning technology into the system.

TABLE 1-17. FUTURE POWER CONDITIONING
EQUIPMENT ESTIMATES

SOURCE: G. C. Szego (Ref 1.4)

(data from the General Electric Co.)

Output 28 V (Mil Spec 704)

<u>DC to DC</u>		<u>lb/kw</u>	<u>Full Load Efficiency</u>	<u>in³/kw</u>
25 Watts	1 Volt Input, 100% Voltage Regulation	28	0.85	360
	9 Volt Input, 100% Voltage Regulation	18	0.90	240
	20 Volt Input, 100% Voltage Regulation	8*	0.90	100
150 Watts	6 Volt Input, 100% Voltage Regulation	18	0.90	240
	12 Volt Input, 100% Voltage Regulation	12	0.92	160
	20 Volt Input, 40% Voltage Regulation	6	0.94	85
1,000 Watts	12 Volt Input, 100% Voltage Regulation	11	0.91	140
	20 Volt Input, 40% Voltage Regulation	5.5	0.93	75
10 kw	20 Volt Input, 40% Voltage Regulation	5	0.92	70

* Assumes eventual availability of high-speed, switching transistors not now in hand.

DC to AC

Weights and Volumes > Twice those above

Efficiency about the same, i. e., 0.90+.

DC-AC INVERTERS

(data from Westinghouse Electric Corporation)

3 ph, 115 v, 400 cps, fully regulated output

<u>Application</u>	<u>Power Out, KVA</u>	<u>V_{in}, vdc</u>	<u>lb/kw</u>	<u>F. L. Eff., %</u>	<u>in³/kw</u>
Apollo	1.25	25-31	30	80	1,000
Centaur	0.6	25-31	42	77	1,000
Lunar Base	10.0	50-60	16	90	420
MOSS	1.0	50-60	25	81	1,000
DC-DC Converter (42 vdc, current limited, output)					
Saturn	1.0	50-60	11	90	180

Although an ac prime power source is versatile and simplifies the power conditioning system, the majority of applications today are based on the use of low voltage dc originating from solar cells, thermoelectric converters and batteries.

The Ranger spacecraft power system is one example of a design which utilizes solar panels, batteries, distribution in the dc phase, and individual conversion at the location of the using device.

1.7.2 Radiation Damage, Semiconductors

Refer to paragraph 1.6.2.

1.7.3 Transmission Lines

The various nuclear systems will often require locating the reactor remotely from the load system for safety of personnel. In such cases, a power transmission line is required. An example is the lunar colony where reactors may actually be located over the horizon or in a crater for shielding. This remote operation leads to some special considerations:

Arcing in Vacuum. In vacuum or low pressure environments, arcing or ionization conduction may be initiated whenever thermionic or photoelectric emission occurs from exposed conductors. This can cause noise as well as power loss.

Electrical Noise. Long lines carrying ac tend to radiate electromagnetically and can induce hum in nearby electronic equipment. Transformers can cause inductively coupled noise in nearby closed circuits, and any arcing or corona discharge can cause a "frying" noise in radio equipment.

Efficiency, Long Lines. Where low voltage dc is transmitted over any considerable distance, a significant I^2R loss must be allowed for. This can be overcome by transforming to high voltage ac at the generator, then back to usable voltages at the load, but the added efficiency is purchased in terms of extra equipment weight.

Reliability. Properly designed transmission cables can operate for many years unattended, but mechanical damage due to vehicle tracks, seismic motion or meteoritic erosion may limit cable life in planetary surface systems. The extreme thermal cycling on the moon's surface must also be considered as presenting a dielectric materials reliability problem.

REFERENCE LIST FOR SECTION 1

- 1.1 Space Aeronautics, June 1956, page 18 (Editorial comments).
- 1.2 Gene B. Stafford, "Space Power Subsystem Capabilities," AIAA paper No. 65-468 presented at San Francisco, July 26-29, 1965.
- 1.3 "Aerospace Technical Forecasts, 1962-1972 and 1960-1970," Aerospace Industries Association of America, Inc., 1725 DeSales Street N. W., Washington 6, D. C.
- 1.4 George C. Szego, "Space Power Subsystems State of the Art," AIAA paper No. 64-525 presented at Washington, D. C., June 29 - July 2, 1964.
- 1.5 A. D. Tonelli, et al, "Auxiliary Power Generating System for a Manned Mars Mission," presented to AIAA at Palo Alto, California, September 30 - October 1, 1963.

Section 2

DATA PROCESSING

2.0 General Comments

For this study data processing encompasses all equipment, and functions needed to receive, store, process and condition signals in any form, including the integrated data display to be used for human or machine decision making. These data may be within the basic system or transferred to external systems. Data processing of task instrumentation signals is included in this section, but data processing for navigation and for stabilization and control are discussed in their respective sections.

In general, the computer state of the art is somewhat more advanced than that of other subsystems, however the study has disclosed some significant technological deficiencies in the light of the mission requirements found, and these are discussed briefly under keywords (as used in the concept sheets of Volume II).

2.1 DIGITAL COMPUTERS

In general, computers are size-rated in terms of clock frequency and number of binary bits (or number of words of a prescribed word length in bits) of memory storage capacity.

Some computers may have variable frequency clocking through programmed count-down from the master clock as a function of mission time. For example, data handling requirements may be as little as 10 to 100 computational cycles per day during midcourse on a planetary fly by, then abruptly rise to 100,000 per second during encounter.

The trend toward higher and higher clock rates has been possible due to the use of ever smaller electronic components. This also has permitted increasing the arithmetic operation rates (multiply, divide, add, etc) through use of parallel logic and multiple arithmetic units.

A typical aerospace computer has a clock frequency of 500 kilocycles, an add time of 2 to 5 microseconds and a memory capacity of 4,096 16-bit words. It would occupy about 0.05 cubic meters, weigh about 10 kilograms and draw about 80 watts running power. Compact computers with clock frequencies of one megacycle and memory capacity of over 16,000 words of 36 bits each are within reach of the present state of the art. Such a computer would compare functionally with the large-scale, ground-based,

scientific computers of the early 1960's, but would not include the printers, typewriters, card readers and multiple tape storage abilities of such machines.

Progress in packaging such functionally large machines in physically small units has been spectacular as a result of progress in solid state electronics and the end is not in sight. It is expected that aerospace computers of the mid-1970's might weigh less than a kilogram, occupy a space of less than 5,000 cubic centimeters, have a clock frequency of one to five megacycles, and a memory capacity in excess of 32 thousand 36 bit words, if the present promises of thin film and integrated circuitry electronics are fulfilled. Such computers should have MTBFs in excess of 10,000 hours.

Figure 2-1 shows the expected growth in electronic packaging density. Figure 2-2 shows the expected usage trend for all aerospace electronics.

Figure 2-3 shows the expected trend of some important computer parameters referred to 1962 state of the art as the standard. Note that a reduction of two orders of magnitude is expected in weight, size and power by 1972. Speed is expected to increase by a factor of 10. Figure 2-4 shows a similar increase in capacity and speed for ground based computers.

2.1.1 Reliability

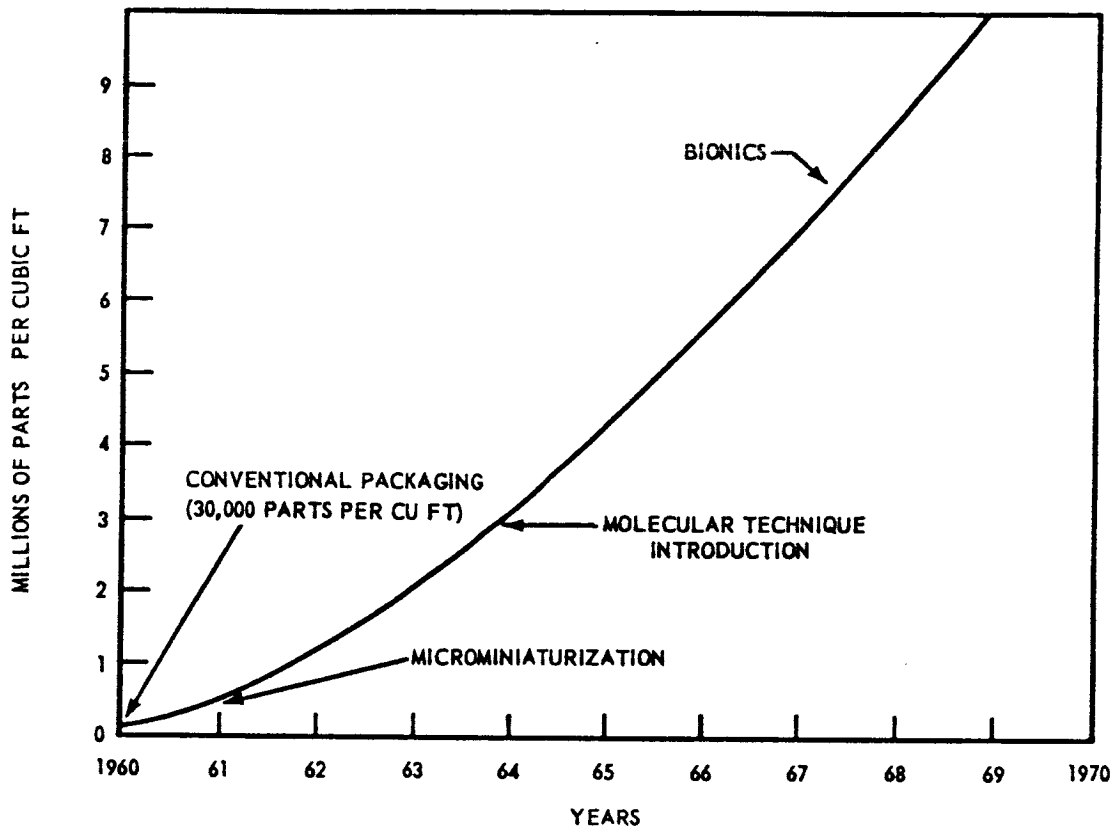
Figure 2-3 shows that MTBFs of computers are expected to increase by a factor of almost 100 by 1972. This would lead to computers with MTBFs in the 50,000 to one million hour class. Much of this may be achieved by elimination of moving parts and by the rugged, compact nature of integrated, solid-block circuitry.

However, such improvement will not come without much emphasis upon development of the new electronics.

Present aerospace computers with MTBFs between 500 and 10,000 hours may not offer adequate overall system reliability for some of the long term space missions.

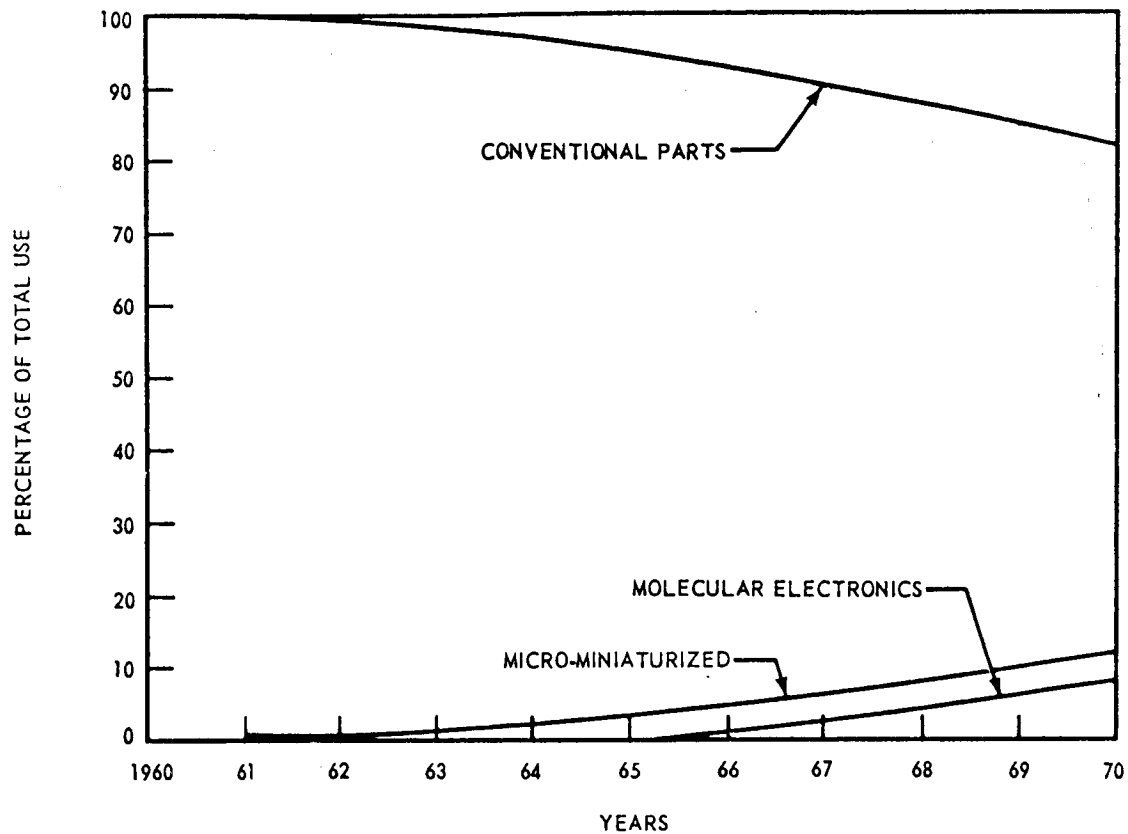
2.1.2 Memory Density

Two types of memory are considered here; tape recorders and computer internal storage. Where large quantities of scientific data are to be gathered and stored for future readout and/or recovery, the memory density problem refers to the weight and size of the required recorder and its tapes.



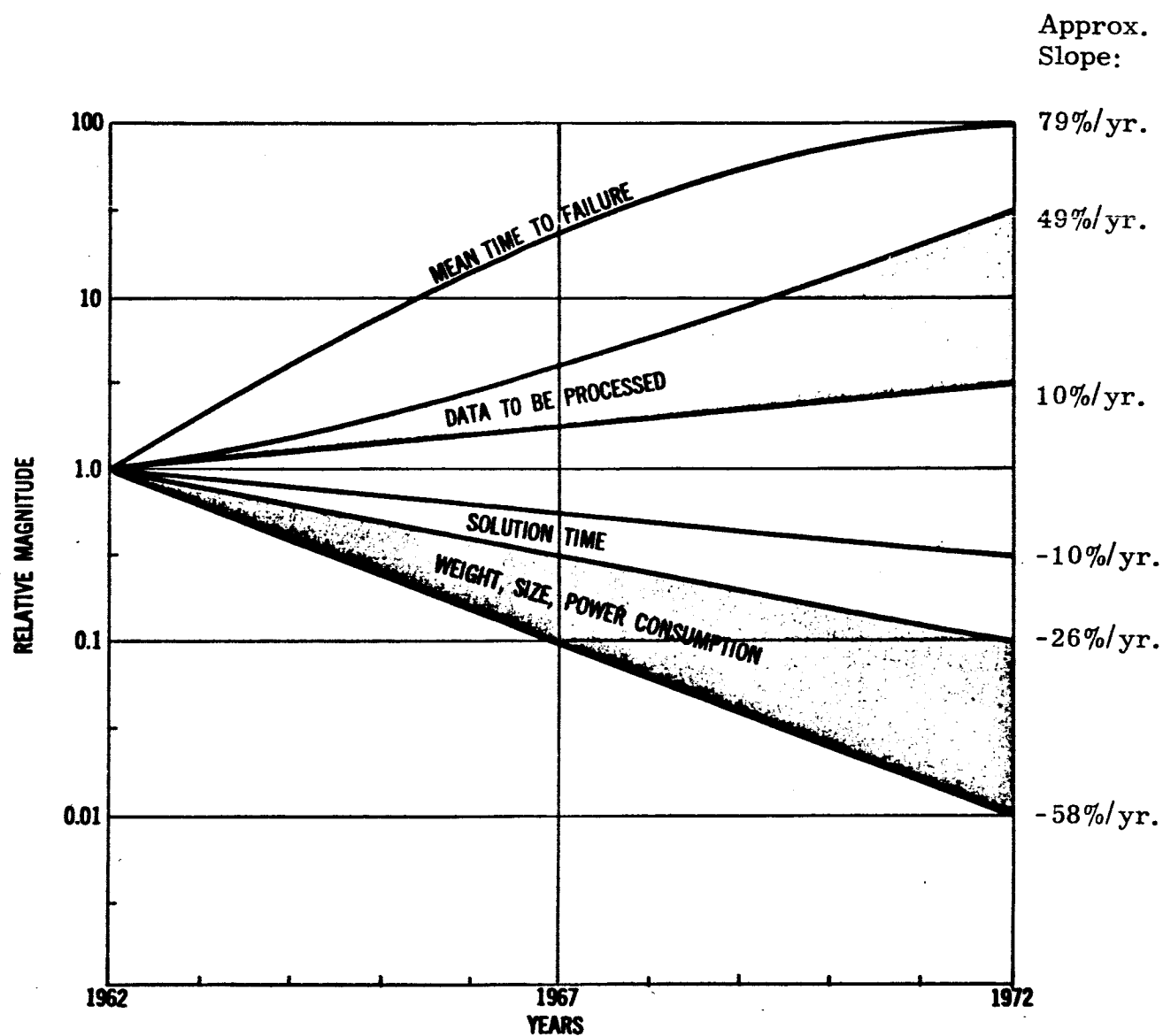
SOURCE: Ref. 2.1

FIGURE 2-1. ELECTRONIC MAXIMUM DENSITY TRENDS



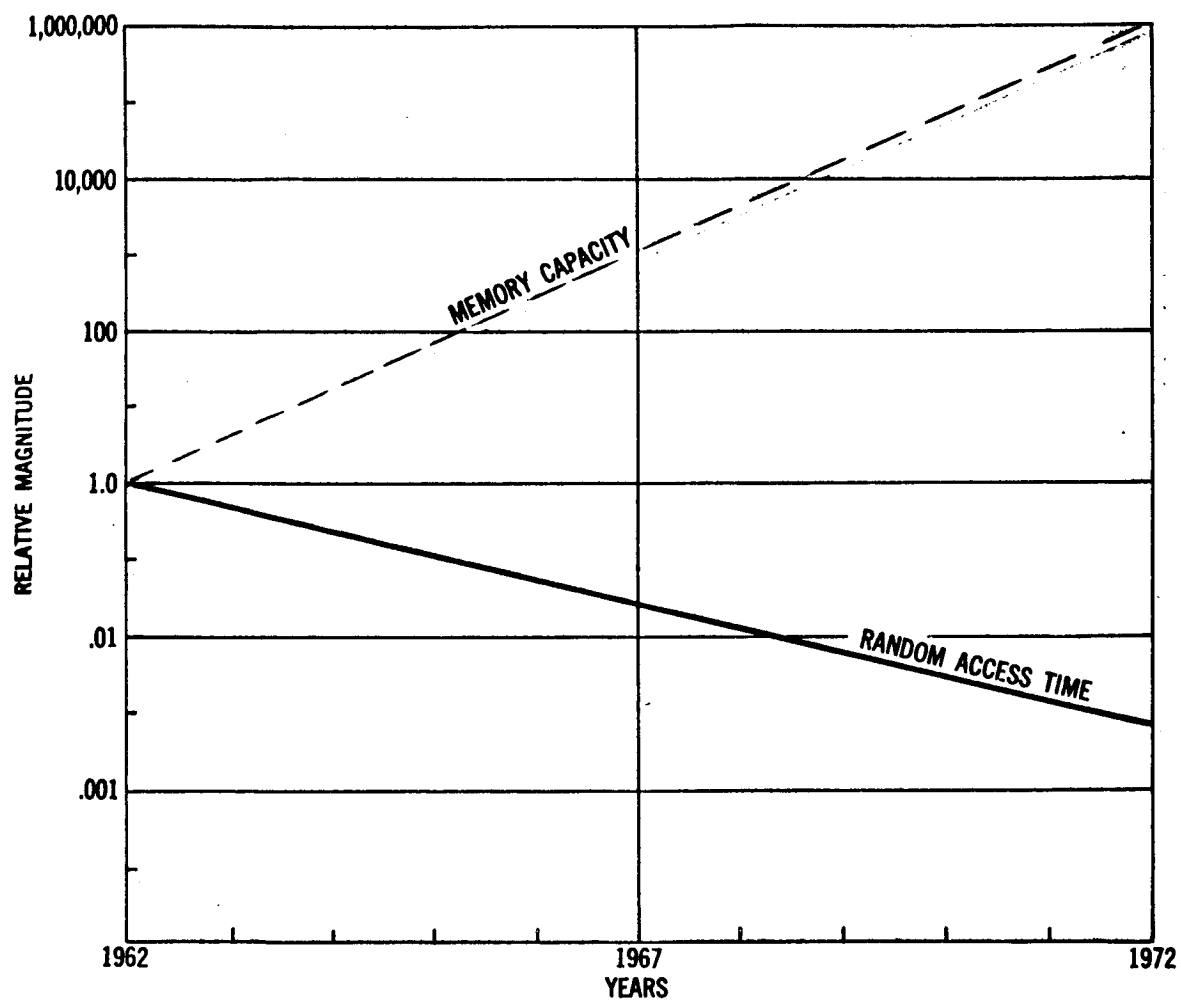
SOURCE: Ref. 2.1

FIGURE 2-2. USE TRENDS IN
AEROSPACE ELECTRONIC EQUIPMENT



SOURCE: Ref. 2.1

FIGURE 2-3. ON-BOARD GUIDANCE AND
NAVIGATION COMPUTERS



SOURCE: Ref. 2.1

FIGURE 2-4. GROUND BASED DATA PROCESSING

Figure 2-5 shows some comparative storage densities.

A standard 7 inch reel can hold 1,800 feet of 1.5 mil mylar tape. Since the problem is linear we may establish a rule of thumb and say that for 1/4 inch tape with four tracks:

$$10^{12} \text{ bits} = 184 \text{ reels}$$

$$10^{11} \text{ bits} = 18.4 \text{ reels}$$

$$10^{10} \text{ bits} = 1.84 \text{ reels}$$

$$10^9 \text{ bits} = 0.184 \text{ reels}$$

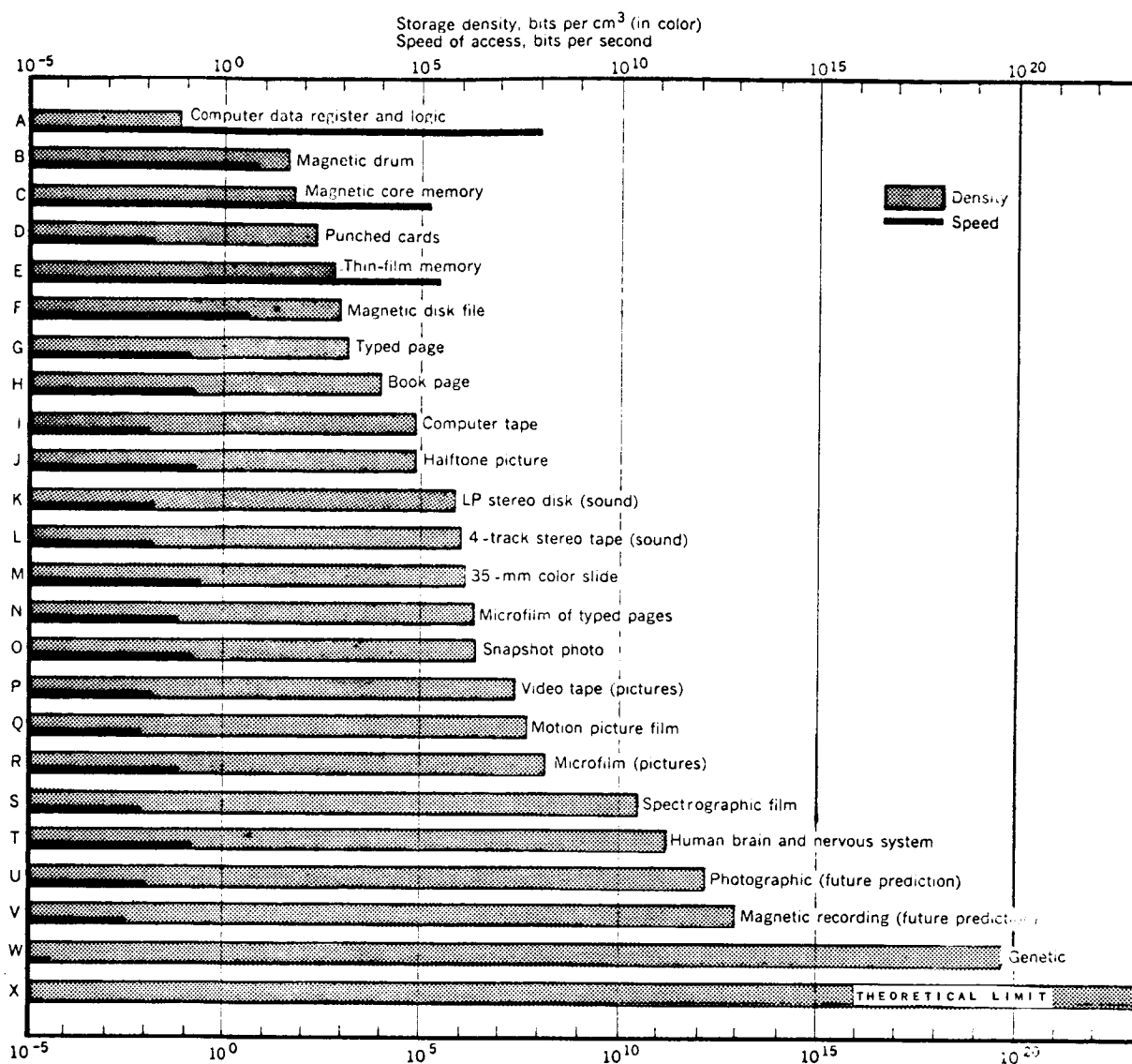
With a tape 2.5 inches wide having 40 channels, we might be able to store almost 10^{11} bits on one 2,400-foot reel.

However, present digital computer tapes can handle only about 10^3 bits per cubic centimeter. Our 2,400 foot by 2-1/2 inch reel with present technology could handle about 5×10^9 digital bits. Some recent experiments indicate that a density of 500000 bits per square inch is possible.

2.1.3 Data Compression (from Ref 2, 5)

"Inclusion of much extraneous data makes it highly desirable to be able to screen the complete data, to select the usable portion for telemetry. If this is accomplished early in the process, a better utilization of the greater portion of the processing capability can be achieved. Because the volume of data involved is significant a rapid data screen is desirable. Careful consideration of this possibility in both ground systems design and satellite data systems design is important.

This area of concern is called data compression. It might be called an extension of the rapid data screen to the spacecraft itself. Criteria for data compression are a function of the experiment and its environment. Work in this area is vitally concerned with the basic information content of the data, and a determination of the means for practically measuring and conveying that information in an adaptive manner. An early testing of any development in this area might be accomplished by applying the techniques to the screening of data at the data acquisition site (or sensor, or instrument) so that the limited communications capability available could transfer to the user a greater knowledge of the experimental results in any given time.



SOURCE: Marvin Camras (Ref 2.3)

FIGURE 2-5. VOLUME STORAGE DENSITIES AND SPEEDS

The bits of information stored in a cubic centimeter of the record are compared with a theoretical limit of 10^{23} atoms per cubic centimeter (Avogadro's number). The DNA genetic system approaches the ultimate in compactness. For each kind of record the random access speed is shown, except in the case of sound records and movies.

Utilization of a general-purpose computer throughout a significant portion of the data processing operation has provided a great deal of capability and flexibility. As more experience is gained with the specific and unique requirements of processing data from scientific satellites, it becomes clear that a machine of somewhat special capability might be required. This effort will concern itself with the basic organization of the machine from the standpoint of volume-data input and output, and the fundamental instruction repertory required to facilitate many of the data processing operations (end of quote). In other words, the data compression problem is chiefly one of software and computer organization after machines of the required size and speed become available and of identifying the relevant information desired. "Associative memories", now under R&D may help greatly in the data screening problem.

2.1.4 Maintainability

The capability of astronauts to make simple module replacements and minor repairs often becomes a significant factor in applying low reliability equipment to long term missions.

It is possible, in principle, to so design a computer that many modular blocks of circuitry are identical and interchangeable. If these are arranged for easy disconnect, removal, and replacement one or two spare modules of each type could permit repair in flight.

Such design adds to the weight and size of the hardware and reduces its reliability even further by introducing additional connectors.

2.1.5 Pattern Recognition

It will often be desirable to have data processors capable of recognizing groups of data as belonging to a class or as being a distinct member of a set. A common simple problem is recognition of a particular star by its position in a constellation among a few billion bits of information relating all star images as seen from the vehicle.

To a man the problem is ridiculously simple; to a computer it is a task of gargantuan proportions.

The star pattern problem is relatively simple compared to some of the desired functions, such as 'print out positions of all industrial installations in the terrain being flown over.'

That computers have a long way to go in pattern recognition is obvious when you consider, for example, the decision that the subject of a black and white aerial photograph of the Washington monument is the same as the subject of a color photograph of the Washington monument made at ground level. This is a simple decision for a man and a formidable decision for a computer.

What we are seeking here is a computer with gestalt perception and the capability to process abstractions. Computers have been taught to recognize written words, and there is real hope that bionics research can provide software and processors with near android capabilities. When ? -- no comment.

2.1.6 R&D Required

Wherever it is believed that the required computer could not be built (or could not be programmed) with today's technology, this deficiency has been noted on the relevance tree, and in general indicates that some specific advancement in the state of the art is required. For example "R&D required, pattern recognition" indicates the problem discussed in Paragraph 2.1.5.

2.1.7 Navigation Software

Refer to concept 6(E025)G. The problem of computing the positions of the user vehicles from the available information has been solved in principle as a part of the overall concept. However, it is a fairly expensive step from this in-principle method to a practical software program for a specific ground based computer to perform the required service automatically. In addition to the computational program, the proper input/output routines and equipment must be designed to mate with the communication subsystem, and provide simultaneous, multiple access to the facility. This implies either buffers and an executive routine or parallel processing of many separate sets of information.

2.1.8 Radiation Alteration of Memory

Usually, magnetic storage is not adversely affected by a few high energy particles or photons; hence, magnetic cores or films are preferred as memories in high energy flux environments. However, the electronics driving the magnetic cores can have extraneous noise pulses introduced by carrier injection into a junction area. Such a pulse in a most significant place of a number can result in such things as destruction of program continuity.

Adequate shielding for all semiconductor electronics can reduce the risk factor to almost insignificant values, except in close proximity to an operating nuclear reactor or during a solar flare. Excess heating due to radiation can raise the magnetic core temperatures above their "curie point", but such fluxes are met only in close proximity to a reactor core. Continued exposure over a long time can cause deterioration of magnetic film material by producing chemical changes and transmutations.

2.2 DIGITAL PROCESSOR

This term includes all of the equipment required to process the digital data, from input encoders to telemetry modulators and display/output devices, even where actual computation is not involved. It may include interface with command and telemetry radio equipment and experiment control equipment. In some cases, it may also include television, photoscan encoders, and mapping radar encoders. The conversion of task instrument outputs to digital form for processing is also included. (In most concepts "digital processor" means a "digital computer" with its input/output equipment.)

2.2.1 Reliability

When stated alone reliability refers to the reliability of the entire data processing complex. When followed by a specific key word, it denotes the reliability of a particularly critical item (for example, Reliab. Variable speed tape recorder). (See also paragraph 2.1.1.)

2.2.2 Data Compression

Refer to paragraph 2.1.3.

2.2.3 Memory Density

Refer to paragraph 2.1.2.

2.2.4 Maintainability

Refer to paragraph 2.1.4.

2.2.5 High Temperature Operation

Operating temperatures of semiconductor computers are limited by the critical temperature at which average random thermal motion energy of carriers within the semiconductor material exceeds the energy of the junction potential barrier. In germanium this is around 100°C; in silicon it is around 150°C.

Usually forced cooling is required and may be by air motion, liquid heat exchanger fluids or thermoelectric junction refrigeration.

The trend toward smaller, micropower circuitry tends to reduce the internally generated heat, but is no help to the problem of extremes of environmental heat.

Eventual perfection of means for growing and selectively etching and doping diamond and carbide wafers may permit semiconductors which operate at temperatures in the 1,000°C region. Some organic semiconductors may permit higher critical temperatures eventually, but little progress in these directions is expected before the late 1970's.

Magnetic core materials are limited to operate below their so-called "curie point" temperature.

2.3 TAPE RECORDER

This generic term usually refers to hardware for magnetic recording on oxide coated plastic tapes, but can also include punched paper tapes, thermoplastic tapes, metal tapes and photoelectric recording upon photosensitive film.

Recorders are available with speed capabilities ranging from 1 bit/sec to 5 megabits/second. However, no one recorder is available to cover the entire speed range. High speed (Video) recorders usually perform a scanning motion across the relatively slow moving tape. Some types use a rotating read/write head for scanning, and consequently have tape and head wearout problems. Recently, non-mechanical scanning methods have emerged and are expected to eventually replace the spinning head types.

Analog recording at slow speeds usually requires that the data be superimposed as frequency modulation upon a high frequency carrier. If readout at higher speeds is required, the carrier frequency then becomes a function of readout speed. With high density recording, extreme speed ratios can lead to a requirement for microwave carrier pickoffs which are beyond the present state of the art. Speed ratios of 100:1 are practical, but some data burst (memory dump) problems could require ratios as high as 10^6 :1. Channel splitting, with sequential recording and simultaneous readout of adjacent channels, can serve to reduce speed ratios required for data dumping.

2.3.1 Reliability

Being electromechanical devices, with wearing parts and spinning shafts, tape recorders are subject to the wearout, jamming, bearing seizure and vibration problems usually met in other "spinning wheel" devices.

Spaceborne tape data recorders have operated for as long as one year without attention. Claimed MTBFs of tape recorders in general range from 50 hours to over 1,000 hours. Such refinements as air bearing floated head bearings, tape cartridges, pneumatic clutch-brake tape drive servos and brushless dc motors are being applied to reduce computer down time. Usually, the repairs required after a function failure are minor, and can easily be made by the human operator. The same failures in an unmanned mission could result in total loss of mission data.

2.3.2 Memory Density

See paragraph 2.1.2.

2.3.3 Research and Development Required, High Speed Readout

As previously mentioned, the problem of recording data at a relatively low rate for a long time, then rapidly transferring a copy of the data tape via telemetry during a short time of optimum communications presents some real difficulties with tape speed ratios and readout bandwidths wherever more than 10^7 bits of data are involved.

Obviously, some data compression research must be applied, but it is quite likely that even with a high ratio of data compression, high speed readouts (beyond one megabit/second) are still needed.

2.4 VARIABLE SPEED TAPE RECORDER

(Refer to Subsection 2.3, specifically paragraph 2.3.3.)

Recorders with write/read speed ratios of 10:1 and 1:10 have been used in space. Situations will exist where speed ratios of 10^4 or 10^5 to one would be useful.

In varying recorder speed, the data carrier frequency also varies and becomes a function of motor speed control. This can present a tuning problem in the demodulation circuits. Since FM is used to record analog data at slow speeds, motor speed variations appear as noise during readout. Magnetic domain recording of digital data permits zero speed readout, but high speeds still are limited by the bandwidths of the associated pulse amplifiers.

2.4.1 Reliability

See paragraph 2.3.1.

2.4.2 Weight

In general, the total weight of a tape recorder system is a direct function of the total number of bits capacity, plus a fixed weight which is a function primarily of data rate.

Audio tape and data recorders weigh from 1-1/2 to 50 pounds. Video recorders weigh from 100 to 5,000 pounds, depending upon bandwidth, reliability, ruggedness, access methods, etc. The Mariner 4 data tape machine weighed 17.04 pounds; its data encoder weighed 22.45 pounds. 21 pictures of 250,000 bits each (5.25 megabits total) were stored. The desired aerospace data recorders should weigh less than 20 pounds, plus the required tape. Considerable development work remains to permit packaging all of the desired features into a small reliable unit.

2.5 VIDEO ENCODER

This keyword means all of the hardware required to process picture information into a form suitable for recording and/or transmission over the communications links.

When analog television transmission is used, the term merely means those camera circuits required to provide the composite video signal.

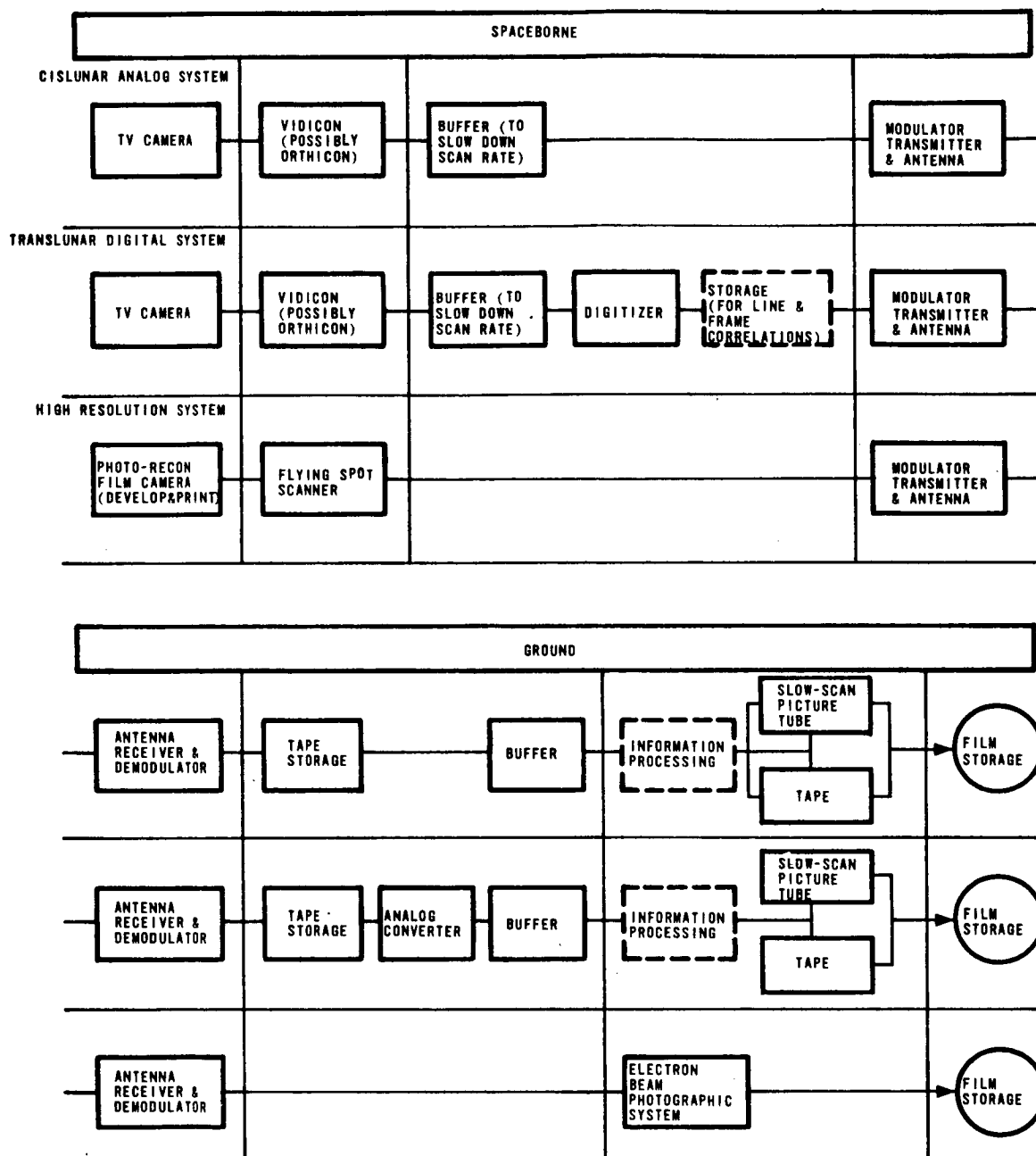
When digital television is used, the term means a special device whose function it is to generate a series of digital numbers representing picture element locations, gray scale and frame color or their derivatives.

Figure 2-6 shows the three principle television methods. Figure 2-7 shows bandwidth requirements.

Table 2-1 summarizes some present aerospace television designs.

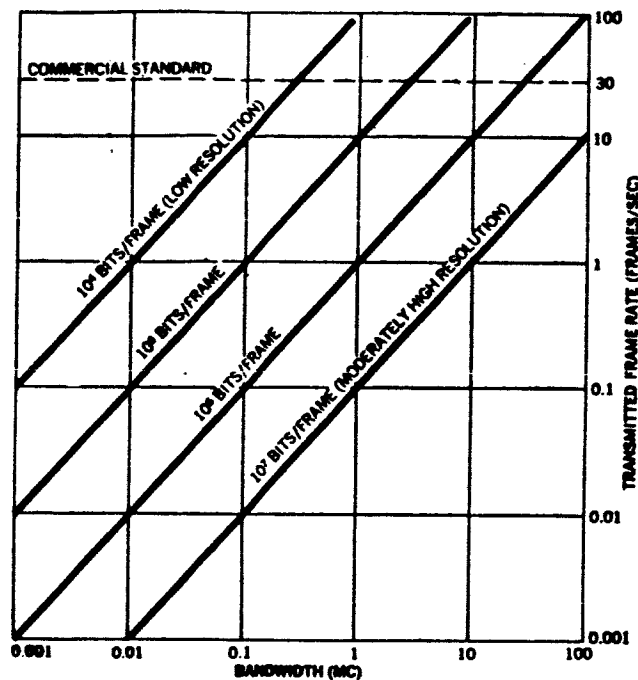
2.5.1 Reliability

The encoder is critical to television system operation. The failures of Ranger's television systems received widespread attention and led to Congressional discussions. The design was good, but bad luck certainly was a factor. No system is perfect, and one this sophisticated has many chances for failure. Other experiments aside, the television pictures of Mars were what classed Mariner IV as a success. Without them, the public would have had little interest in the experimental results. To this extent the television encoder is a critical item and its reliability certainly deserves priorities in design effort.

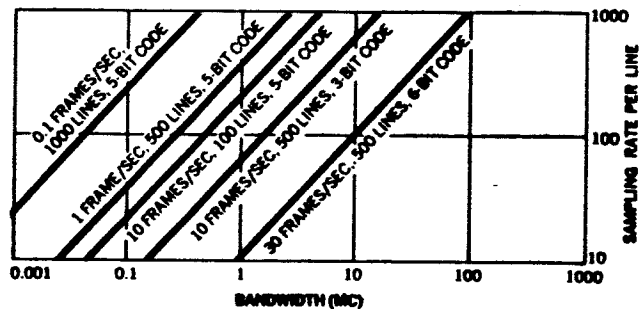
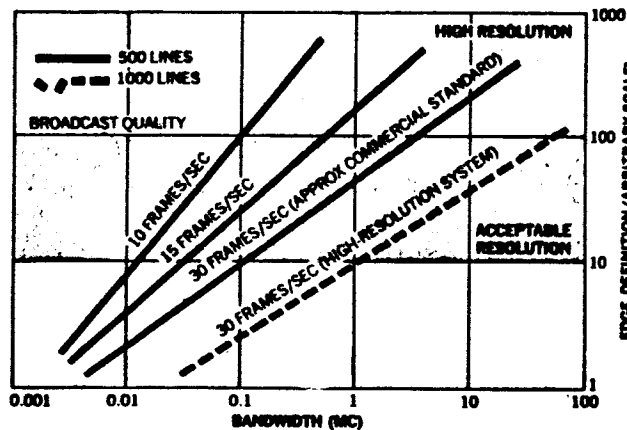


VARIOUS combinations of subsystems can be used to relay video from space. Mission goal, available power and weight, subject to be viewed—all these determine the tradeoffs. Some typical systems are roughly indicated here. There are significant differences between analog systems, suitable for cislunar travel, and digital systems, most likely required for longer trips. (The buffer function can also be performed at times by a storage vidicon.) High resolution calls for a still different system.

FIGURE 2-6. TELEVISION METHODS
(Frank Leary, "Space/Aeronautics"
March 1964 - Ref 2.7)



BANDWIDTH requirements can be reduced most handily by stretching the signal in time (that is, reducing the number of frames transmitted per second), as this graph shows. However, the nature of the analog signal puts an effective lower limit on frame rate.



REAL-TIME transmission of narrow-bandwidth signals imposes limitations on both analog and digital systems. In analog (top), it results in pictures of low contrast, indicated here in terms of edge definition. In digital (bottom), it forces a low sampling rate.

FIGURE 2-7. TELEVISION BANDWIDTH REQUIREMENTS
(From Frank Leary, "Television from Space", Space/Aeronautics
March 1964 - Ref 2.7)

TABLE 2-1. SOME CURRENT AEROSPACE TV SYSTEMS*

<u>System</u>	<u>Tiros</u>	<u>Ranger</u>	<u>Nimbus (APT)</u>	<u>Nimbus (AVCS)</u>	<u>OA0**</u>	<u>Apollo</u>
Vidicon size (in.)	1/2	1	1	1	1	1
Resolution (TV lines)	350	650	800	800 (10% response)	310	225
Dynamic range (ft candle-sec)	0.01-1.0	0.005-0.5	0.03-0.7	0.004-0.4	Star of magnitude 2 = 5	0.01-1.0
Linearity (percent)	4.0	1.0	0.5	0.5	±1.0	4.0
Image format (in.)	0.25 sq	0.44 sq	0.44 sq	0.44 sq	0.44 sq	0.5 x 0.38
Shutter time (msec)	1.5	5.0	40	40	Open	No shutter
Power consumed (watts)	9	19	19	15	11.2	6.5
Weight (lbs)	9.7	17	19	18.8	17.8	4.5
Frame time (sec)	2	2.5	200	6.5	1.0	0.1
Scan lines	500	1,130	800	800	350	320
Bandwidth required (kc)	62.5	200	1.6	60	60	400

* Data courtesy RCA Astro-Electronics.

** OA0 system required only for tracking star position.

Source: Frank Leary, "Television from Space", Space/Aeronautics, March 1964. (Ref. 2.7)

In such missions as the lunar logistics vehicle 6(LD02)Y, the television subsystem (part of G&N) is necessary to assure a safe, unmanned lunar landing of essential payloads. In other instances, television failure may only represent an annoyance or a loss of desirable data.

2.5.2 Noise

Referring to Figure 2-7 the required bandwidths are based upon an assumption of good signal to noise ratio. The results of noise in video circuits are either loss of picture data, or a requirement for redundancy (and consequently greater bandwidth).

The process of encoding television will introduce some noise. In low resolution systems, this may cause total loss of a significant fact (an artifact on the Moon might be missed). In high resolution systems, noise due to encoding may be only a minor annoyance.

2.6 PHOTOGRAPHIC FILM

Photography in space will be a principle means for permanent recording of data. Color photography is of particular interest, since it adds a dimension (radiation frequency) to the information recorded on the image. Both natural color and scaled IR color photography will be important.

In cases where film is to be recovered from space, the main consideration is to prevent damage to unprocessed film.

Several concepts will require complete automatic photography with processing and facsimile type video encoding of the finished picture. This process is sometimes referred to as on-board photo-scan.

Lunik III employed a rather unsatisfying type of onboard photo-scan telemetry. Color photography by the Land (Polaroid) process and the thermoplastic (GE) process both promise to improve photo technology for space work, eventually.

2.6.1 Onboard Processing

Development effort will be necessary to provide the automatic, high resolution, full color photography, color processing and high resolution facsimile/color transmission of the pictures. Adequate systems do not now exist, but color Polaroid shows promise of a simple method. Color facsimile is entirely feasible, but requires development. The use of

multi-frame (separated colors) in thermoplastic photography may help to improve the onboard processing problem by elimination of the chemical development cycle.

2.6.2 Radiation Protection

Films capable of responding to low intensity visible light can also be marked by x-rays, gamma rays and ionizing particles. Shielding against some of these involves a significant weight penalty. Film research may develop emulsions with high energy cut-off characteristics, although this is very doubtful. Better shielding methods and materials may be found.

Long duration missions, such as planetary landers will receive significant doses of high energy radiation prior to the surface photography cycle.

Color films returned to Earth from the second, manned Gemini flight showed little serious degradation after over a week in orbit.

A possible development is film that is not sensitive to radiation until shortly before exposure, and is insensitive shortly thereafter.

The thermoplastic process conceivably can sidestep the problem, since the film is sensitized just prior to exposure and becomes insensitive after heat setting. However, its use for color photography may require exact registration of separate frames for each color band.

2.7 COUNTDOWN ELECTRONICS

This term applies only to the electronic circuitry associated with the atomic clock for the relativity experiment. A periodic readout of onboard real time, as presently indicated from a countdown of the Cesium oscillator, will be telemetered to Earth for comparison with the earth-bound "twin" clocks indicated time.

Electronic circuitry for such a function is existing hardware, but must be re-designed if it is to be used unattended in space.

2.7.1 Reliability

Over 0.99 reliability for a one-year mission is desired.

2.7.2 R&D Required

More development than research is needed to supply clock electronics with high reliability for a one-year mission.

2.7.3 Temperature Control

Very little improvement in present technology is required. The temperature control is primarily for the quartz oscillator used as a "flywheel" on the cesium clock.

2.8 THERMOPLASTIC COLOR PHOTOGRAPHY

Black and white photography is accomplished by exposure to light of a soft thermoplastic film upon which a uniform surface charge has been sprayed. The light discharges the areas exposed. The latent electronic image is fixed by heating the film in an electric field. The bound charges cause plastic flow along field lines, raising (or depressing) the darker areas more than the partly discharged (illuminated) areas. Polarized light permits projecting a black and white picture from the film.

2.8.1 R&D Required

Either a multilayer film with filter sandwiches, or a beamsplitter and filter optics and separate black and white images for each color might permit color photography by this process, but experimental development will be required.

The process promises extremely high resolution (infinitely fine "grain") photography and, in addition, may be able to survive large radiation doses prior to and after exposure to light. Elimination of chemical solution processing is a definite plus for automated operation.

2.8.2 Spaceborne Packaging

R&D must be followed by careful design work to provide a fully automated color still and motion camera system capable of reliable operation during automated interplanetary flights. Means must be provided to protect the developed film from high temperatures.

REFERENCE LIST FOR SECTION 2

- 2.1 "Aerospace Technical Forecasts, 1962-1972 and 1960-1970," Aerospace Industries Association of America, Inc., 1962, 1725 DeSales Street N. W., Washington 6, D. C.
- 2.2 Thomas C. Bartee, Digital Computer Fundamentals, McGraw-Hill, 1960.
- 2.3 Marvin Camras, "Information Storage Density," IEEE Spectrum, July 1965, page 98.
- 2.4 Adi J. Khambata and Roger A. Urban, "Space Age Digital Computers - Components and Capabilities," ARS paper 1932-61.
- 2.5 Albert G. Ferris, "Data Acquisition and Processing from Scientific Satellites," NASA SP-11, Vol I, Proceedings of the NASA-University Conference on the Science and Technology of Space Exploration, page 293.
- 2.6 James McDermott, "Optical Data Processing," Space/Aeronautics, January 1963, page 79.
- 2.7 Frank Leary, "Television from Space," Space/Aeronautics, March 1964.

Section 3

ESCAPE AND RECOVERY

3.0 GENERAL COMMENTS

This subsystem includes all fault sensing, warning, ejection, and recovery functions needed to insure safe crew escape from vehicles encountering catastrophic malfunctions resulting in aborted missions, as well as planned recovery of crew and equipment from successful missions (usually during the terminal phase of an operation).

Except for launch escape and deboost from Earth orbit, the entire problem of escape and recovery appears to be a technological deficiency.

3.1 SINGLE MAN EJECTABLE PODS 6(E006)S

The space lifeboat concept would be an escape pod, somewhat smaller than the Mercury capsule, but similar. An ejection charge would separate the pod from the mother craft. Solid retro rockets would fire to deboost the pod from orbit after manual alignment and triggering by the occupant. Parachutes would open and deploy automatically at a pre-set barometric altitude. Life support could be limited to the spacesuit pack (no pressurization of the cabin would be required).

3.1.1 Vehicle Stability, Pod

The occupant can hold attitude constant during retrofire with a simple, fly-by-wire reaction jet system. However, a simple means for maintaining stability during entry is needed. Preferably this would be some consequence of the aerodynamic design and operate passively, rather than requiring pilot control.

3.1.2 Weight, Heat Shield

If such a pod were built now, its heat shield probably would not weigh less than that for the Mercury capsule. Lighter weight, but still effective, heat shields are needed to lessen the payload penalty of carrying an escape pod.

3.1.3 Research and Development Required, Hypersonic Operation

Research and development will be required to develop the optimum, minimum weight, one-man pod which still can survive hypersonic flight — preferably in a passive and unguided manner.

3.2 ESCAPE POD - ROCKET SEPARATED 6(E008)Y

See Subsection 3.1.

3.2.1 Research and Development Required, Escape Pod

See Paragraphs 3.1.1 through 3.1.3.

3.3 USE NORMAL COMMAND MODULE FOR RE-ENTRY 6(MD02)S
6(MD04)S 6(MF02)S 6(MO04)S 6(MO04)X 6(PF02)S 6(YF02)S

(State of the art. No abort possible after escape injection)

3.4 USE NORMAL MEM RETURN 6(MD02)L, 6(MD04)L

Capability for extended survival on the Martian surface is doubtful. A heavy dose of morphine (intravenous), at the astronaut's option, is suggested as an alternative in order to avoid pain in termination of mission. A really far-out idea is based upon using cryogenic propellants to quick-freeze men in a state of suspended animation, until eventual rescue can be accomplished.

3.5 VACATE VEHICLE AND RELY ON EVA UNTIL RESCUE 6(MD02)X
6(MD04)X 6(MF02)X 6(VO02)X 6(YF02)X

(In case of fire, explosion or wreck of lander vehicle.) Similar action during midcourse flight at least permits temporary prolongation of existence; usually, no abort from midcourse is possible except with operative space vehicle.

Section 4

GUIDANCE AND NAVIGATION

4.0 GENERAL COMMENTS

Guidance and navigation includes all sensing and computation functions for display, or automatic actuation of stability and control, or propulsion subsystems for guiding the vehicle along its prescribed mission path. It includes sensing and computation functions of rendezvous and docking.

This section is grouped by mission phases (i. e., injection, midcourse, etc.) rather than by configurations.

4.1 INJECTION GUIDANCE

This term refers to guidance during the thrusting and coasting periods required to establish a ballistic orbit, such as orbital injection, injection to escape, injection to translunar orbit, etc.

4.1.1 Radio Command

The oldest, and so far the most reliable, method for injection guidance requires remote tracking and radio commands to the stabilization and propulsion subsystems of the rocket. World-wide networks exist for orbital radio command guidance. Primary concentrations of ground facilities are at AMR and PMR.

Line of Sight Range. The principle limitation upon radio command guidance is that of line-of-sight range. However, by transferring control successively to a network of ground tracking stations around the world, this limitation can be largely overcome. Guidance behind the Moon or a planet cannot be performed by radio command from Earth with present facilities and methods.

Reliability. Since radio command guidance requires the simplest on-board mechanization (except for open-loop pitch program guidance), it is the most reliable. Multiple redundancy and easy maintainability are possible at ground stations where the majority of the work is done. Reliability of the on-board system is a function of command receiver and SCS subsystem reliabilities, since both are critical. To a lesser extent telemetry feed

back of command confirmation also affects reliability. Overall confidence in radio command guidance can be made higher than 99 percent.

Velocity Accuracy. During the first few seconds after lift-off, the booster is tracked by the AN/FPS-16, C-band radar at AMR. This radar gives position ± 130 ft and velocity ± 2 or 3 ft/sec. Data from the AN/FPS-16 feeds into the IP-7090 computer at the Cape. The output of the IP-7090 also is carried directly to Goddard Space Flight Center for display and initial calculations. A standby, S-band radar (VERLORT) also tracks the booster. RADOP is a generic term including the C-band and S-band radars, AZUSA and UDOP installations. RADOP is presently operational.

When the booster reaches sufficient altitude to permit tracking by the AMR AZUSA Mark 2 System, the IP-7090 input is switched to this more accurate tracking information source (angular error of radar reduces cross course accuracy as range increases).

A new X-band, CW system called MISTRAM is expected to give instantaneous booster velocity to about 0.2 ft/sec and position to about ± 30 feet (± 10 feet above 10 degrees elevation). However, with adequate smoothing intervals of about $1/2$ second, an RMS range uncertainty of 0.4 ft and an RMS range-rate uncertainty of 0.02 ft/sec is expected.

The operating ranges of MISTRAM are:

20 to 1,000 miles - range

0 to 50,000 ft/sec - velocity

0 to 750 ft/sec^2 - acceleration

0 to 360 degrees - azimuth

5 to 85 degrees - elevation.

MISTRAM is expected to be able to measure relative position and velocity between a station and ferry during rendezvous much better than ± 10 ft and 0.5 ft/sec. Hypothetically, such a system used for radio guidance of translunar injection could permit cut-off accuracies of ± 0.14 ft/sec and ± 0.001 degrees, with a consequent lunar impact CEP of less than 10 nautical miles. However, the dynamics and geometry of insertion would require moving the station by about 400 miles per day to have it always suitably located for control of injection. Such a multiplicity of MISTRAM stations

is not likely during the Apollo time period; nevertheless, it is of interest to contemplate the possibilities of such advanced radio guidance.

MISTRAM definitely will be available for tracking ascent to Earth orbit during the Apollo period and is expected to be capable of the following accuracies for direct ascent:

$$\begin{array}{ll} \Delta X = 5.4 \text{ ft} & \Delta \dot{X} = 0.11 \text{ ft/sec} \\ \Delta Y = 0.6 \text{ ft} & \Delta \dot{Y} = 0.04 \text{ ft/sec} \\ \Delta Z = 14.5 \text{ ft} & \Delta \dot{Z} = 0.62 \text{ ft/sec} \end{array}$$

This matrix would propagate to an exemplary apogee matrix of:

$$\begin{array}{ll} \Delta X = 185 \text{ ft} & \Delta \dot{X} = 0.15 \text{ ft/sec} \\ \Delta Y = 30 \text{ ft} & \Delta \dot{Y} = 0.10 \text{ ft/sec} \\ \Delta Z = 1,350 \text{ ft} & \Delta \dot{Z} = 0.73 \text{ ft/sec} \end{array}$$

based upon a cut-off parameter set of:

$$\begin{array}{ll} X = 0 & \dot{X} = 0 \text{ ft/sec} \\ Y = 1.119 \times 10^6 \text{ ft (down range)} & \dot{Y} = 25,213 \text{ ft/sec} \\ Z = 0.426 \times 10^6 \text{ ft} & \dot{Z} = -717.5 \text{ ft/sec} \end{array}$$

With present boosters, the General Electric/Burroughs guidance system takes control of the vehicle between 20 and 60 seconds after lift-off. This can provide Goddard or Houston with position and velocity data similar to that available from the IP-7090 Complex. A data quality monitor determines the best data for use by two 7090's which operate independently, then cross-check results.

Target dates before 1967 are set for systems to achieve the following accuracies at AMR:

$$\left. \begin{array}{l} \text{Velocity } \pm 0.01 \text{ ft/sec} \\ \text{Position } \pm 0.01 \text{ ft} \\ \text{Acceleration } \pm 0.01 \text{ ft/sec}^2 \end{array} \right\} \begin{array}{l} \\ \\ 0 \text{ to } 500 \text{ ft} \end{array}$$

Velocity ± 0.1 ft/sec	}	500 to 5,000 ft
Position ± 0.1 ft		
Acceleration ± 0.1 ft/sec ²		
Velocity 1 ft/sec	}	5,000 to 25,000 ft
Position 1 ft		
Acceleration 1 ft/sec ²		

These programs include such things as laser, radars, improved transponders, improved optical trackers, etc.

Present data word length of 36 bits is expected to be increased. Bit rates of 64,000/sec are expected to be required (in telemetry data links) during the EGO satellite operational period. Very likely the tracking and control link data also will be greatly increased by 1967.

Electro-optical vehicle attitude determination systems are under study, and performance out to 600 miles range is planned with the following accuracies:

Data rate 100/sec, (1 part in 10^9)

± 0.1 degree (attitude angles)

± 1.0 degree/sec (attitude rates).

4.1.2 Inertial System

Inertial platform or platform IMU refers to a servo gimbaled set of accelerometers and gyros with associated electronics which is capable of yielding position, velocity, and attitude of the vehicle at any time by open loop dead reckoning based upon integration of observed accelerations and knowledge of the gravity field and initial conditions of motion.

Low Cost. The cost of inertial systems range from \$50,000 to \$500,000 for space applications with \$100,000 being a usual figure. Price is expected to decrease for some applications, particularly aircraft, as demand and standards become stable. Being an ultra precision electromechanical device, the platform simply cannot be built and used without extensive "tender loving care" by skilled people. Such labor is expensive.

Reliability Inertial System. MTBFs much in excess of 1,000 hours cannot be expected from gimballed platform systems for space flight in the near future. Five hundred hours MTBF represents a good design. This problem is expected to be overcome by eventual abandonment of platforms in favor of the simpler hardware in strapdown and ESG strapdown IMUs.

Alert Time. Initial warmup, alignment and initialization of platform systems requires in excess of 15 minutes. Preferably, several hours would be allowed. Fast platforms are needed for some military applications, but accuracy compromises are indicated when fast response is mandatory.

Temperature Sensitivity. With platforms which use floated inertial sensors, the viscosity of the flotation fluid is quite temperature dependent. Usually a fluorocarbon lubricant is used which is almost a solid at room temperatures and becomes quite thin at 180° F. This permits heater control (through amplifiers) of fluid viscosity.

Temperature gradients across a platform system can cause physical distortions which lead to guidance and attitude errors. Ideally, the platform should operate in a pressurized, temperature stabilized container, and never in direct sunlight.

Attitude Drift. The principle attitude information is taken from the gimbal pickoffs which depend upon the gyro loops. If gyros were perfect, gimbal loops could be made as steady as star lines. However, even with compensation, residual drifts usually in excess of 0.01 degree per hour will exist. Directional information confidence becomes worse with time in an almost linear manner -- in space stabilized systems -- so that after 100 hours from alignment attitude could not be trusted to better than ± 1 degree. This usually means that an external attitude reference (such as star trackers) must be used.

Since dead reckoning integration of velocity vectors is resolved on the basis of platform attitude information, velocity and position information deteriorate with increasing mission time.

4.1.3 Strapdown Inertial System

By torquing the input axes of body-mounted gyros to return them to body axis alignment whenever rotation of body axes occurs, the gyro torquing power may be related to angle turned through about body axes. Precision digital torquing loops have been developed.

Similarly, pulse rebalanced accelerometers yield pulses representing quanta of components of velocity change along their sensitive axes.

By using a high speed, parallel computing, digital differential analyzer the rotation quanta can be resolved into net rotation from alignment to obtain attitude. Similarly a matrix multiplication permits transforming attitude information into any known coordinate reference frame, whereupon the velocity change quanta may be resolved to get velocity and position vector changes in that coordinate frame.

The outputs from such a strapdown system are indistinguishable from those of a platform at the guidance computer inputs.

Research and Development Required. Strapdown systems have been built but have not yet been tested in space flight. The PRIME re-entry test vehicle is to use a strapdown inertial system.

Temperature Sensitivity. The problem is essentially the same as that for platforms (see paragraph 4.1.2) except that absence of a gimbal structure permits more direct thermal coupling to a heat sink. This can result in development of troublesome hot and cold spots within the sensor block. Block designs are usually subjected to a rigorous thermal profile analysis before fabrication.

Attitude Drift. Gyro drifts have about the same effect on strapdown systems as on platforms. However, additional attitude errors result from the non-commutativity of finite rotations, and from computer truncation and round-off errors. These effects add an attitude error (depending upon total-angle-turned-through) to the time dependent drift error.

Present strapdown systems are suitable for guidance during thrust but are not adequate for long term navigation.

The use of strapdown ESGs is expected to overcome this limitation.

Restart and Align in Orbit. From a reliability point of view it is generally preferable to shut down the inertial subsystem when it is not being used in order to conserve operating time on the wearing parts. However, the calibrations of inertial sensors tend to change between "run-ups" and of course inertial attitude reference is lost at shut down.

Realignment to inertial coordinates may be achieved by reference to star trackers, but usually it is desired to establish reference to "local vertical"

coordinates. This requires a vertical and azimuth reference. In present systems the alignment is limited in accuracy about all three axes because of the uncertainty in pitch and roll (of vertical) as indicated by a horizon scanner. (Accuracy of gyrocompassing to the orbit plane is limited by the "level" reference accuracy.)

What is needed is either precise position information or a precision vertical reference. However, even with precision alignment references, a recalibration of sensors is needed (or, alternatively, sensors with extremely small calibration shifts between runs).

Shutdown Reliability INS. The reliability of an inertial system during a continuous run is not simply related to the reliability during several successive runs. Reliability tends to decrease as some function of the number of previous shutdowns. Effort is needed to explore the deleterious effects of on-off cycling, and prescribe corrective measures which will permit several hundred such cycles without serious sacrifice in reliability of restart.

Some of the deleterious environmental influences are acting upon the INS whether it is running or not, so that reliability also is related to total mission time and to the severity of the environment, whether in use or shutdown.

Weight, Inertial System. Platform inertial systems will weigh between 15 and 150 pounds, strapdown systems between 5 and 100 pounds (depending upon functional requirements and design details, redundancy, environmental protection, etc.).

The general trend of INS weight has been downward and linear extrapolations predict that a one-pound system is possible by 1980, even though we cannot presently see how this might be accomplished. If it is to be accomplished, there will have to be breakthroughs in sensor technology coupled with steady progress in microelectronics.

Deboost and Ascent Calculation. It is possible in principle to have the guidance computer repeatedly cycling through a calculation of the required ΔV for injection, aborts, etc. if they were to be commanded at the next second. This calculation would take into account the real time motions of the earth, the present state vector and other conditions (aerodynamic, heating, g-limits, etc.). This wise guidance computer then would always "know" the correct action to order, whatever the circumstances. Since all the significant parameters are time varying, the calculations would have to be done for each instant during the mission. In a sense, a large

proportion of the computer's capacity would be assigned the task of "worrying" effectively about things that probably will not happen. Since the correct actions depend upon past events, they cannot be computed and stored in advance. Nevertheless, having such up-to-date information instantly available is a worthwhile thing.

Direct, brute force calculations would involve a large scale computer onboard. However, it is possible to pre-compute and store the coefficients (partial derivatives) of deviations within limits from a nominal set of conditions, as a function of future time. By referring to these stored partials as generated by a tape or clock driven cam device, the onboard computer can greatly reduce the arithmetic involved and still give acceptably accurate answers. Software research properly coordinated with system design probably can yield improved mission reliability through application of this method.

4.1.4 Programmed (Injection Guidance)

The oldest form of rocket gyro guidance simply involves sending attitude commands to the stabilization system as a function of real mission time. Simple cam timers can thus order the rocket to approximately follow a precomputed gravity turn power track for injection. If the orders of the programmer are not carried out, there is no way to modify subsequent orders. The system is completely open-loop or blind. The first satellites were guided into orbit by this method.

Engine cutoff may be controlled, open loop, by a timer also, but sometimes a roll axis accelerometer and integrator were used to generate a velocity magnitude cut-off signal.

Velocity Accuracy. Velocity magnitude can be controlled to within thirty meters per second, but direction may be in error by several degrees.

The method is adequate when it is desired only to get into some nominal orbit where orbital parameters are not at all critical.

Position Accuracy. Since programmed guidance is completely open loop, position errors at cutoff are usually in the tens to hundreds of kilometers. Unless the rocket thrust were precisely predictable as a function of time, there is no way to closely control position in such an open loop system.

4.2 MIDCOURSE AND ORBITAL

This term refers to that period after injection and before termination of a leg of a mission. Usually it is primarily an unpowered coasting motion in which the rocket is under the control of Newtonian forces alone.

4.2.1 DSIF

During this period, ground tracking facilities can accumulate much data and apply filtering to permit refinement of their knowledge of the actual trajectory. They then can compute velocity corrections which could be applied at some future time to bring the rocket closer to the desired terminal state vector. These ΔV corrections usually are executed by radio command to the rocket's stabilization and propulsion subsystems.

Figure 4-1 shows the approximate errors of the DSIF network.

Line of Sight Range. See paragraph 4.1.1.

Reliability. See paragraph 4.1.1.

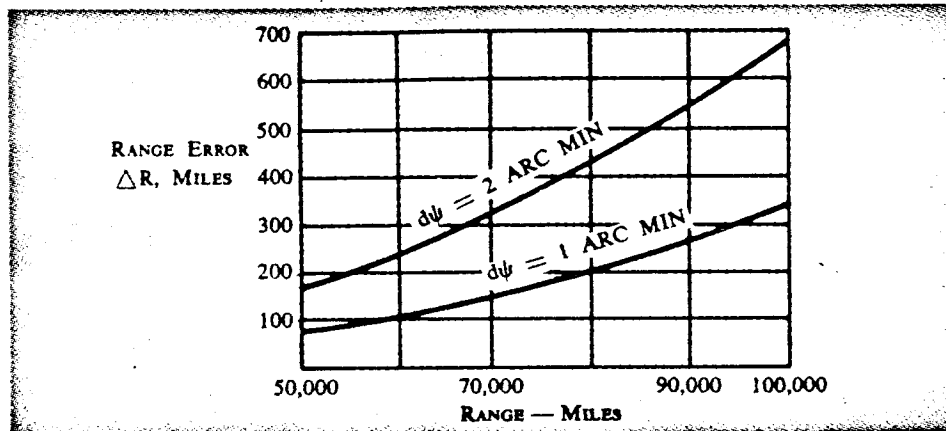
Reliability of DSIF radio command guidance has been amply proved by ranger and mariner flights. Terminal errors at the Moon were only a few kilometers and at the planets only a few hundred kilometers. Failures of on-board command receivers have been experienced. The Russians have had more trouble in this respect than the Americans.

Velocity Accuracy. Velocity accuracy along course improves with tracking time. Since it is a doppler measurement, it is relatively independent of range. However, cross-course velocity depends upon antenna attitude pickoffs. While it improves with range, it cannot be as accurate as the doppler measurement. Overall accuracies are a few meters per second at planetary distances.

Uncertainty in Asteroid Mass. The perturbations of other bodies due to asteroids are so small that accurate estimates of asteroid masses are extremely difficult to obtain. By observing perturbation of the close fly-by probe orbits, such estimates will be improved.

Uncertainty in Ephemeris. Knowledge of the absolute dimensions of the solar system in laboratory units has been very elusive. As a result, there is some residual uncertainty about the positions and velocities of heavenly bodies as a function of future time. These uncertainties enter

1065-89A



Source: Gansler (Ref. 4.1)

FIGURE 4-1. RANGE VERSUS RANGE ERROR
FOR NASA DEEP-SPACE RADAR NET

into space navigation as errors in a manner similar to that in which uncertainties in positions of bench marks enter as errors in an earth's surface survey (or in surface navigation by landmarks).

Radar astronomy has improved our knowledge of distances and velocities. More powerful radars are expected to continue such improvement, but the distances involved have so far ruled out radar skin tracking of the outer planets from the earth.

4.2.2 Inertial System

Manned interplanetary vehicles, notably Apollo, will use inertial platforms as the heart of their on-board guidance systems. Few of the unmanned vehicles will use them beyond the Moon, due to maintenance impossibility and low reliability.

In Apollo, the platform system will be aligned to the stars by manual/optical means before each ΔV . The platform will monitor and control execution of the commanded ΔV , but can be backed up by the strapdown SC subsystem.

Quoting Jacques S. Gansler (Ref. 4.1)

"Mechanizations and Limitations. There are three possible extreme splits for mechanization of a space navigation system: (1) a completely automatic and self-contained system, (2) a complete radio navigation, and (3) manned landmark sighting system. In general, most space navigation systems will lie somewhere in between these, but largely they will utilize one or the other of these as the basis for a prime system and the other as a basis for a backup system.

"A typical self-contained system, using today's hardware, might have the approximate characteristics shown in Table 4-1."

TABLE 4-1. POSSIBLE SELF-CONTAINED
SPACE NAVIGATION SYSTEM

	<u>Size</u> <u>(ft³)</u>	<u>Weight</u> <u>(lbs.)</u>	<u>Power</u> <u>(watts)</u>
Inertial Subsystem (Platform and Electronics)	0.5	30	120
Computer	0.3	20	30
Star Tracker (Gimbaled)	0.36	19	45
Horizon Scanner	0.15	10	9
Interface Electronics	<u>0.1</u>	<u>8</u>	<u>14</u>
Totals	1.41	87	218

Cost. See paragraph 4.1.2.

Reliability. See paragraph 4.1.2.

Alert Time. See paragraph 4.1.2.

Temperature Sensitivity. See paragraph 4.1.2.

Attitude Drift. See paragraph 4.1.2.

4.2.3 Strapdown Inertial

Since the inertial system in a space vehicle is to be used primarily for controlling ΔV maneuvers and hardly at all for navigation, the long-term drifts are unimportant. There is little accuracy difference between the ΔV control by platform and strapdown; but, strapdown I. N. systems offer greater MTBF by a factor of 6 or more. They weigh much less, cost much less and draw less power. A typical strapdown system weighs 35 pounds including computer, draws 75 watts and has an MTBF of over 3,000 hours.

R&D Required. See paragraph 4.1.3.

Temperature Sensitivity. See paragraph 4.1.3.

Attitude Drift. See paragraph 4.1.3.

4.2.4 Manual Optical

The planetary center (or a surface landmark of known position) can be referred to the celestial sphere by two or more simultaneous star sights, or by referring the local vertical (or landmark sight) to an inertial platform recently space stabilized by star sights. This establishes one line-of-position at the time of sighting.

If two or more such LOPs could be obtained simultaneously from two or more near bodies, a geometric fix would result. However, the sighting task becomes complex. Two men, tracking different landmarks versus the same inertial reference and designating (by trigger) when their landmarks are centered in their sights, can obtain a series of such geometric fixes. However, more sophisticated curve fitting and computer filtering methods are applied to single star versus landmark sightings in the Apollo system, and these result in essentially the same information.

Table 4-2 shows characteristics of some typical star trackers.

Complexity. The sophisticated manual/optical/stellar (inertial systems, such as that designed for the Apollo CM, will eventually be replaced by somewhat simpler systems based upon strapped-down ESGs versus two-body sextants, which will permit a rapid string of geometric fixes to be obtained by one man. These will be filtered by a digital differential analyzer coupled to a general-purpose digital computer to permit calculation of a $\Delta \vec{V}$ at the next designated junction.

Manufacture	Identification	Total Field of View (degrees)	Instantaneous Field of View ($\min$)
Bendix	Proposal No. 9011-62-51	5	30
Bendix	OA0 Contract No. Nas 5-2108	--	--
I. T. T.	Type A	8 sq. Search 16 sq. Track	--
I. T. T.	Type B	--	--
Barnes	Image Dissector	4 x 10	60 x 600
Barnes	Photomultiplier	30 pitch 4 roll	300 pitch 18 roll
Honeywell	Proposal No. R-ED 25243	1 x 1 Star 10 Sun	--
Nortronics	Sun-Star	0.5 Star 1 Sun	--
Nortronics	Strapdown for Gemini and Apollo	4 - 0.167	--
Nortronics	Gimbale for Gemini and Apollo	2	10
Nortronics	Advanced System	1	5
Astronautics	Ungimbale	16 roll Search 1 el. 1 roll Track 16 el.	--
Aeronutronics	Ungimbale	1 roll 4 el.	30

Information Rate Out' (cps)	Head Accuracy (sec)	Type of Tracking	Axis Single or Double	Size (inch ³)	Weight (lb)
25	±9	--	--	95	3
--	5	--	Pitch ±60 deg. Roll ±60 deg.	--	--
5	5	--	--	196	9
10	30	--	--	155	5.7
1,000	360	Null Proportional	both	276.5	5
--	--	--	single	155.5	12
2.5 8 Sun	10 rms 3 Sun	--	double	101	7.5
400	10 rms	Null Seek	double	106	9
8	120	Null Seek and Prop. Read	double	280	9.5
8	120	--	double	700	23
8	60 to 2	Null Seek and Prop. Read	double	55.4	5.75
--	--	--	--	141	5
--	--	--	--	78.8	6

Reliability (MTBF)	Av. Power (watts)	Availability	Comments
--	5	--	Accuracy for stars brighter than $M = +3$
--	--	--	--
--	6.5 Search 7.9 Track	--	Accuracy for stars brighter than $M = +3$
--	4.5	--	Accuracy for stars brighter than $M = +2.5$
3 yr.	4	End of 1963	--
--	8	--	--
--	8	Approx. 1964	Accuracy for stars brighter than $M = +1.5$
31,000 hr.	8.5 for both	1964	--
13,000 hr.	12	1964	Accuracy for stars brighter than $M = +2.5$
13,000 hr.	18	1964	Accuracy for stars brighter than $M = +2.5$
31,500 hr. 2 sec System	10	1966	New detector and electronic scanning
--	4	--	--
--	3.25	--	--

TABLE 4-2. SOME TYPICAL
STAR TRACKERS

3

The romantic picture of the ancient space navigator with a hand-held sextant, sighting through a porthole, probably will remain fiction (although in principle space navigation can be performed this way). Complex equipment very likely will continue to be used.

Speed. The effectiveness of pure manual navigation is limited by the speed of manual calculations versus vehicle speeds. At several kilometers per second, there simply isn't much time for deep thought, except perhaps during planetary midcourse.

TABLE 4-3 TYPICAL ACCURACIES ACHIEVABLE

$t_{\text{smoothing}}$ = one orbit ($h = 150$ nautical miles)

$\Delta \theta_{\text{horizon scanner}} = 0.15$ degrees

$t_{\text{sampling}} = 2$ seconds

• (All errors 3σ)

$\Delta x = \pm 0.1$ nm

$\Delta V_x = \pm 1.6$ ft/sec

$\Delta y = \pm 0.1$ nm

$\Delta V_y = \pm 0.7$ ft/sec

$\Delta z = \pm 0.2$ nm

$\Delta V_z = \pm 0.6$ ft/sec

Accuracy. Table 4-3 shows some typical accuracies achievable in partly manual, self-contained navigation of a satellite. With completely manual methods, one would be indeed lucky to do a tenth as well. These accuracies are limited chiefly by horizon sensor error in local vertical determination.

4.2.5 Acceleration Sensor

In simple radio command guidance by DSIF the onboard inertial system is reduced to the SCS attitude gyros plus a simple inertial sensor and cutoff computer. The device senses thrust acceleration along the nominal thrust axis and generates a cut-off discrete (including consideration of a shut-down lag prediction) whenever actual, accumulated ΔV magnitude equals that of the commanded ΔV .

Reliability Acceleration Sensor. Failure of the acceleration sensor could lead to a "go for broke" firing of the rockets. This would send the spacecraft far astray instead of correcting its orbit. The acceleration sensor

might be backed up by a timer cut-off which limited overthrust to 2 or 3 percent of the required ΔV . Ideally, reliability of the system would be such that the timer back-up is unnecessary.

4.3 APPROACH

During approach to a terminal body it becomes necessary to measure relative vector velocity as well as relative vector position. To do this, the navigator must have range and range rate as well as lines of position versus real time. Upon the basis of this information, he must calculate his approach hyperbola and his ephemeris on it and decide upon a time and a ΔV to effect orbital capture, fly-by at a specified altitude, or direct atmospheric entry.

Ranges of present on-board radar devices are inadequate, and accuracies of planetary optical measurements are presently inadequate, for optimum approach guidance.

As yet, the best guidance available is by radar tracking from the Earth.

4.3.1 Planet Tracker

This device differs from a simple horizon sensor in two ways: (1) it tracks a variable size image from a near point source up to a 180 degree horizon at touchdown, and (2) it must obtain altitude during approach by measuring subtended angle of the apparent disk (or crescent) as well as establishing the line of sight to the planetary center.

Horizon Resolution. This is an accuracy problem whenever other than a truly homogeneous, spherical, airless body is being approached. The Earth is particularly difficult in this respect. Figure 4-2 shows errors in local vertical as a function of altitude with two values of horizon uncertainty (Δd).

Reliability. Since no operating planet trackers have been used for space navigation, their reliability is only a matter of analytic speculation about specific designs.

Many conceptual designs require spinning optics as well as sophisticated photoelectric sensors, some of which require cooling systems. Their MTBF ratings are unlikely to exceed 1,000 hours of active use in the next ten years.

R&D Required. Some fully electronic/optical concepts have been proposed within industrial organizations, but these ideas have not yet been developed, probably due to lack of a firm requirement. At any rate, the designs with no moving parts are the most promising for long term missions, if a firm need can be established early enough to permit their development.

Horizon Scanner. The term horizon scanner has been used to mean the local vertical reference function of a planet tracker. The technology deficiencies called out were:

Horizon Definition Mars. The smoothness, oblateness, cloud cover, atmosphere, etc., of Mars are very poorly described at present. The Mars orbiter mapper missions may improve this, but they will need horizon sensing for their attitude control. Conceivably, better information can be gained from telescopes in earth orbit.

Probably the resolution of Mars' horizon is better than that of the earth, but how much better is only a guess. Vertical accuracy probably will be worse than ± 0.05 degrees.

Horizon Definition Venus. Present thermal maps of Venus pose many more questions than they answer. It is not known whether IR comes from the surface or the atmosphere. The IR brilliance seems only slightly related to the terminator.

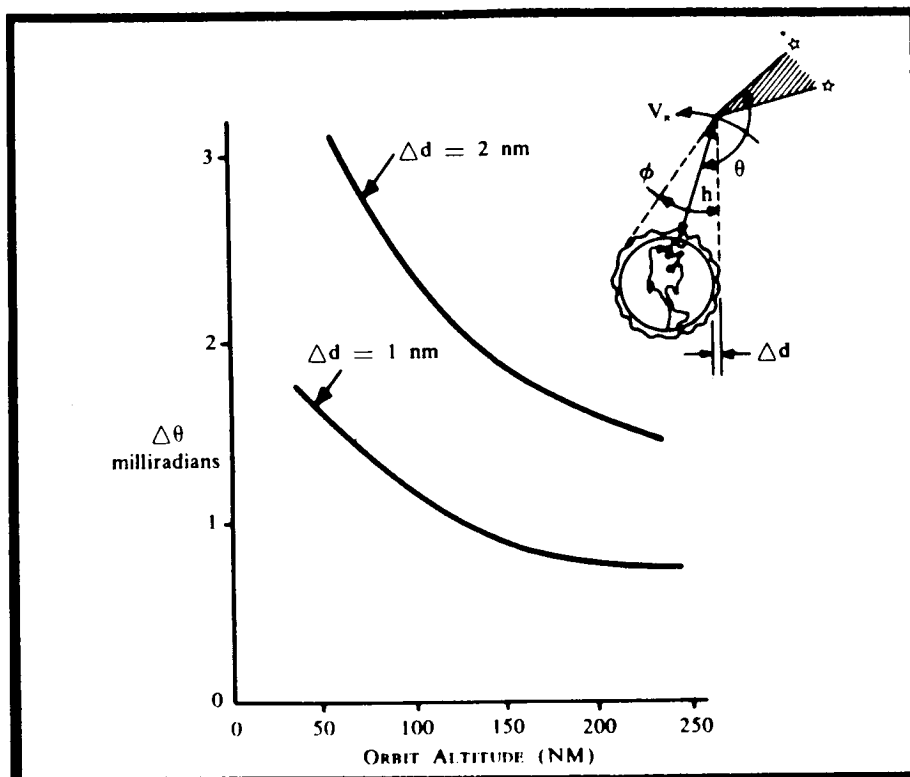
The horizon at Venus probably cannot be resolved in IR to better than ± 0.5 degrees, until much more is known about the planet's atmosphere dynamics.

Horizon Definition, Mercury. The close proximity of the sun may make IR horizon sensing impractical at Mercury, unless some sophisticated logic and sensor gating schemes are devised. The problem will require design of a special Mercury tracker. In addition, so little has been learned about Mercury's surface that we are even uncertain about its period of rotation.

Reliability. Since no operating planet trackers have been used for space navigation, their reliability is only a matter of analytic speculation about specific designs.

Many conceptual designs require spinning optics as well as sophisticated photoelectric sensors, some of which require cooling systems. Their MTBF ratings are unlikely to exceed 1,000 hours of active use in the next ten years.

1065-86A



Source: Jacques S. Gansler (Ref. 4.1)

FIGURE 4-2. ERROR IN MEASUREMENT OF LOCAL EARTH VERTICAL DUE TO EARTH/ATMOSPHERE IRREGULARITIES

4.3.2 Radio Command

Radio command guidance by DSIF is the only method so far used in our interplanetary and lunar flights for approach. It will likely continue to be the primary means for years to come.

Reliability. See paragraph 4.2.1.

Velocity Accuracy. See paragraph 4.2.1.

The accuracy of relative velocity depends largely upon the accuracy of the ephemeris of the planet being approached. Skin tracking of the planet is more difficult than tracking the transponder on the vehicle. The vehicle's track must be added to the planet's ephemeris to get relative position and velocity.

Signal Transit Time. At planetary distances, several minutes elapse between the time the vehicle moves and the time its motion is observed at DSIF. The ground based computer must take this into account in determining present motion. However, since the planetary ephemeris data is acquired with the same one-way lag, the relative motion difference is less than would at first appear. It is when commands for execution of action are sent that this propagation lag must be particularly noted (unless they are commands for future action to be triggered by the on-board clock).

4.3.3 Manual Optical

Control of approach by manual optical means is limited by the calculation speed of a human operator. With optics pickoffs and automatic computation, manual sighting and tracking of the planet versus an inertial reference can give reliable information in time for it to be useful. (Refer to paragraph 4.2.4)

Complexity. Refer to paragraph 4.2.4.

Speed. Refer to paragraph 4.2.4.

Accuracy. Refer to paragraph 4.2.4.

4.3.4 Doppler Radar

Planetary orbiter mappers will require altitude and velocity information for navigation as well as to improve the quality of measurements by image motion compensation.

R&D Required. Doppler radars useable above 30,000 meters have not yet been developed. The orbiters will need doppler data at several hundred kilometers. This involves several orders of magnitude improvement in output power, receiver sensitivity and frequency measurement accuracy.

Reliability, Doppler Radar. Without the doppler information, the data will be seriously deteriorated, but the mission will not necessarily be a failure. Reliability of the radar is important but not critical in planetary orbiter missions.

4.4 RENDEZVOUS AND DOCKING

Radar acquisition and tracking will be necessary to within a few hundred meters from the target. After this, the problem is primarily one of manual (seat of the pants) guidance.

4.4.1 Radar

Table 4-4 lists some typical rendezvous radars (accuracy of the supplied information is not confirmed).

Tracking Accuracy (Range Rate and Acceleration Accuracy). Refer to Table 4-4.

Power. See Table 4-4.

R&D Required, Radar. Upgrading of present Gemini/Apollo rendezvous radars is desirable, but not essential.

Reliability. See Table 4-4.

Improvement by a factor of 10 or more is expected through use of micro-electronics phased arrays instead of mechanical scanning antennas.

4.4.2 Laser

Laser ranging systems for rendezvous guidance have been designed and are being developed. In principle, ranges can be obtained plus or minus a few centimeters and range rates to plus or minus a few centimeters per second. Considerable R&D remains before such LIDARS, OPTARS, LASARS, etc., become operational in space.

Tracking Accuracy Laser. While range and range rate accuracies make laser devices attractive for rendezvous guidance, their beam widths are

Manufacturer and Name of Radar	Type	Range
★ Westinghouse Gemini Rendezvous Radar	Cooperative Present Radar/Trans- ponder	250n.m.
	Cooperative Future Radar/ transponder	500n.m.
* Space Tech. labs Space Tech. labs Tracking and Command (Stelatrac)	Cooperative Radar/ Transponder	500 n.m. * See Remarks
* Emerson	Cooperative Radar/ Transponder	400 n.m. (max) 100 ft (min) 110 n.m. for altimeter mode
	Search Mode Uncooperative Track Mode	50 n.m. (max) 50 n.m. (min) 50 n.m. (max) 30 ft (min)
* Raytheon Tri-Mode System	Cooperative Radar/Trans- ponder	200-400 n.m. (max) 5 ft (min)
	Uncooperative Rescue System	20 n.m.
	Hostile Mission system	
* Hughes Aircraft	Utility Radar Radar/Trans- ponder	200 n.m. (max) 99% (probability) of beacon acq. to 100 ft (min) with no modifications
	Pulsed Short Range Radar	500 ft (max) 5 ft (min)
	Doppler Velocity Sensor and	0 to 100,000 ft normal altitude 0 to 50,000 ft 45 degrees attitude
	Radar Altimeter	10 to 100,000 ft normal attitude

Remarks: * Additional Literature Available

★ Secret Information Available on Uncooperative

Range Resolution and/or Accuracy	Angular Coverage	Angular Resolution and/or Accuracy	Range Rate
1 % or $\pm 50'$ Digital 5 % or 50' analog	50 degrees solid cone	3 mr	obtained from differentiation of analog range
1 % or $\pm 15'$	50 degrees solid cone	3 mr	same as above
± 0.04 % $\pm 2'$ (36) Accuracy Resolution Limited only by processing technique employed	Hemispherical at Radar and Transponder	± 3 mr (30) Accuracy 10 min Resolution	-200 ft/sec to 6000 ft/sec * see remarks
5 % of 1000 ft	Hemispherical		
50 n.m. to 1000' crude 1000' to 30' ± 5 % or ± 2 ft	AZ ± 25 degrees EL ± 25 degrees ± 25 degrees cone	1 degree to 3 degrees accuracy	200 ft/sec (max)
± 2 ft ?		0.3 mr	
± 2 ft (at close range)	± 60 degrees each plane	0.5 mr at 1000'	
± 20 ft or ± 0.01 % Range plus Transponder Error	AZ ± 70 degrees EL ± 70 degrees	0.8 mr (rms) Accuracy	
± 2 ft or ± 1 %			
			0 to 3000 ft/sec
Low Range ± 4 ft ± 1 % (35) High Range ± 40 ft ± 5 % (30)			0 to 1000 ft/sec

RENDEZVOUS

RADAR

Range Rate Accuracy	Angular Rate	Angular Rate Accuracy	Peak Power (Transmitted)	Average Power (Transmitted)
± 5 degrees or 1 ft/sec	Angular rate is not sensed -- Can be obtained from angle data		1150 watts	0.29 watts
± 0.1 ft/sec	Same as above		1150 watts	0.115 watts
± 0.1 ft/sec	1 rad/sec	± 0.3 mr/sec Predominantly bias error	0.1 watts can increase to 0.5 watts or greater	0.1 watts use phase modulation
				Radar 0.5 watts Transponder 0.1 watts
	1 degree/sec, to 2 degrees/sec	0.2 mr/sec		
0.1 ft/sec				0.5 watts
0.1 ft/sec				7 watts
				70 watts
	1 degree/sec	± 0.5 mr/sec	2000 watts	
			1000 watts	
± 1 ft/sec plus 2 degrees (30)				6 watts (2 watts/beam)

Weight	Size	Reliability MTBF %	Availability
68 lbs.	1.75 cu. ft.	Transponder receiver 10,000 Hrs. Trans- ponder Trans. 2000 Hrs Radar 1000 Hrs.	In production
29.6 lbs.	1.25 cu. ft.	Transponder receiver 19,000 Hrs. Trans- ponder Trans. 13,000 Radar 6,000 Hrs.	
Radar 45 lbs. Transponder 14 lbs.	Radar 0.43 cu. ft. Transponder 0.34 cu. ft.	Transponder, Radar Data Processor 9200 Hrs. Antenna 16,400 hrs.	Prototypes within one year
Radar 32.8 lbs Transponder u. 1 lbs.	Radar (less amt) 0.51 cu. ft. Transponder 0.098 cu. ft.	- 99977 for 8.4 HR. mission	18 months
100 lbs.			2 years
Radar 20 lbs. Transponder 6 lbs.		Radar 13,000 Hrs. Transponder 30,000 Hrs.	
40 lbs. including antenna		10,000 Hrs.	
45 lbs. including antenna	Radar 1 to 1.5 cu. ft. including antenna	13,000 Hrs. (Station Keeplog) 3,000 Hrs. (acquisition)	
39 lbs			
26 lbs.			

System Functional Characteristics

Measures Range, Range Rate and Angle to Transponder.

Measures Range, Range Rate and Angle to Transponder.

Range is determined by a swept sideband and range rate by cycle counting. The radar antenna is gimballed although the system has been experimental in configuration angle and angle rate is measured by angle transducers and frequency for the gimballed antenna and by relative phase and frequency for the beam. The system presented for a high accuracy config. can be adapted to perform functions of 1.

Modulation - FM/CW K_u band antenna - 2 ft. dia. Single shot.

Target Size - 1 square meter 28 P.

Modulation - FM/CW X-band solid state Antenna - 12" diameter.

Antennas - 2 elliptical antennas - 2 ft diameter. Assumed target size - 40 sq

Assumed target size - 1 square meter.

Transmitter - limited tune pulsed magnetron modulator - silicon controlled rectifier array, diameter equals 20 ft. Beam width 45 degrees. Signal Processing ranging.

Transmitter - Grid pulsed triode oscillator. Antennas - Separate fixed receive diameter equals 12 feet, Beam width (3db) equals 30 degrees, Target size equals

Antenna - Split paraboloid, diameter equals 18 ft. Beamwidth (2 way) 4 degrees

	Frequency CPS Band Type	Total Power (Watts)
he system is coherent. luted in an interferometer ency of pulse torquing, respectively ted antenna. The specifications are or vol. & alt.	5.2×10^9 to X 10.9×10^9	
	16×10^9 Ku FM - CW	Radar 97 watts Transponder 30 watts
	Pulse	
	5.2×10^9 to X 10.9×10^9 FM-CW	50 watts including servo no-load loss
are meters.		150 watts excluding servo power
		150 watts excluding servo power
ctifier - magnetic Antenna - Gimbale ing-Fixed threshold-digital	9.3×10^9 X Pulse	116 watts at 28 Voc
e and Transmit, slotted arrays, als 1ft. 12.	2.5×10^9 S Pulse	
es.	13.9×10^9 K CW (3 Beam)	450 watts
	12.9×10^9 K FM - CW	

TABLE 4-4. SOME TYPICAL
RADARS

so sharp that acquisition and tracking difficulties have been encountered. Lobing methods for AM/FM scan tracking have yet to be reduced to practical systems.

Power. Pulsable, CW lasers are essential. Present lasers of this type have efficiencies less than 15 percent and powers less than 40 watts although experimental lasers have been operated CW at as high as 150 watts output. R&D is needed.

R&D Required. See paragraph 4.5.6.

Reliability. There is insufficient data as yet. Experimental units have displayed disappointingly short MTBFs. However, in principle, systems with MTBFs of many years should be possible.

4.4.3 Manual Optical

So far the best method for terminal guidance to rendezvous seems to be by manual optical (seat of the pants) under control of a well-trained pilot. Simulators for this situation have been built for Gemini and Apollo training.

Complexity. The required motions are complex, but with training should become habit. The equipment is extremely simple and centers around target displays to give the pilot good visual clues of range and relative bearing.

Speed. Manual docking should not begin until relative velocity has been reduced to less than 20 meters per second. Velocity within one meter of the dock should be less than two centimeters per second to prevent vehicle damage.

This allows adequate time for human decision, mistakes and corrections.

At initial acquisition speeds, manual control should not be attempted (the Russian rendezvous attempts demonstrated this. During the second editing of this paper the successful rendezvous of Gemini 6 with Gemini 7 - using the radar and computer system - had just been announced. CWB).

4.5 ILS TERMINAL GUIDANCE

The guidance to final touchdown at a particular surface position requires some destination oriented navigation aid. The lunar projects very likely will use some sort of simple beacon transponder to assist astronauts in landing within easy walking distance of previous payloads on the moon.

Updated versions of existing instrument landing systems (ILS) may be employed to guide the final touchdown of the orbital ferry vehicles, used in concept EO08 W, after their return from the space station.

4.5.1 Acquisition and Vectoring, ILS

In order to use ILS at all, the vehicle must enter the glide slope and localizer beams over the outer marker. To get to this gross position probably is not practical without assistance of ground radar, Decca, Loran or some similar aircraft navigation system. The hardware for acquisition involves a fixed weight payload penalty throughout the mission.

4.5.2 Auto Beam Coupling, ILS

It would be desirable to have landing approach performed by the autopilot during inclement weather. To do this requires coupling the ILS receivers through logic circuits to the SCS computer. Some R&D is required to instrument the automatic approach.

4.5.3 Velocity Accuracy, ILS

ILS does not yield a velocity signal; therefore, air speed during approach must be supplied by another sensor and controlled by throttling through logic circuits so that the correct velocity is achieved at touchdown. Some R&D will be required to integrate the ILS autobeam, air speed indicator (or doppler radar) with INS and SCS in such a way that automatic landing of even an unmanned logistics vehicle could be accomplished safely in inclement weather.

4.5.4 Radar Altimeter

During unmanned soft landings on the moon, accurate altitude data is essential. The failure of four Russian attempts has been nominally blamed upon inaccurate altitude data. There may be a tenuous lunar ionosphere (close to the surface) which produces false returns for radar.

Reliability, Radar Altimeters. Without accurate altitude data the mission probably will end in a crash. If the retro rockets fire too early, hover at high altitude will be followed by a straight drop crash. If the retro rockets wait too long to fire braking will be inadequate and impact occurs. Since the payloads of the LLV are vital to the lunar bases, the radar altimeter must be regarded as a critical item.

R&D Required, Radar Altimeters. Very probably unmanned test flights involving comparison of measurements at several frequencies will be required to resolve the uncertainties about the lunar surface and the meaning of radar returns.

The radar should have an altitude range in excess of 50 miles, yet be useable at 50 feet.

4.5.5 Approach TV

While landing on the moon or rendezvousing with a near earth asteroid, the control of an unmanned probe might be improved by closed-loop TV during "touchdown or approach." Such TV also can help in bringing unmanned objects together in earth orbit.

The specific need during lunar landing is to prevent setdown on a steep slope or edge of a hole and to verify the surface features near the landing site.

The control loop to the moon will involve a time lag or "backlash" of about 2 1/2 seconds.

R&D Required, Approach TV. The present state-of-the-art is exemplified by the Ranger TV. A somewhat more rapid frame rate is desirable for real time remote control of landing.

Resolution. The more detail and sharpness in the pictures the better the remote pilot's judgment. Ideally a 5000 line resolution could be used, but 525 would be acceptable.

4.5.6 Laser Planet Ranging

Altitude and altitude rate (at altitudes greater than 100,000 miles) relative to the target planet are essential, if safe entry and landing are to be made. Radar equipment with adequate microwave power output is likely to be excessively large and heavy. Laser ranging, on the other hand, shows promise of effective operation at great distances with lower power output, due to its greater directivity.

Reliability, Laser. Insufficient experience exists to permit estimating the reliability of laser rangars. See paragraph 4.4.2.

R&D Required, Laser. Small ranging sets have been built and larger ones are possible in principle, but progress is needed in laser efficiency, power output, modulation and beam scanning control. In addition, some thought is required to design the proper spaceborne packaging.

4.5.7 Heading Reference and Scanning Radar

R&D Required, Heading Reference, Mars. Since the rotation of Mars is about the same as that of the earth, gyrocompassing should be feasible as a surface heading reference. This can be backed up by a Zenith TV camera. (Ref. 4.2).

R&D Required, Heading Reference, Venus. Since Venus rotates very slowly if at all, has heavy cloud cover, and probably has no magnetic field the problem of a surface heading reference is almost completely "blue sky" and a major breakthrough will be required. Possibly IR sun sights can be made from the surface to very gross accuracy. An ESG pre-aligned in Venus orbit might serve for several weeks on the surface.

R&D Required, Scanning Radar. The problem is chiefly one of providing proper spaceborne packaging for present, small search radars.

REFERENCE LIST FOR SECTION 4

- 4.1 Jacques S. Gansler, "Space Navigation," TNB (General Precision, Inc. Technical News Bulletin), Vol. 8, No. 1, 1st Quarter 1965.
- 4.2 Charles W. Benfield, "Study to Estimate the Guidance and Navigation Subsystem for the Lunar Roving Vehicle," Honeywell R-ED 25223, 15 November 1962.

Section 5

LIFE SUPPORT AND ENVIRONMENTAL CONTROL

5.0 GENERAL COMMENT

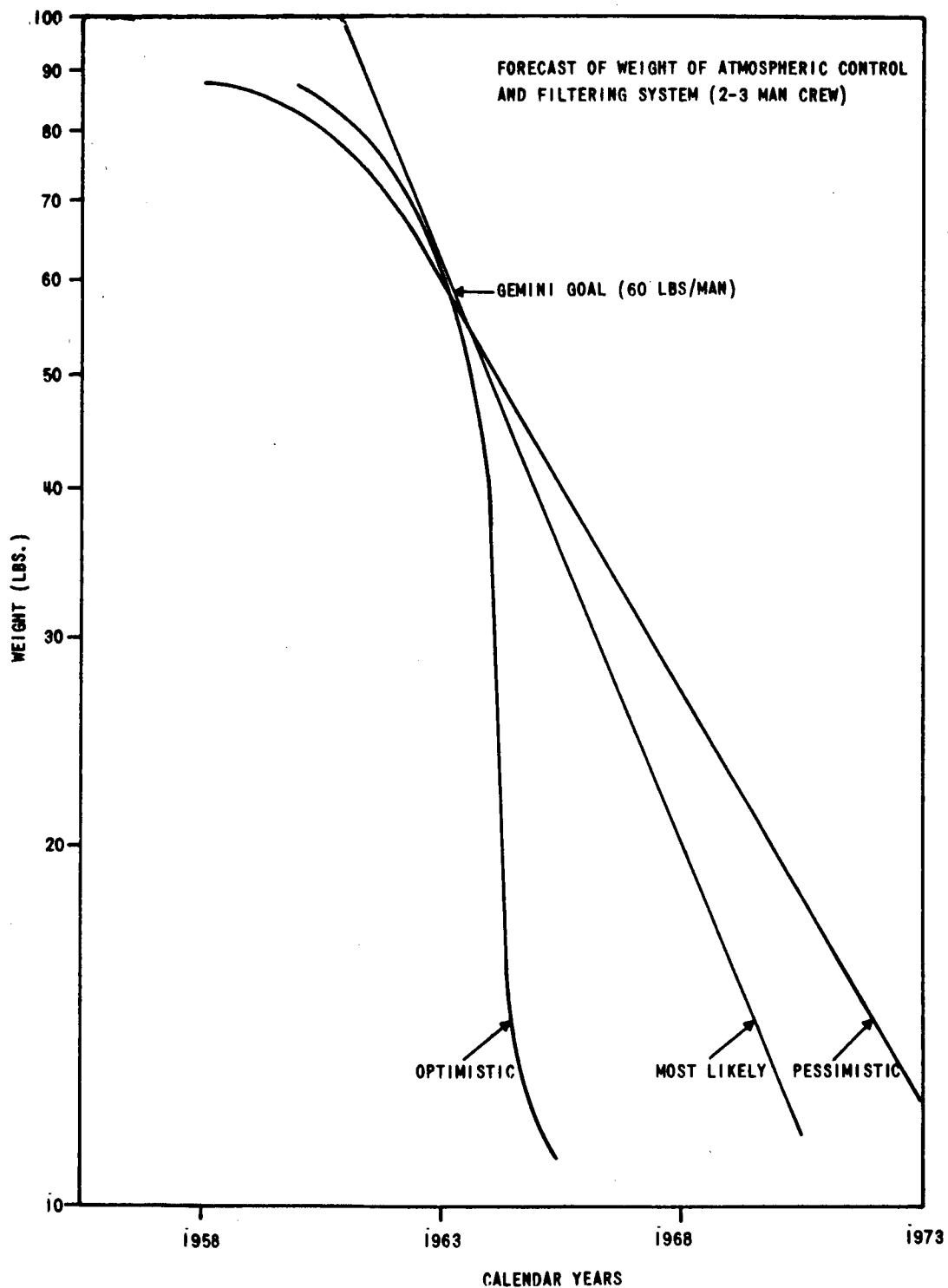
Life support includes all equipment functions needed to sustain human life and to maintain crew functional proficiency. This includes vehicle environmental control, ecological systems and skill retention devices, and extra-vehicular personal equipment.

Prior to 1957 very little direct effort had been devoted to the design of life support equipment for spacecraft. Fortunately, a considerable background of life support technology from submarines, diving equipment, and high altitude aircraft existed. As a consequence, the Mercury life support system and space suit was developed in a remarkably short time. The new technology has come to the point where no insurmountable problems appear. Astronaut White's walk in space and the associated eight-day Gemini flight have demonstrated a high probability that man can survive a trip to the Moon and back.

Generation and control of oxygen for man's environment in the aerospace system differs according to mission length. For short duration flights current technology (using compressed liquids, supercritical storage, or two-phase storage plus vapor delivery) requires system weights of 250 to 300 pounds, or about 100 pounds per man. Savings in heating coils (now about 40 pounds) are expected to result in the decrease noted in Figure 5-1.

5.1 NITROGEN/OXYGEN AIR

Use of a so-called normal, two-gas atmosphere introduces the possibility of serious bubble formation in the blood (bends) whenever sudden depressurization occurs. This may be fatal, unless repressurization is immediate. The difficulty can be reduced by using either pure oxygen or an inert gas as a substitute for nitrogen.



Internal Source Unidentified

FIGURE 5-1. PROJECTED LIFE SUPPORT SYSTEM WEIGHT

5.1.1 Contamination Detection

Some contaminants are self-revealing; others are much more subtle. Chemical detectors exist for some well known industrial gases as a result of chemical process leak detection needs. For other gases no easily instrumented detection system has been devised.

The human sense of smell gradually adapts to an ever present odor, and learns to ignore it, even though it may represent a hazard. Therefore, development work is needed for automatic, quantitative detection of those contaminant gases which can exist within the sealed cabin.

Contaminants other than gases, such as spores, pollen, dust, bacteria, smoke, colloidally suspended liquids, etc., also can present a hazard. The need for automatic detection methods may become critical for long manned missions. Water vapor also can be a contaminant.

5.1.2 Odor and Contaminant Removal

Trash and crumbs can be removed by "vacuum cleaner" strainer systems with periodic washing of the strainers and dump of the wash water overboard.

Electric precipitators can remove most colloidally suspended particles, but may introduce dangerous quantities of ozone (O₃) into the atmosphere.

Wash chambers can remove water soluble gases and refrigerating dehumidifiers can reduce the humidity created by the washers.

However, long term simulation of the conditions within a space vehicle will be needed before committing men to space stations or interplanetary flights. Hopefully, this will reveal the overlooked contaminants (which may exude from structures) and should suggest removal methods.

5.1.3 Leak Detection and Alarm

On short Earth orbital flights, minor leakage is made up from the stored gas reserves. However, during long missions in space, even minor leaks can become expensive and can even have an effect upon guidance and navigation by creating random microthrusters.

Whenever a gas expands it must take up heat; therefore, it seems reasonable to assume that the vehicle skin near a leak would be cooler than ambient. It should appear so when viewed with an IR image tube.

The sooner a leak is found and sealed over, the less danger it represents. However, leak alarm systems remain to be developed.

5.1.4 CO₂ and CO Monitor, Alarm, Removal

Regenerable CO₂ removal systems have been developed to the point that flightworthy hardware should be available early in this forecast period (about 1968). Tehcnology projections indicate that a reliable oxygen reclamation system may be available toward the middle part of the period (1975).

CO might be removed by a catalytic oxidation process.

5.1.5 Reclamation of Breathing Oxygen. Recovery of oxygen from CO₂ is more of a problem, requiring an endothermic reduction in the presence of a catalyst. Semipermeable membranes may exist which can pass CO but not CO₂. In any event, R&D is indicated.

5.2 INERT GAS ATMOSPHERE

In certain work spaces, it may be desirable to maintain an inert gas atmosphere to permit shirt-sleeve operation by personnel, using masks for breathing oxygen. This would permit welding and other operations that might produce toxic gases or explosive conditions in an oxygen atmosphere. Helium is considered the most likely choice for this atmosphere, but considerable research is required to evaluate possible deleterious effects of working in such an environment.

5.2.1 Leak Detection and Alarm

See paragraph 5.1.3.

5.2.2 Contaminant Detection and Removal

See paragraphs 5.1.1 and 5.1.2.

5.2.3 Recovery of Breathing Oxygen

See paragraphs 5.1.4 and 5.1.5.

5.2.4 CO₂ Removal from Helium Atmosphere

Speculation:

Conceivably some difficulty may be encountered with absorbent systems in that helium may be absorbed as readily as CO₂ to some agents. There seems to be no serious obstacle to progress here, but the CO₂ removal unit may require some research to adapt it for use in a helium atmosphere.

5.2.5 Odor and Contaminant Removal. See paragraph 5.1.2.

5.3 CLOSED ECOLOGICAL SYSTEM WITH BOTANICAL OXYGEN SOURCE

The use of algae for closed-cycle atmospheric systems may become popular for long duration missions. Man consumes one-third of his own volume (about one cubic foot) of O₂ per hour in respiration. Chlorella can photosynthetically produce 100 times their own cell volume of O₂ per hour. Therefore, optimum design would have a volume 1/300 of the volume of the men involved. The weights of these systems is far below gaseous storage techniques. These figures and the associated graphs are based on thermophilic strains of Chlorella which have a maximum logarithmic growth 3.5 times that of ordinary strains. Figures 5-2 and 5-3 provide data of forecasted algae system performance and requirements.

5.3.1 Leak Detection

See paragraph 5.1.3.

5.3.2 CO₂ and CO Monitor, Alarm, Removal

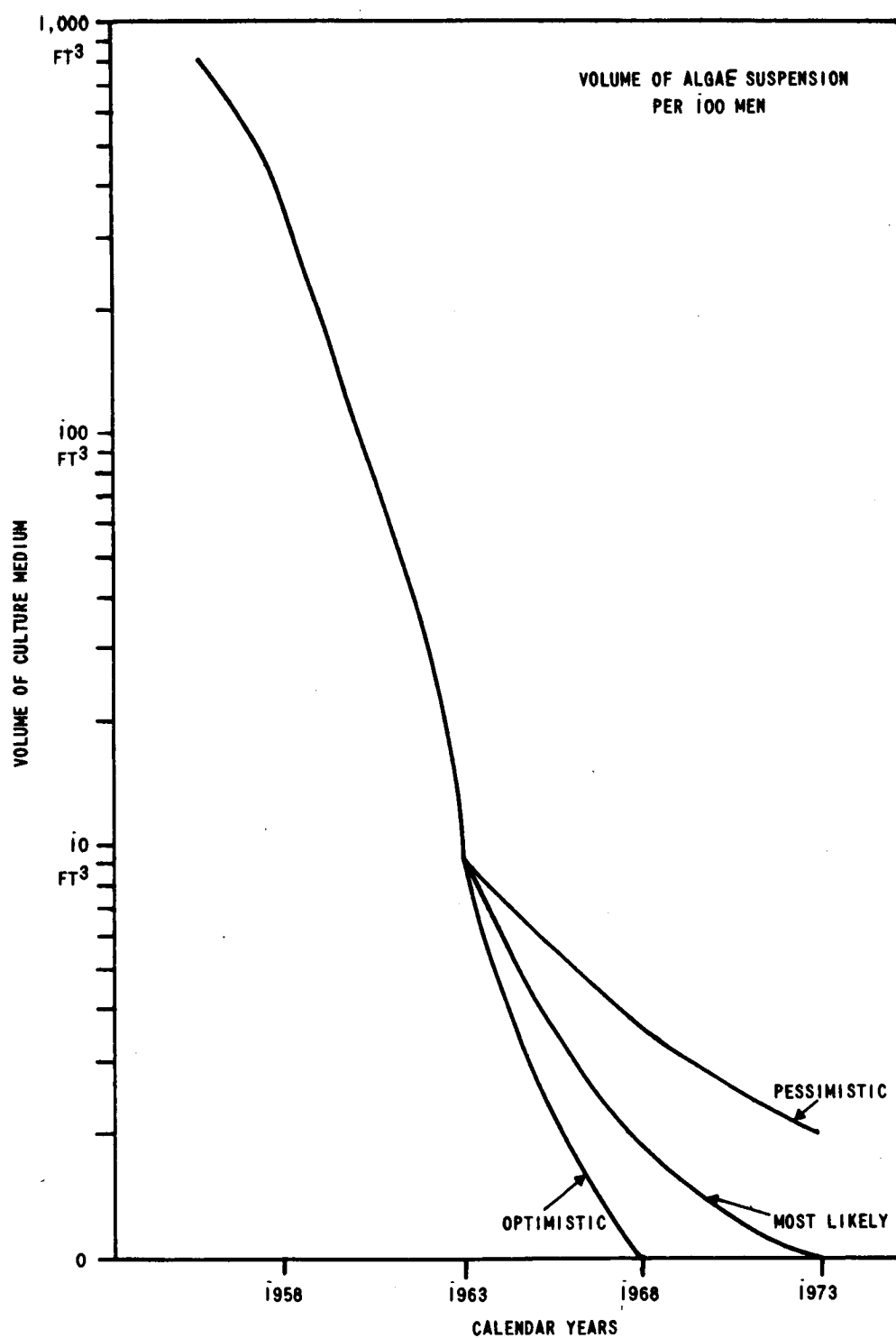
See paragraph 5.1.4.

5.3.3 Contaminant Detection

See paragraph 5.1.1.

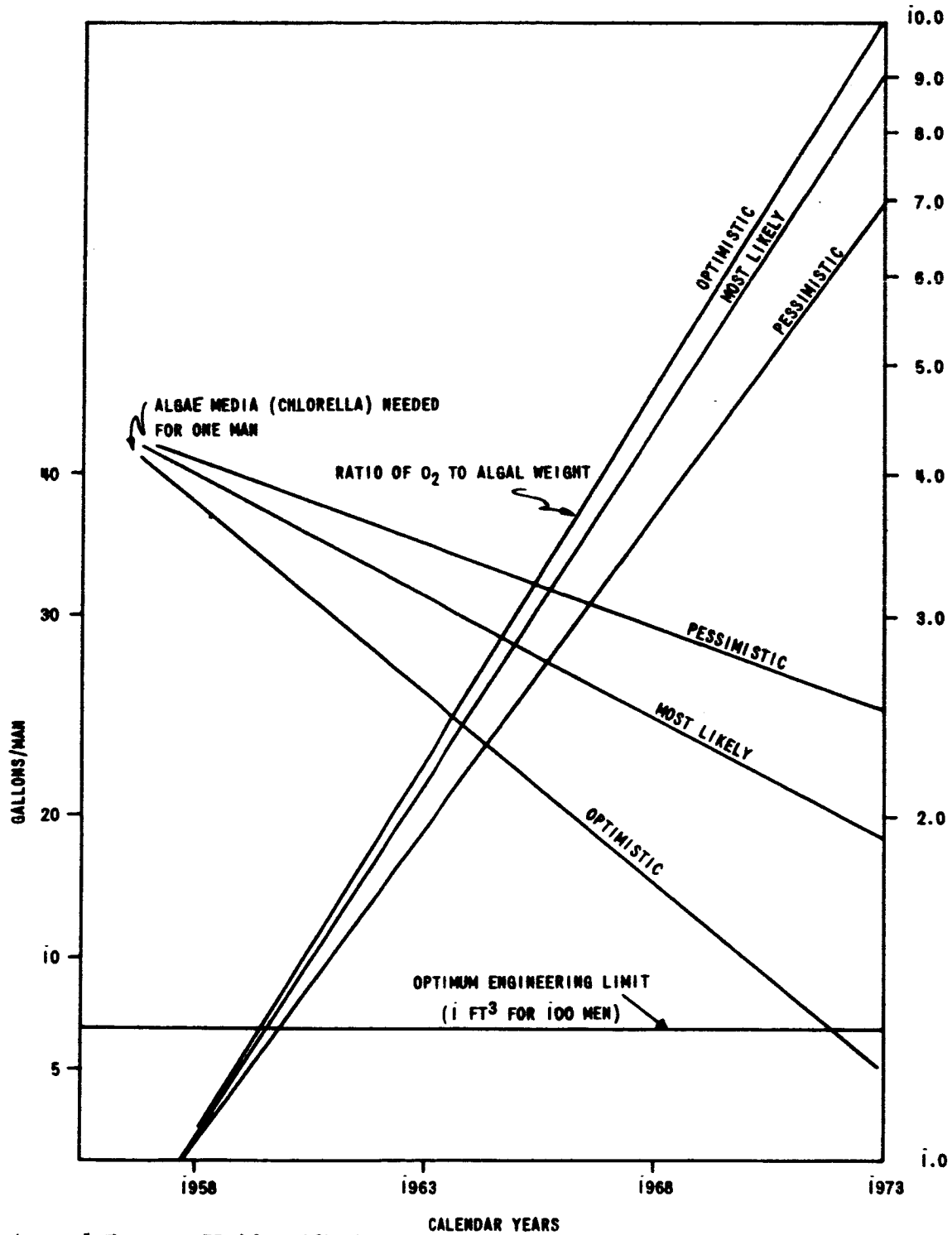
5.3.4 Odor and Contaminant Removal (See paragraph 5.1.2)

Some odors will be generated by the slime in which the algae exist. Conceivably such odoriferous compounds can be oxidized by passing air over an incandescent heater coil with subsequent electrostatic precipitation of the resulting oxides, and return to the algae tanks.



Internal Source Unidentified

FIGURE 5-2. ECOLOGICAL SYSTEM CULTURE VOLUME FORECAST



Internal Source Unidentified

FIGURE 5-3 FORECAST OF ECOLOGICAL SYSTEM DEVELOPMENT

5.3.5 Life Support System Integration

R&D is required to optimize design and avoid unwanted sloshing of algae cultures.

Proper illumination of algae may become a problem, since they tend to cling to any lighted surface and eventually make it opaque.

5.4 TANKED WATER

Tanked water has been used in man and animal flights up to the present. For short-term missions, this is the preferred method; however, as mission times increase beyond a week or two a serious payload penalty exists and tanked water becomes a luxury.

5.4.1 Ion Removal and pH Control (Tanked Water)

This is simply accomplished by sanitary storage of pure water in clean, non-reactive containers initially. (No problem exists.)

5.4.2 Bacteria Destruction (Tanked Water)

No problem exists if sterile water is stored in sterile tanks. Small amounts of a tetracycline compound might maintain sterility without serious physiological effects.

5.5 FUEL CELL WATER

On missions up to one or two months, fuel cells are the most likely auxiliary power source. They operate by catalytic oxidation of pure hydrogen in a potassium hydroxide electrolyte. The results are electricity and water. However, the water is contaminated with a deadly caustic (potassium hydroxide). This might be neutralized with carbonic acid (H_2CO_3) to produce a relatively harmless potassium carbonate, getting rid of CO_2 in the bargain.

Water with a pH of seven is not the same as de-ionized water. Preferably, the excess fuel cell water would be completely de-ionized by distillation. Conceivably, the fuel cell input gases could serve to cool the condenser.

5.5.1 Psychological

It would require great faith to partake of neutral water produced by simple automatic pH control through a balance of deadly poisons. There is less reason to fear water produced by distillation of the KOH solution; however, ask yourself how you would feel about drinking a glass of water which you know had contained a lethal dose of red devil lye less than an hour ago.

Present fuel cell systems do produce "potable" water, but some problem may occur with trace contamination however carefully the de-ionization process is handled.

5.5.2 Ion Removal and pH Control

R&D are required to assure potability and safety of fuel cell water.

5.6 RECYCLED WATER SYSTEMS

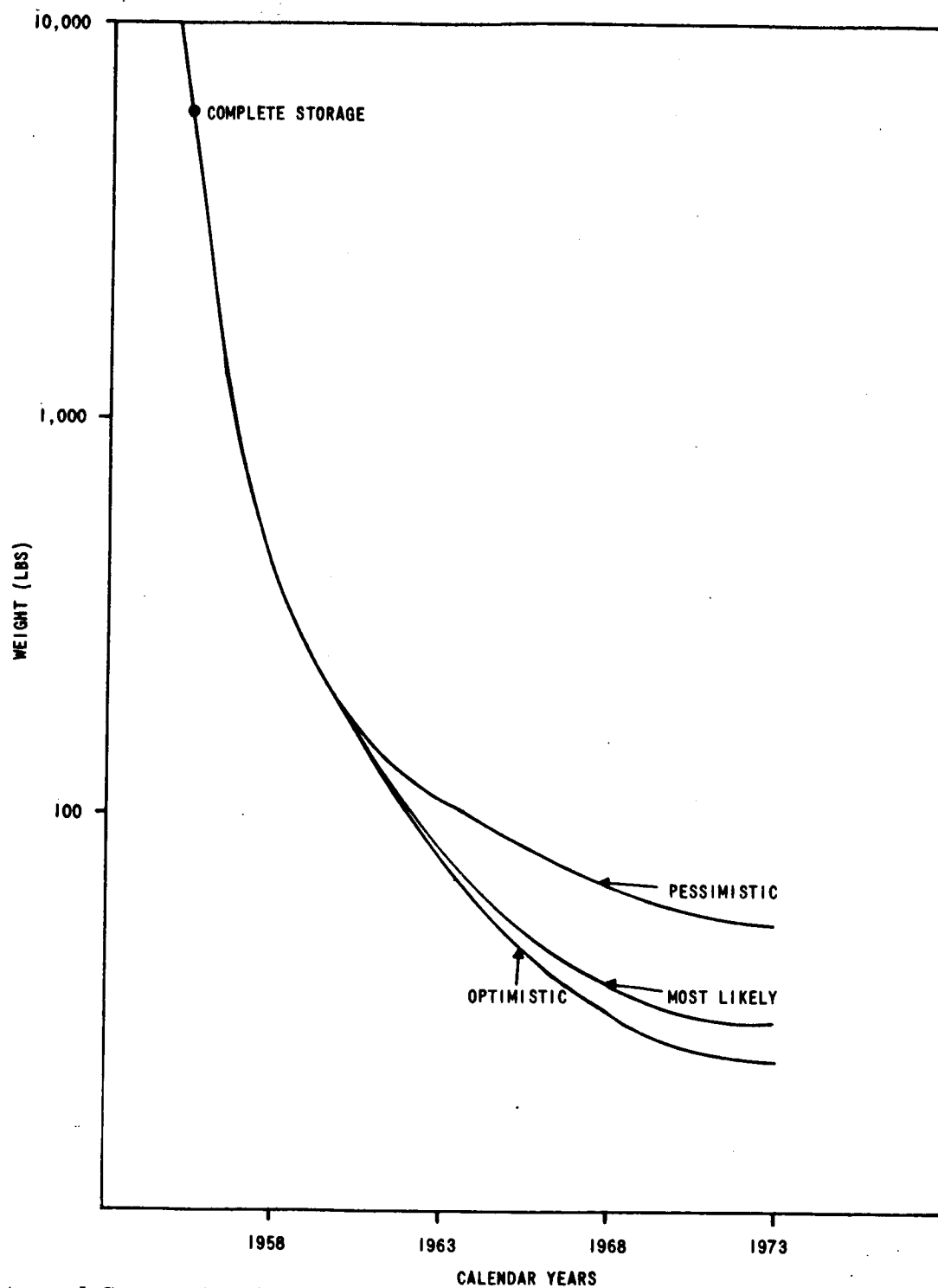
Food, water and waste disposal represent prime considerations in space vehicle man sustenance systems. A considerable savings in system weight occurs on long duration missions once storage is replaced by reclamation (closed versus open cycle). Currently, these techniques are in early development and considerable progress can be expected.

Supply of water to the manned spacecraft is shown in Figure 5-4 to rapidly decrease as technology in reclamation improves. This trend is also shown in Figure 5-5 where limits are placed on system with and without pre- and post-H₂O recovery.

5.6.1 Psychological Implications

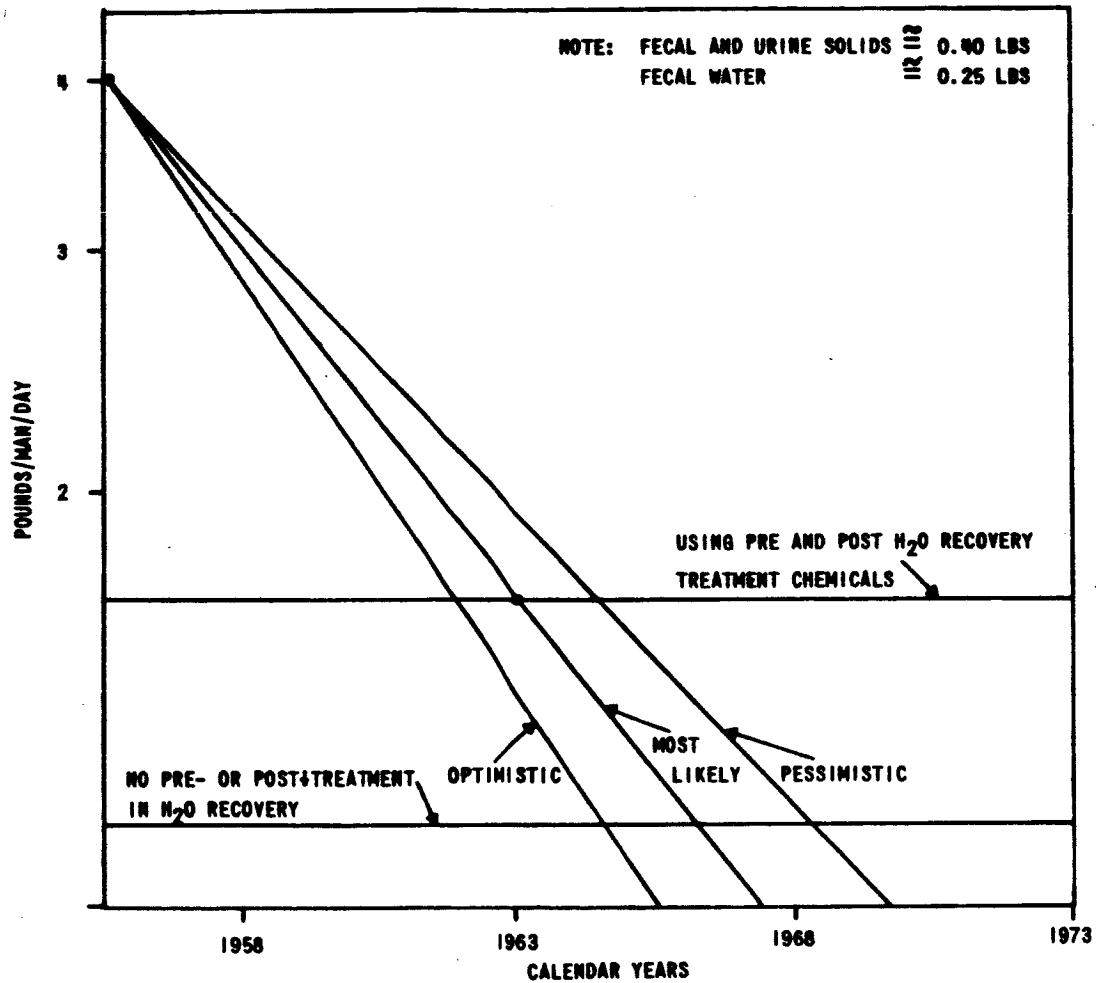
Man has lived all his life in a closed ecological system (the Earth). He sometimes may philosophically reflect that every breath he takes has been breathed before. His every drink has been drunk before (by many animals and men). Most molecules of his fresh food may have been eaten many times before. Somehow this doesn't concern him, because of the anonymity and remoteness of the previous users of his present sustenance.

However, confront a man with the prospect of having nothing to drink but water reclaimed from his neighbor's cesspool and he may rebel at the thought. Whether logic and necessity can successfully prevail over emotion is an open question, as yet. If the adjustment cannot be made by astronauts, the payload penalty in tanked water will become a serious problem (see Figure 5-4, "complete storage" point).



Internal Source Unidentified

FIGURE 5-4. FORECAST OF WEIGHT OF WATER SUPPLY SYSTEM FOR 1,000 MAN-DAYS (6-MAN CREW)



Internal Source Unidentified

FIGURE 5-5. FORECAST OF WEIGHT OF DISPOSAL HUMAN WASTE QUANTITIES: FECAL MATERIAL, FOOD PACKAGING, RESIDUE AND EXPENDABLE SUPPLIES STORAGE

5.6.2 Ion Removal and pH Control

Simple distillation followed by ion exchange units to precipitate volatiles may suffice for sterilization and potability control of reclaimed water. Re-generation of the ion exchange media may become a problem. This entire subject is under study.

5.6.3 Trace Contaminants

Despite filtering, distillation and ion removal, some contaminants may be overlooked and not affected by the processes. Progressive buildup of concentration with time might result in mysterious physiological disorders. Long term simulated runs of closed systems are required to evaluate the potential dangers of such things as organic polymers and metallic compounds in solution. The dead bodies of bacteria and animalculae can act as trace contaminants.

5.6.4 Bacteria Destruction

The heat of distillation usually can be relied upon for sterilization at normal atmospheric pressures. However, low temperature distillation is possible at low pressure and may be considered by designers. In such cases a superheater should be followed by a cooler prior to tankage of reclaimed water. Bacteria destruction by chlorination is not recommended.

5.7 STORED FOOD

Most likely all of the manned missions included in the present study would depend primarily upon stored foods. In zero-g, the packaging is almost entirely limited to squeeze tubes, although some solid foods of sticky consistency may be tolerated. Gemini and MOL results may permit some variety to be introduced.

5.7.1 Monotony

Experimental animals have survived indefinitely when limited to the identical menu, meal after meal. They also seem not to resent the limitation. With man, eating has become somewhat of a goal rather than a necessity. His morale suffers when variety goes out of his life. There may be some serious morale effects as a result of surviving upon artificially flavored "goo" from squeeze tubes for months. Controlled, long term experiments, using prison inmate volunteers, might shed some light upon this doubtful area. Some nutritionists claim that occasional "live" foods (such as raw

vegetables and fruits) are essential to health. Knowledge of human total nutritional requirements is presently quite meager. Whenever artificial gravity is available, the crew might vary their diet with ordinary canned groceries. (This is yet another reason for providing spin.)

5. 7. 2 Waste Disposal

Previous manned flights have stored fecal matter and empty food containers. Transfer of such material to the stage which will next be separated and abandoned may be feasible. However, useless mass must have its freight paid for in propellant at the next burn. It would be preferable to discharge it overboard. No garbage guns have been designed because the problem is trivial on short flights. However, on a multi-manned Mars mission the propellant saving could be significant and the problem deserves further study.

5. 7. 3 Preservation (Of Foods)

Squeeze tube foods can be kept under refrigeration for several weeks. However, for long missions, refrigeration of large cargos of groceries represents a waste heat rejection problem. Hermetic canning of a day's ration of squeeze tubes may be a preferable means of preservation.

Research in sterilization of foodstuffs with high energy radiation may provide means for ambient temperature storage of sealed squeeze tubes.

5.8 CLOSED ECOLOGICAL SYSTEM WITH BOTANICAL FOOD SOURCE

In adapting himself to the harsh environments of other planets, it is quite probable that man will depend upon small models of the closed ecology of the Earth. Atmosphere regeneration and bulk food may be provided by plants, which feed from waste material and solar energy. These botanical life support systems probably will continue until the process of photosynthesis is sufficiently well understood to permit construction and operation of photosynthetic chemical processing machinery.

Algae or Chlorella, being primitive living creatures, are difficult to control. Chlorella tend to stick to any illuminated surface, and are not clean growing organisms. As a source of food they are generally rather unappetizing, especially when harvested from tanks of human waste. However, atmosphere regeneration systems based upon algae have been operated. Better plants than algae are needed, but an even greater need is for hardware systems which will use solar energy to reduce CO and CO₂ into amorphous carbon and breathable oxygen.

The first approach to better plants is selection of the best algae for man-compatible atmosphere regenerators.

In the long term, the lunar and planetary base structures are visualized as transparent, pressurized, self-sealing domes. These domes will be insulated with a dead air space to augment the greenhouse heating effect. Within the domes, in properly treated soil, the colonists will grow the same strains of food vegetables to which they were accustomed at home. However, being a small scale model of the Earth, these domes will not have the systemic inertia of the great blue planet, and could be catastrophically upset by such factors as too little CO₂ in the air, too much oxygen, too little nitrogen, etc. Conceivably, an automatic hardware control system will be needed to back up the closed ecology in order to see it through small emergencies.

Food Production. For the next 15 to 20 years, food for space flight very likely will be transported as part of the payload. Until space cargo transportation costs are drastically reduced, this supply mode will put food costs in the hundreds of dollars per man-day in space. Early explorers will tolerate a certain amount of privation or even short-term starvation, but establishment of useful bases upon other planets probably requires either agriculture and synthetic food production or high volume logistic support.

Space flight it not the only motive for upgrading our technology in food production. The explosive population growth on our native planet will make

these problems severe soon enough. However, space flight does provide an early motivation for research into more efficient food production methods.

It is highly doubtful that palatable (or even safe) natural foods will be discovered on the other planets.

Waste Processing. A primary consideration is removal and purification of water from waste material. In closed ecology systems, the remaining solids will serve as plant food, but in the near future systems such solid waste must either be stored or ejected. It is expected that in the distant future nuclear rockets will be able to utilize a slurry of solid waste as reaction mass.

Contamination of Planets. It is almost inevitable that when man ventures upon another world he will thereby transfer to that world billions of Earth creatures in the form of viri, fungi, bacteria, protozoans, algae and possibly higher life forms such as lice, fleas, cockroaches, fruit flies, mushroom spores and weeds. A few of these may survive the rigors of the new environment, adapt to it and prosper. Very possibly, early Mars expeditions will carry a variety of hardy grains, grasses, and vegetable seeds to be planted in the most likely spots, where these earthly invaders then will join the battle for survival against whatever life forms already exist on Mars. While the probability of survival of any given Earth organism is extremely small, there may be some which find the new environment completely free of those natural enemies which on Earth held them in check.

It is the writer's own opinion (C.W. Benfield) that the unmanned voyager landings may be eternity's last look at an uncontaminated Mars. Subsequent manned landings may alter the entire future course of Martian history in some subtle way as a consequence of the tramp creatures carried along-- however sanitary the astronauts, and whether or not we colonize the red planet.

5.8.1 Selection and Development of Botanical Systems

Rather extensive R&D is required. Due to the unhurried (and unhurriable) nature of biological experiments, applied research should begin soon and be guaranteed continued funding.

5.8.2 Life Support Systems Integration

The biological participants in human life support systems will require serious design thought regarding support of their own existence. The interactions of a closed ecology system are likely to present instability problems for which damping mechanisms must be devised. The problem

is not likely to yield to simple mathematical model analysis. Full-dress simulation experiments will be necessary before the subject can become an engineering discipline.

5.8.3 Trace Contaminants

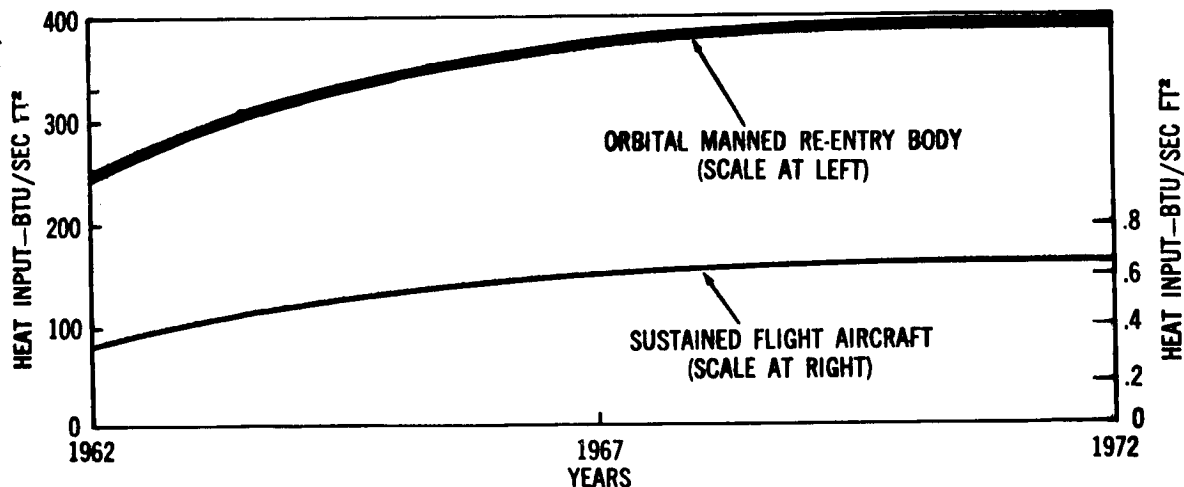
One of the instabilities referred to in paragraph 5.8.2 can be the result of uncontrolled accumulation of trace contaminants. Long term oscillations in ecological balance and even drifts toward total extinction have occurred on the Earth as a result of trace contaminants introduced by man. (Detergents in the streams, industrial wastes, fertilizers in solutions, etc.)

Control of trace contaminants may become a critical problem in short cycle closed ecology systems.

5.9 AUTOMATIC TPH (TEMPERATURE, PRESSURE, HUMIDITY)

(Quoting Reference 1)

"As aircraft speeds increase, the heat that must be dissipated to permit effective functioning of personnel and equipment also increases. Space vehicles will require cooling systems capable not only of maintaining satisfactory environments during extended space flights, but also during the tremendously high heating of atmospheric reentry. Estimates of heat inputs are shown in Figure 5-6.



Source: Ref. 1

FIGURE 5-6. HEAT INPUT DURING FLIGHT

On manned space vehicles, cooling is only one of the many problems in providing an inhabitable enclosed atmosphere for extended periods. Oxygen must be provided and excess moisture, carbon dioxide and other contaminants must be removed.

In the recent past, the growth of solid state electronics started a trend to a greatly decreased heat load from electronic equipment and, even more significantly, has revealed the need for new concepts in spot cooling in the next decade.

The recent increase in extremely low temperature requirements (cryogenics) is a major trend and a major requirement for further development (cryogenic storage of propellants for long periods represents a serious technical problem area-- Ed.).

During the past decade cooling system requirements in both heat load and temperature range have expanded tremendously. This expansion is expected to continue through the next decade. The primary change foreseen is that cooling systems will operate in small, well defined areas as contrasted to cooling for overall environmental control." (End of quote from Ref. 1)

5.9.1 Fast Action Pressurization

In pure oxygen atmospheres, penetration of the hull by a meteorite could cause a flash explosion followed by a severe implosion, due to combination of all the oxygen with hot metals. If men and equipment are to survive, repressurization must be immediate through automatic and rapid release of stored, compressed gas. This also points up one of the reasons for considering a two-gas atmosphere (pressure could not go to zero through simple combustion of magnesium).

5.9.2 Fire Extinguishers, Non-Toxic

The usual aircraft type of CO₂ fire extinguishers could result in serious overloading of the CO₂ removal system and conceivably could be fatal to occupants. Other common types of fire extinguishers use some highly toxic materials, such as carbon tetrachloride and sulphuric acid.

5.9.3 Reliability

Failure of the TPH controls can be disastrous. The highest reliability achieved will still leave a margin of dangerous uncertainty.

5.9.4 Waste Heat Rejection (At Low Temperature)

Rejection of heat by radiation to space from a low temperature source requires extensive radiator area. This problem is common to all thermodynamic machinery used in space vehicles. An alternative is to evaporate water or cryogenic propellants into the vacuum of space in order to cool a heat exchanger. Either method involves a payload penalty.

5.10 MEDICAL/PSYCHOLOGICAL

Quoting from Ref 1.

"Bioinstrumentation. Reliable telemetered physiological data from early space flights are essential to improve the design of ecological systems. Careful monitoring of physiological data during early flights is also necessary to increase the probability of the safe return of the crew.

For extended space flight, instrumentation that permits the astronaut to know at all times what is happening to his environment and physiological condition will be very important. These data must be telemetered to the scientists on Earth and supplemented by the astronaut's observations communicated by means of a reliable voice link.

The following parameters are tabulated as a guide to data which may be measured during space flight:

Environmental Factors

a) Dry Bulb Temperature	g) Radiation
*b) Relative Humidity	h) Acoustic Noise
*c) Total Pressure	i) Vibration
*d) Oxygen Partial Pressure	j) Toxic Gases
e) Carbon Dioxide Partial Pressure	k) Noxious Gases
*f) Acceleration - Profile	l) Isolation

* Data collected during the first two 1962 United States Astronaut Flights.

Behavioral Functions

- a) Performance (of tasks)

Physiological Measurements

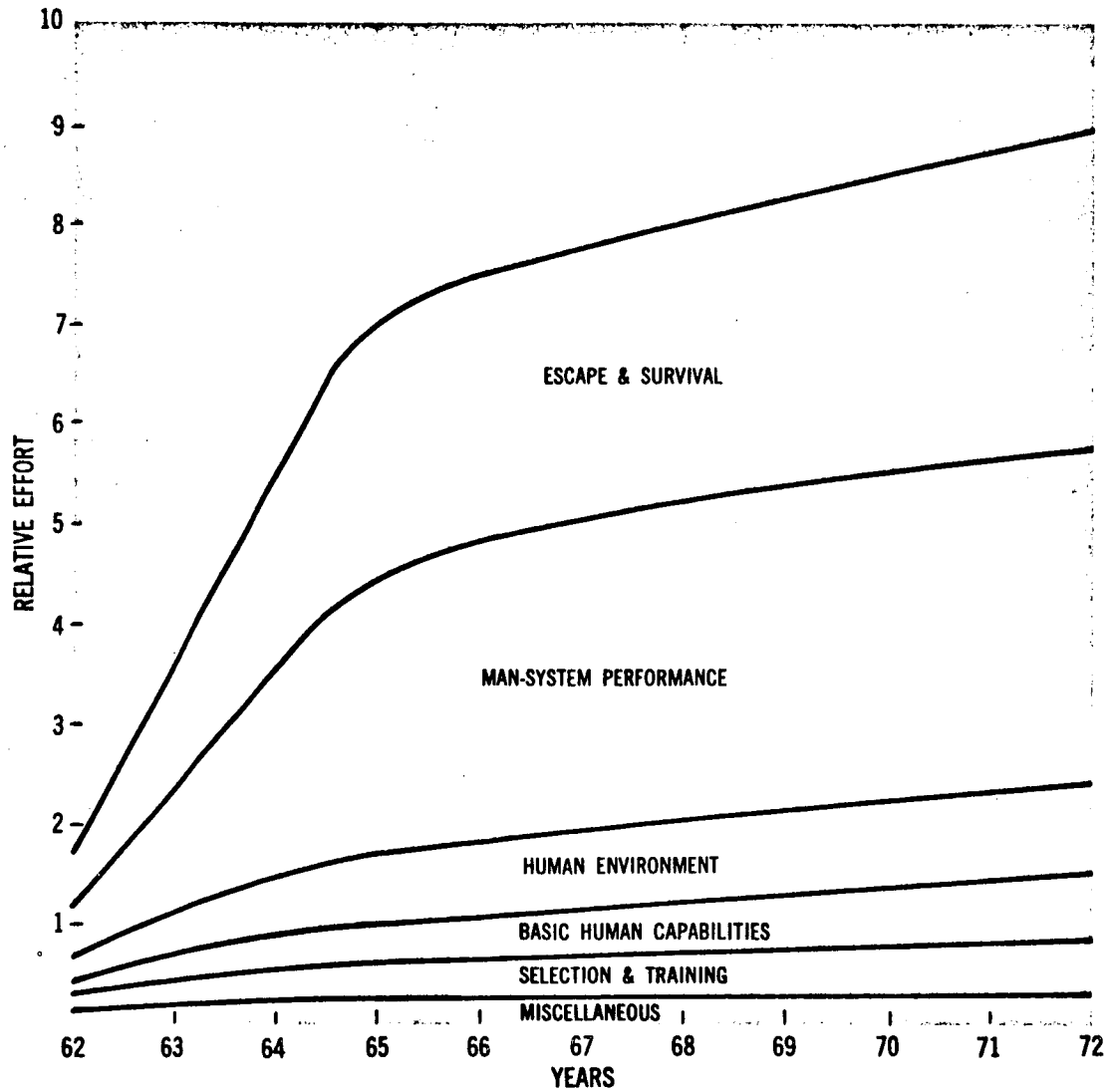
a) Skin Temperature - Several Locations	i) Blood Flow
*b) Internal Body Temperature	j) Skin Color - Circulation
*c) Respiration Rate and Depth	k) H ₂ O, O ₂ , and CO ₂ in Blood
d) Respiration Waveform	l) Measurements of Hormones and Waste Products
*e) Electrocardiogram	m) Saliva Analysis
*f) Heart Rate	n) Urinalysis
*g) Systolic Blood Pressure	o) EEG
*h) Diastolic Blood Pressure	p) Blood pH

It is not intended that all the above variables should be recorded at one time or that all are required on any specific flight. The availability of equipment and type of program will determine the degree of instrumentation.

The number of parameters which can be measured and telemetered depends upon the adaptation, miniaturization and integration of existing equipment and the required development of new instrumentation and techniques.

Human System. The anticipated manpower requirement in various human factor areas during the period 1962 to 1972 are reflected in Figure 5-7. Primary effort for the next ten years will be concerned with the performance of operators as components within systems loops and with the problems of escape and survival. The effort devoted to human problems associated with prolonged space missions will be almost as large. The ability to provide suitable human environment for long periods is the probable ultimate limiting factor of space exploration. A relatively modest effort will be expended on basic research in human capabilities and limitations, to determine these capabilities within the context of specific systems. Other critical research problems are closed ecological systems and complex aspects of human behavior, for example, information processing, decision making, detection and recognition. Among the miscellaneous problem areas, emphasis will be placed on human reliability and accident prevention.
(End quote from Ref. 1)

* Data collected during the first two 1962 United States Astronaut Flights.



Source: Ref. 1

FIGURE 5 - 7. HUMAN SYSTEM RESEARCH AND DEVELOPMENT
(ANALYTICAL AND LABORATORY ACTIVITY)

5.10.1 Morale

Americans have characteristically devoted entirely too much attention to the trivial morale factors among military personnel, while almost totally ignoring the essential morale factors. It is not likely that this will change during the early phases of planetary colonization, since it is a culturally ingrained matter and not subject to rapid change however great the need. Perhaps adequate compromises can be reached if the obvious needs can be proven to be important by "sociological research" or psychiatric case histories.

However exciting space flight may be during the first few days on orbit, the privation, confinement, discomfort and boredom of a long interplanetary flight could lead to generation of intolerable hostilities, particularly among young men.

5.10.2 Sanitation

Water and space are precious aboard the contemplated interplanetary vehicles. The Gemini astronauts described their condition after several days in orbit as "pretty ripe". Bathing will be difficult, even in a "shirt sleeve" environment. It may prove to be nearly impossible at zero g. However, the thought of several months without a bath would frighten all but the hardiest of pioneers.

Dirty skin makes a fine culture for fungi which can cause serious disability and soreness.

Shaving and toothbrushing may become rather trying tasks at zero g.

Perhaps sanitation presents another set of reason for spinning a part of the vehicle.

5.10.3 Diagnosis and Treatment Health

It is probable that biotelemetry and voice communication will be maintained throughout manned flights. Diagnosis of ailments will be done by remote control.

Selection of medications for a long flight represents a serious problem. Almost any of several thousand drugs might be needed while the probability of any one being essential is quite small.

Antibiotics, fulvicin, sedatives, antiseptics and needle anesthetics probably will be needed, but beyond that the choice widens and becomes subject to much speculation and debate.

5.10.4 Surgical Implant of Biomedical Sensors

Continuous biotelemetry from active men and animals can hardly depend upon their wearing a harness of wires and transducers for long periods of time.

Considerable progress has been made in the application of microelectronics to biomedical telemetry implants capable of transmitting to a recorder several hundred feet away. Upgrading of this technology perhaps could improve reliability of biomedical data-taking in space.

5.10.5 Coriolis Effect

In a spinning space station, when one climbs a ladder (toward the hub), he will have a feeling that the ladder is curving under him or that the direction of "down" is shifting rapidly. A similar sensation can be experienced by walking in toward the hub of a moving carousel. Coriolis force (due to conservation of momentum) causes accelerations which seem contrary to intuition. In certain individuals, this can lead to severe intestinal disturbances and nausea.

The effect perhaps may be overcome by training in a rotating simulator. Some sailors never overcome seasickness; some astronauts perhaps may be plagued with this trouble during the entire flight.

5.10.6 Reentry g Physiology

Spending a long period at zero g is somewhat like going to bed for a month. The normal stresses being removed, the body adapts to the easier life and gradually discards its unneeded strength. Sudden application of reentry g after several months at zero g may prove to be more than flesh and bones can bear. (Recall how heavy and weary you felt when you crawled out of the water after a half-hour in the swimming pool. Suppose you'd spent a month in the pool.)

Apparently there is actually a progressive de-calcification of the skeleton at zero g. The rate of progress of this must be determined. The question of whether nearly total absorption of the skeleton ever occurs and how long it would take must be answered.

5.10.7 Surface g Physiology

When man first lands on the Moon, he likely will be exhilarated by his lightness and apparent super-strength. After several weeks at this reduced gravity, he may find the superman illusion disappearing due to an effect similar to that mentioned in paragraph 5.10.6.

It is not known whether permanent physiological deterioration would result from the reduced gravity on the Moon or Mars.

After such a period of reduced gravity, the astronaut must once more subject himself to the high accelerations of orbital injection. How long he can stay before he becomes too weak to come home is a question that must be answered.

5.10.8 Zero g Physiology

At zero g, the strong man's heart may continue trying to pump his blood against a six-foot head of pressure. Or it may progressively slow down and become lazy, since it no longer has to work so hard to stay alive.

There may be many long-term effects of zero g exposure, which previous flights have been too brief to disclose. Biosatellite experiments may lead to considerable insight. Space station flights are needed to evaluate any such effects before committing expeditions to deep space.

Some effects of extended zero g may not show up until years after the exposure.

Then again-- there may be no problem.

5.10.9 Deterioration of Life Forms at Zero g (Zero g Physiology)

Some plants and animals are known to require gravity for normal growth and development. Some appear to be immune or indifferent to the g vector as far as morphology is concerned.

What sensors do plants use to tell which way is down? How would they grow at zero g?

Frog eggs held upside down will hatch deformed tadpoles. Will they hatch at all at zero g?

Is gravity essential to reproduction?

Will genetic mutations result from zero g?

Is an animal's size gravity dependent?

Perhaps biological studies at zero g and reduced gravity will shed new light on fundamental life processes.

5.10.10 Reduction in Natural Immunities

The effects of the space environments upon natural immunities to disease are unknown. There may be no effects at all, or there may be a serious modification of the natural healing and antibody production processes.

5.10.11 Healing in Space Environment

Low pressure, pure oxygen atmospheres appear to promote rapid healing of wounds. A better understanding of the healing process may result from research into this supposed phenomenon.

REFERENCE LIST FOR SECTION 5

- 5.1. Aerospace Technical Forecasts 1960-1970 and 1962-1972, Aerospace Industries Association, 1725 DeSales St., N. W., Washington 6, D. C.
- *5.2. Kovit, Bernard (Assoc. Ed.), "The Space Suit," Space/Aeronautics, June 1965, page 43.
- *5.3 Gaume, James G., "Life Support Systems for the Lunar Base," ARS Paper 1227-60.

*Though not specifically cited, these references supplied some background material.

Section 6

TASK INSTRUMENTATION

6.0 GENERAL COMMENT

This section lists the instruments used to make typical scientific measurements involved in space exploration.

Task instrumentation includes only the payload instrumentation -- the scientific handling and measuring equipment, and/or the instrumentation necessary to measure the status, performance and responses of equipment connected with a scientific or technology experiment.

Specific instrument sets were not firmly selected for the concepts considered. More extensive future study is expected to enlarge and detail this part of the PATTERN tree.

6.1 CLASSES OF SCIENTIFIC MEASUREMENTS

Table 6-1 shows the type of measurements made by various payload types for the different space exploration mission modes.

6.1.1 List of Instruments. Abstracted from Honeywell R-ED 25280 dated 27 October 1965. (Numbers are arbitrary designators based on page numbers.)

<u>Instrument</u>	<u>No.</u>
Active Seismic Explorer (Seismic Refractometer)	91
Aerosol Profile Meter	11
Airglow Analyzer, Spectrometer	44
Alpha-Scattering Soil Analyzer	108
Analytical Chemical Water Detector	16
Anemometer	19
Backscattering Cloud Detector	31
Beta Densimeter	8
Bistatic Radar (Radio Prop.) (Occultation)	107
Borehole Drill	6
Bottomside Sounder	10
Capacitor Detector	75
CO ₂ - CO Infrared Spectrometer	28

TABLE 6-1. CLASSES OF SCIENTIFIC MEASUREMENTS

Payload Type	Measurement Type	Manned/Unmanned	Mission Modes
1. Interplanetary	Particles & fields	M&U	Flyby, orbiters, interplanetary, intergalactic
2. Planet Oriented	UV, IR, etc., topside sounder	M&U	Flyby, orbiters
3. TV & Photo Mapping	----	U	Flyby, orbiters
4. Photo Mapping	----	M	Flyby, orbiters
5. Planet Entry	Ionospheric & atmospheric vertical structure	M&U	Landers, impactors, lander-rover
6. Surface atmosphere and sky oriented	Winds, local temp., pressure, composition, bottomside sounding, sky brightness, heat, flux, UV, IR up-looking	M&U	Landers, lander-rovers
7. Surface Chemistry and Physics	Seismometry, soil analysis cores	M&U	Landers, lander-rovers, rovers (1 only or L or R)
8. Surface Sampling	Manual sampling	M	Carried by man on any surface vehicle (one per mission)
9. ABL-Life Detection	Automatic biological lab	M&U	Landers, lander-rover, rover (no duplication-- same one on lander as on rover)
10. TV Surface & Micro	----	U	Landers, lander-rover, rover (1 only on L or R)
11. Photo Surface & Micro	----	M	Landers, lander-rover, rover (1 only on L or R)
12. Radar Mapping	----	M&U	Orbiter, flyby
13. Bistatic Radar	Electron density, etc.	M&U	Planetary orbiter, flyby, interplanetary, intergalactic, comets, asteroids
14. Solar	X-rays, gamma, telescopes, spectrographs	M&U	Solar close approach missions

NOTE: Concepts may require addition or subtraction of one or several instruments in the package. Ranges suggested for selection of pressure, for instance, by the concept.

<u>Instrument</u>	<u>No.</u>
Charged Particle Telescopic Spectrometer (Electron S.)	56
Charged Particle Telescopic Spectrometer (Proton S.)	67
Cloud Analyzer	18
Copper Iodide Ultraviolet Photomultiplier	66
Curved Surface Plasma Analyzer	54
Differential Thermal Analyzer (Phase-Transition Detector)	86
Dipole Moment Concentration (Humistor)	40
Dual Light Level Indicator	98
Electrical Conductivity Meter	90
Extended Microwave Spectrometer	23
Faraday Cup	53
Fluorescent Spectrometer	83
Fluxgate Magnetometer	82
Frost-Point Hygrometer	47
Gamma-Ray Densitometer	9
Gamma-Ray Spectrometer	89
Gas Chromatograph	1
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6.2 INSTRUMENT PACKAGE

For lack of a detailed schedule of exact experiments, the instrument configurations have been lumped under the keyword "instrument package" for most of the concepts. This is to be broken down into details when adequate time and information are available for the work. Under this blanket heading, several generalized technological deficiencies have been identified.

6.2.1 Sterilization of Instruments

Many of the sensors are susceptible to temperature damage; therefore, sterilization by an extended warm soak is preferred over high temperature sterilization. Chemical sterilization is effective against some biotics, but not all. The requirement for sterilization imposes rather serious design constraints and must be considered a part of the spaceborne packaging problem.

6.2.1 Spaceborne Packaging

Many of the instruments contemplated exist as manual, semiautomatic or even totally automatic devices for laboratory use. However, producing the same measurement accuracy (and interfacing results with telemetry) in

the space environment, by totally automatic operation, requires orders of magnitude greater design effort. Not only must size, weight, and power consumption be minimized, but the instrument must be ruggedized against damage and calibration shifts during extended periods in an alien and hostile environment.

6.2.2 Measurement Techniques

Some of the required measurements will make it necessary to develop new techniques of measurement for which instruments and procedures do not now exist. A specific list of these deficiencies, unfortunately, is beyond the scope of the present study, but the voter will recognize some of the difficulties from a consideration of the concept's operational limitations. For example, determination and mapping of sea water salinity from orbit at first glance appears impossible. However, new microwave techniques show considerable promise of providing such a capability.

6.2.3 Calibration Standards

In order to assure that false data is not being received, it is desirable that each instrument be capable of periodically checking itself against a known standard onboard the spacecraft. Provision of adequate check point standards may in some cases be as much of a problem as that of designing the measuring device. In a few cases no practical standards will be possible without major breakthroughs.

6.2.4 Accuracy, Instruments

This generalized comment can be applied to almost any measurement. Whatever accuracy is easily possible, someone will need one or two more significant figures. The weight of this TD is to be voted after discussion of each problem by the group. The case is rare wherein higher precision and accuracy are available than that "required" for the experiment.

6.2.5 Reliability, Instrument Package

If funds sufficient to place the instruments at the desired end point are to be expended, then it is axiomatic that instrument failure implies a near zero return on investment. The unfortunate history of the early Ranger shots illustrates this. As mentioned in paragraph 6.2.1, part of the spaceborne packaging problem is that of providing for instrument survival in a hostile environment. In many cases there is inadequate time for advance life testing, and often the cost of hardware for reliability testing is prohibitive. Reliance upon parts life test data and reliability theory

represents a calculated risk. In extreme cases, multiple redundancy of function is essential when mission success probability must be high.

6.2.6 Thermal and Radiation Protection

This generalized requirement is applicable to all spaceborne instruments, but in some cases (near solar probes and Van Allen belt satellites) it may become a severe limitation upon design in that adequate cooling and shielding may exact a heavy price in propellant.

6.3 METEOROLOGICAL ROCKETS

Staged grenades are discharged as the rocket ascends. Batteries of directional microphones at monitor stations determine range and direction of each detonation through use of sonic interferometry. Telemetry of the detonation from the rocket permits apparent range measurements to be studied as a function of velocity of propagation.

6.3.1 Cost of Grenades

The grenades and launchers actuated by a programmer timer (or by radio command) must be simple springloaded fireworks type devices, to keep costs low.

6.3.2 Size and Weight of Grenades

Cardboard cases should permit keeping grenade weight below six ounces each, but the total number of required grenades imposes a weight limitation.

6.3.3 Meteorological Instruments, Weight and Size

Several may be selected from the list in subsection 6.2, but their design must be based upon minimum cost, size and weight. Commercial toy standards should apply when acceptable accuracy can be achieved.

6.3.4 Response Time of Meteorological Sensors

The rocket ascent will impose rapid changes in the measured environment. A better profile can be obtained through subsequent parachute descent. Even here, low inertia, fast response instruments are needed. The descending parachute may be RDF guided using a parachute with servo capstans.

6.4 48-INCH TELESCOPE WITH IMAGE INTENSIFIER

This balloon-borne telescope is to be attitude stabilized to ± 0.01 arc second. To permit seeing more in shorter times of exposure, an image intensifier tube is to be used at the output optics.

6.4.1 Research and Development Required Image Intensifier

Image intensifiers are in the early application stage, and these leave much to be desired in terms of resolution. It would profit us little to get brighter images with less detail. Sacrifice of color also represents a serious loss of information, particularly for planetary studies. A 5,000 line (or better) full color intensifier system is needed.

6.4.2 Research and Development Required Star Tracker

The stated stability requirement is straining the state of the art of image dissector star trackers. However, by using the 48-inch main optics to track a star, such stability is possible. Unfortunately, when planets are being tracked there may be no suitable reference star in the main optics field. Research and development is needed to optimize star tracker design.

6.5 SOLAR RADIO SATELLITE ANTENNAS

Deployment of four 700-foot booms will require specially designed curl tubes of rather large diameter to provide adequate rigidity. The de Havilland ruled tape scheme may be able to provide adequate, lightweight booms. Several other deployable boom concepts have been demonstrated.

6.6 MANNED TELESCOPE (Reflective Telescope)

An eight or ten foot mirror telescope made of fused quartz would present a severe weight problem. If made of sheet metal, it would present an optical precision and thermal distortion problem. Transportation of the telescope may involve some shock and vibration degradation.

Fabrication of a lightweight mirror with astronomical quality optics presents a challenge to optical engineers. Degradation of optical materials in the space environment requires further experimental evaluation.

6.7 140 MM GIMBALED REFRACTIVE (OR REFLECTIVE) TELESCOPE

The large manned space station will need a telescope for viewing the earth below. This will be gimbaled to the space stabilized hub of a rotating structure. Gimbaling presents a mechanical design problem due to the rotational noise which may be generated by the hub section drive against the rotating structure. Resolution and magnification will be limited by vibrations from the space vehicle.

6.7.1 Pointing Accuracy

With adequate magnification, pointing accuracy becomes a problem because of vehicle rotation and vibration. To get 10-foot resolution at the ground, the field of view must be limited to about a 100,000 foot square. Probably a range of magnifications will be provided with a turret eyepiece.

6.7.2 Vibration Suppression

How to best achieve vibration suppression will require solution. Since the hub section is at zero-g, neutral suspension in limit rings may be substituted for gimbaling to avoid coupling in vibrations.

6.8 HIGH RESOLUTION TV, IR AND UV IMAGING

The disappointing lack of fine detail in the Ranger and Mariner TV pictures lends support to the writer's contention that one good still picture is worth a whole hour of low resolution movies when viewing strange scenery.

Certainly, we are at the threshold of a capability for television imaging with sensitivity, resolution and color discrimination superior to that of the human eye.

A step in the right direction would require a development aimed at providing 5,000 line, one-frame-per-second high fidelity color television, with good response to low light levels. A fast action shutter on the camera would open at the beginning of each frame permitting exposure for about 1/100 sec. The captured optical image would be "stored" on the screens of five separate color-analyzer camera tubes. These color tubes would be scanned out with 5,000 line resolution and with common synchronizing signals. Each color would be sent over a separate telemetry channel. The contemplated five colors are: near infrared, red, green, blue and ultraviolet.

As available bandwidth increases in near earth laser and microwave communication links, the hi-fi TV would be speeded up until 30 frames per second were transmitted.

Transmission of colors simultaneously (rather than sequentially with color burst coding) permits studying the identical scene separately in various parts of the visible and fringe invisible spectrum.

The need for such high resolution color TV would become especially noticeable in remote examination by TV microscope of soil specimens on a planetary surface. Introduction of polarizers ahead of the specimens and polaroid analyzers ahead of the color TV could permit gaining additional information about the materials being examined.

The eventual TV system circa 2000 AD should be detail limited chiefly by the quality of the optics ahead of the cameras.

IR mapping of the earth, moon or a planet from orbit should have the highest image resolution achievable with the state-of-the-art. We should try for at least 2,000 lines per frame in the far IR. Preferably, this would be done without cryogenic cooling. Conceivably it may be necessary to use a mosaic of detectors, rather than scanning a single detector to achieve the needed speed of response on a frame basis.

Research and development in IR detectors and lenses is required.

Far ultraviolet image systems also will require sensor development.

6.9 RESOURCES SURVEY SATELLITE

In attempting to assess the statistics required for a resources survey from orbit, all of the visual information possible can be used. Ideally, a high resolution, high fidelity color TV system would provide a spectral analysis of specific frame areas by superposition and analysis of the scans of those areas from the stored telemetry data. Conceivably, it would be possible to determine (for example) what kind of trees predominated in a particular forest area by merely comparing the spectral analysis of that area with those of known sample areas.

The color range of the imaging system should extend as far on both sides of the visible spectrum as available sensors can take it.

Direct spectrographic analysis of selected regions should be available by color TV transmission of the spectrograph output coded with time, azimuth, elevation, and orbital position data. In addition, digital analysis of the multicolor TV image of the area on a line by line basis could permit rough assessment of spectral density distribution. Some R&D is required to permit design of adequate spectrometers.

6.10 METEOROLOGICAL SATELLITES

Microwave sources upon the earth demark specific, fixed, and knowable earth positions. An adequate interferometer system could provide the directions of these sources from the satellite and identify their frequency, type of modulation and apparent brilliance; by correlating this information in real time with the weather images, we could identify positions of storm features precisely in relation to these known landmarks (even when the surface features were totally invisible).

At the data reduction displays, these sources would be plotted as points directly upon the photographs of storm cover and upon the IR maps.

Research and development will be required to provide such microwave sensors.

6.11 RELATIVITY SATELLITE

Adequate atomic clocks with digital electronics exist now in relay rack hardware. Reduction of these circuits to a spaceborne, unattended microelectronics system without sacrifice of performance will require extensive engineering work, and conceivably research and development.

6.12 JUPITER FLY-BY/SATURN FLY-BY (Long Range Radar Altimeter)

It is desirable to obtain as much information as possible about the Jovian (or Saturnian) atmosphere. Conceivably a high power, constant-amplitude versus swept-frequency, long-pulse, radar altimeter could yield microwave absorption spectrum data from which the chemical nature and conceivably the temperature and depth of atmosphere might be determined.

Present radars have neither the power, the range, nor the quality required.

Furthermore, development of a method for wringing the maximum data from the returns will require considerable study and experimentation to permit choice of optimum frequencies and correlation with known atmospheric characteristics.

Such a radar must also have an MTBF of many years.

6.13 LONG RANGE RADAR ALTIMETER

During planetary approach it is desirable to have range and range rate from the planet. Preferably this data would begin to be available a million or more kilometers out; however, 100,000 kilometers would permit reasonable adjustment to be made to the approach trajectory.

The same radar would provide accurate altitude versus time readings near periapsis, to permit adding meaning to other experimental data.

By emitting frequency sweep, long pulses during which radio frequency output power is constant, then recording the amplitude versus frequency of the echo, one might gain valuable data regarding the chemical composition of the atmosphere from its microwave absorption spectrum.

REFERENCE LIST FOR SECTION 6

No specific references were cited for this section. However, descriptive material from Surveyor and Voyager studies was available, and will be used in future detailing of the tree.

Section 7

CHECKOUT AND MAINTENANCE

7.0 GENERAL COMMENT

Checkout and maintenance includes all equipment functions needed for handling, checking, maintaining, and repairing operational systems and subsystems, both space systems and ground systems.

A breakdown and discussion by configuration and TD key words has been postponed for the sake of brevity.

The following is quoted from the Aerospace Industries Association's "Aerospace Technical Forecast 1962 - 1972". (Ref 7.1)

7.1 CHECKOUT MAINTENANCE AND TESTING

Before the end of the decade there will be trends to completely check out each stage at the factory before shipping. Rapid and simple booster stage tiedown mechanisms and sustainer connections will allow vehicle assembly in a matter of hours rather than days. Electronic checkout and self-verification equipment used just before launch will be an integral part of each system.

Launch cycle time at each pad will be gradually but materially reduced as a result of the design and operational techniques evolved during the next ten years. Cost to launch per pound of payload will in turn be gradually reduced. (The ultimate goal is to require little more preflight checkout than that now given for scheduled aircraft flights-- Ed.)

7.2 REMOTE CHECKOUT

Increasing booster size, manned payloads, and rapid growth in mission duration will greatly alter checkout requirements during the next ten years. Larger boosters and the use of higher energy fuels will increase hazards in the vicinity of the launching pad and require remote checkout and control. This and the need for higher accuracies and increased checkout speed will continue to favor the trend to computer controlled digital measurements and control signals (Figure 7-1). This shift from analog to digital techniques is already apparent, and will result in the development of many new digital

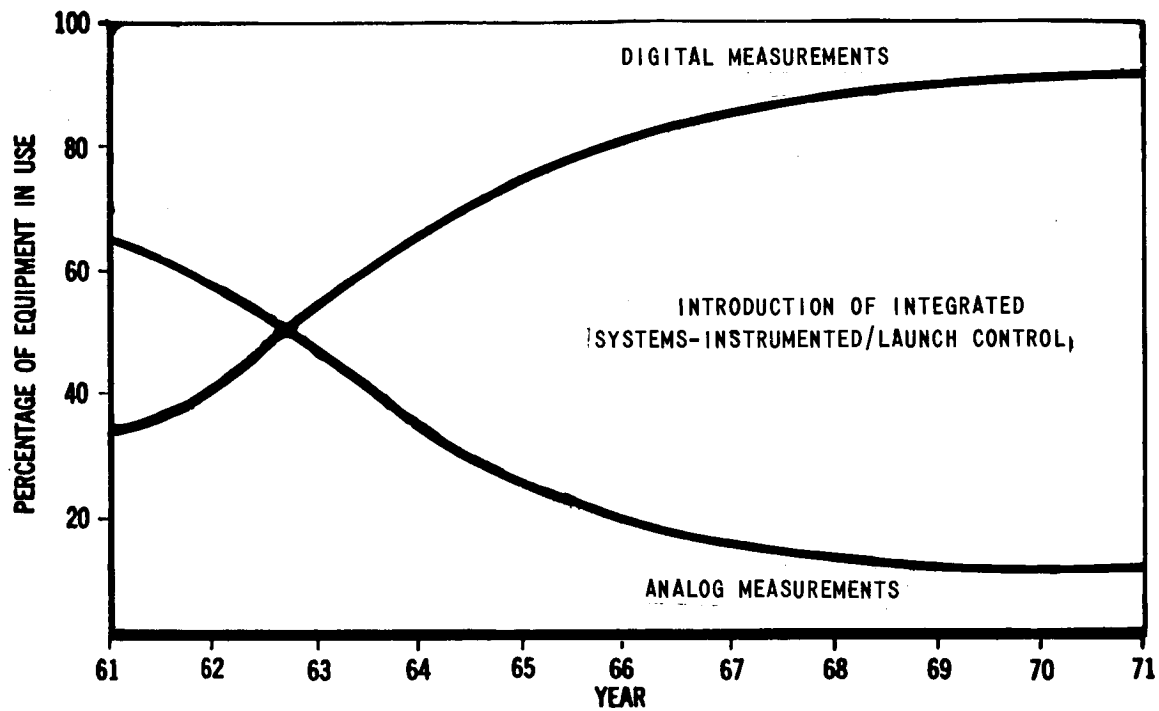


FIGURE 7-1. TREND TOWARD DIGITAL VERSUS ANALOG CHECKOUT EQUIPMENT

SOURCE: Ref 7-1

sensors (pickups), minimizing the need for analog to digital converters. The recently developed encapsulated digital logic element (integrated circuit), or deposited thin film circuits with chip active elements, will rapidly replace conventional semiconductor components in basic building blocks of the ground electronic system. The use of these elements will simplify the equipment, improve its reliability, and increase the circuit density.

7.3 TELEMETRY

A similar trend to digital equipment is taking place in the field of instrumentation and telemetry, paving the way for integration of the two systems. Communications links between the ground checkout equipment and the flight vehicle will be simplified, first by the use of coaxial lines carrying digital pulse trains; later by checkout and control through the RF links comprising the integrated PCM telemetry/launch control and checkout system. Small rugged stimulus generators, calibration standards, and measurement sensors will be developed for airborne use, converting measurement and control signals to serial pulse train digital form within the flight vehicle.

7.4 RELIABILITY AND MAINTAINABILITY

Systems will be designed for high reliability and relatively long operating life. An additional increase in reliability will be attained by decreasing the operating time of the equipment during test and checkout. This decrease in time will be accomplished by the use of automatic test equipment and by parallel test techniques so that time-to-test and checkout will not be a function of the sum-of-the-times for individual tests. New approaches to test and checkout will be developed whereby the missiles are tested in a dynamic rather than static mode, and data taken during the test will be examined afterward for verification of proper operation or isolation to a removable module.

Maintenance considerations will continue the trend toward modular construction techniques. Module size and weight will drop by discrete increments as checkout speed increases, following the trend in airborne electronic equipment. Throw-away modules will include many more functions than they do now. Therefore, the level of checking to isolate a faulty module will not be as fine.

Payload flight systems will be designed for a high reliability and a relatively long operating life and will have to be tested at periodic intervals in flight or in orbit. In-flight checkout systems will be developed to accomplish these periodic tests, and these same systems will be used for preflight checkout of the payload stage. The ground checking of the payload stage will then consist of simple monitoring of the go, no-go condition before launch.

As in the past, considerable specialized factory and launch site maintenance checkout equipment will be required to insure proper operation of the black box level of equipment. Since the airborne vehicle will also be modularized to a large extent, this equipment will be required to isolate faults to the module level. As in the case of booster preflight checkout equipment, computer controlled digital techniques will be used. Near the end of the decade, self-learning computers will be given increased attention. However, one of the major problems facing auto-checkout is determining what to check. Because of the rapid pace of technology, and the dependence of self-learning computers on past history, it is unlikely that computers of this type will replace man in this role in the foreseeable future.

7.5 MANNED AERODYNAMIC SYSTEMS

Research and development is taking place on all items involved in the manufacture and support of manned aircraft systems. They include electric and electronic devices, mechanical handling and transportation equipment, hydraulic and pneumatic servicing equipment, and packaging and development of aids necessary for the training of maintenance personnel and flight crews. The equipment may range from single handling slings to complex automatic checkout equipment, or zero launching equipment. The development of experimental ramjet and rocket aircraft and its support equipment has become a reality. Further development of such aircraft will call for new developments in ground support equipment. To accomplish the objectives that are now being set forth for manned aircraft, ranging all the way from combat to outerspace planes, greater and greater demands are being made on support equipment. In addition to complexity demands, requirements will exist for improved mobility, reduction in size, reduction in cost, and even some elimination of ground support equipment will take place. Testing concepts will change. More and more testing will be shifted to the airborne systems in the form of built-in tests. Highly mobile equipment of small size will be utilized in the actual operating areas with the more extensive, and complete testing confined to factory areas where such testing and checking can be done more effectively and efficiently.

7.6 AUTOMATIC CHECKOUT

Although the ultimate has not yet been attained in automatic testing, actual operational use of automatic testing will pave the way for development of more practical, yet more sophisticated, automatic checkout equipment to attain speed, reduction of test points in the air vehicle system, and to increase the reliability and repeatability during testing.

Increased use of the introspective method of testing appears attractive. The blending of this method with ground checkout complexes will require its early integration into the airborne system design. Such schemes will be necessary to reduce ground systems, thus reducing costs, and checkout time, and assuring minimum air vehicle turn around time with a much higher confidence factor.

Expanded use of general purpose computers for all testing purposes is expected.

Other approaches will feature in-flight recorders whose data could be analyzed by ground computers and immediate decisions made as to the status of airborne subsystems. An additional function of fault isolation of replaceable air vehicle units would also be satisfied by such a system.

7.7 NON-ELECTRONIC SYSTEM TEST AND CHECKOUT

Analytical testing methods will be used in systems other than pure electrical and electronic. Such systems would include primary-secondary flight controls, hydraulic, pneumatic, power plant, air induction control, etc. This will become necessary as the manned air vehicle technology advances. New and more rigid requirements will be placed in these areas to enhance and assure high reliability for advanced vehicles. (New interfacing transducers also may be required. --Ed.)

During the next decade further consideration will be given to hydraulic servicing units. Trends towards higher flows and pressures, elevated temperatures and higher filtration requirements will continue to rise from typical present day levels of 250 gallons per minute, 4000 to 5000 pounds per square inch, 550 to 600 degrees Fahrenheit, and 2 to 5 microns, respectively. Ground cooling units for temperature pull down after supersonic flight will be investigated.

Nuclear powered systems will receive further study. Requirements will be created for remote control mechanical handling equipments, new servicing and test equipment protection methods, and facilities.

Support equipment costs may approach the costs of the air vehicle. This will be partially offset by improved development management of new air vehicles and their support equipment, and the utilization of practical knowledge attained during the last few years." (End of quote from Ref 7.1)

7.8 AUTOMATIC FAILURE MONITORING AND FAULT ISOLATION

7.8.1 Monitoring and Switching

This keyword denotes the general problem of providing for automated on-board checkout.

7.8.2 Fault Isolation

This refers to the problem of deciding rapidly which module should be replaced.

7.8.3 Space Weight and Interchangeability

If the entire electronic system could be designed by one group, it would be possible to so design the equipment that a given spare module could serve as a replacement in many different pieces of hardware. Such design would be less than optimum (weight and volume wise) but would permit considerable reduction of the total number of spare modules necessary to assure in-flight repairability.

7.8.4 Zero-g Work Tools

Problems (such as soldering, nut tightening, metal cutting, etc.) which are easy on earth may become formidable at zero g. Some special tools have been designed (for example, a zero reaction drill) but many others will be necessary if more than simple plug-in replacements are anticipated.

7.8.5 Technical Data Access

Very often nowadays the weight of the instruction manuals exceeds that of the equipment. With emphasis upon microcircuitry, this ratio may become ridiculous; yet rapid access to adequate technical data is necessary for proper troubleshooting and repair. Rapid access microfilm libraries for on-board display of maintenance (and other) data are needed. The smaller they are the more likely they are to be allowed onboard.

7.8.6 Reliability

What checks the automated checker? How much more reliable must it be than is the equipment it checks? These enigmatic questions will often arise and no simple answer is available.

7.8.7 Simplification

The enigma of paragraph 7.8.6 points up the need for using the simplest feasible designs for automated checkout.

7.8.8 Malfunction Detector

This is thought by many engineers to be a mythical device like the TURBOENCABULATOR. Actually, it refers to the real function of the automated checkout equipment. Whenever the sensed sequence of events contradicts the planned sequence or is contrary to logic, a malfunction is indicated. This permits a danger alarm. Fault isolation is the next logical step after a malfunction has been detected (unless abort is initiated by the alarm). Basically the malfunction detection is a computer automated checklist.

7.8.9 Failure Alarm

Initiated by a detected "malfunction".

REFERENCE LIST FOR SECTION 7

- 7.1 "Aerospace Technical Forecast 1962-1972," Aerospace Industries Association, 1725 DeSales Street N.W., Washington 6, D. C.

Section 8

DETECTION, DISCRIMINATION AND TRACKING

8.0 GENERAL COMMENTS

Detection, discrimination, and tracking includes sensing, signal conditioning, data processing, and display functions required to detect, discriminate, identify, classify, and track targets, and compute guidance corrections from fixed sites. Specifically excludes rendezvous and docking equipment.

The present tree questions in this area will have little influence upon overall results; therefore a detailed breakdown by configuration and TD's was postponed. For more exhaustive future PATTERN studies, this section may be expanded.

This section includes data on the subsystems used to sense, track, identify, and control space vehicles from the ground.

For the purpose of this study, it is assumed that the DSIF facility will be available for this task.

Problems will arise when more than one space vehicle must be tracked and controlled by DSIF at the same time; this is regarded as an operational deficiency rather than a technical deficiency.

When a mission involves such extreme distances and/or data rates as to be beyond the expected DSIF capability then that will be considered a technological deficiency.

Tracking of vehicles during launch and injection has been accomplished smoothly and on a routine basis, therefore no severe technology problems are expected as a result of boost tracking requirements in the next 15 years.

(For general information, the following is quoted from Aerospace Industries Association's "Aerospace Technical Forecast 1962-1972". (Ref 8.1)

8.1 SURVEILLANCE AND TRACKING - SENSITIVITY

The increasing use of space systems and quasi-space systems operating at high velocities and long ranges is generating a demand for large improvements in the sensitivity of all types of surveillance and tracking systems. In addition to improvements as illustrated on the accompanying chart, (Fig 8-1),

exploitation of millimeter wave radar (active and passive) and coherent light systems should be aggressively pursued.

To achieve the radar improvement charted in Figure 8-1, it will be necessary to increase power, reduce detector noise, reduce side lobe noise, and improve signal processing techniques. Therefore this is the source of the largest possible gain. Considerable additional gain in sensitivity should result from more sophisticated data processing. Probably the most hopeful source of improvement in optical system sensitivity is image intensifiers but improvements in photographic film should not be neglected.

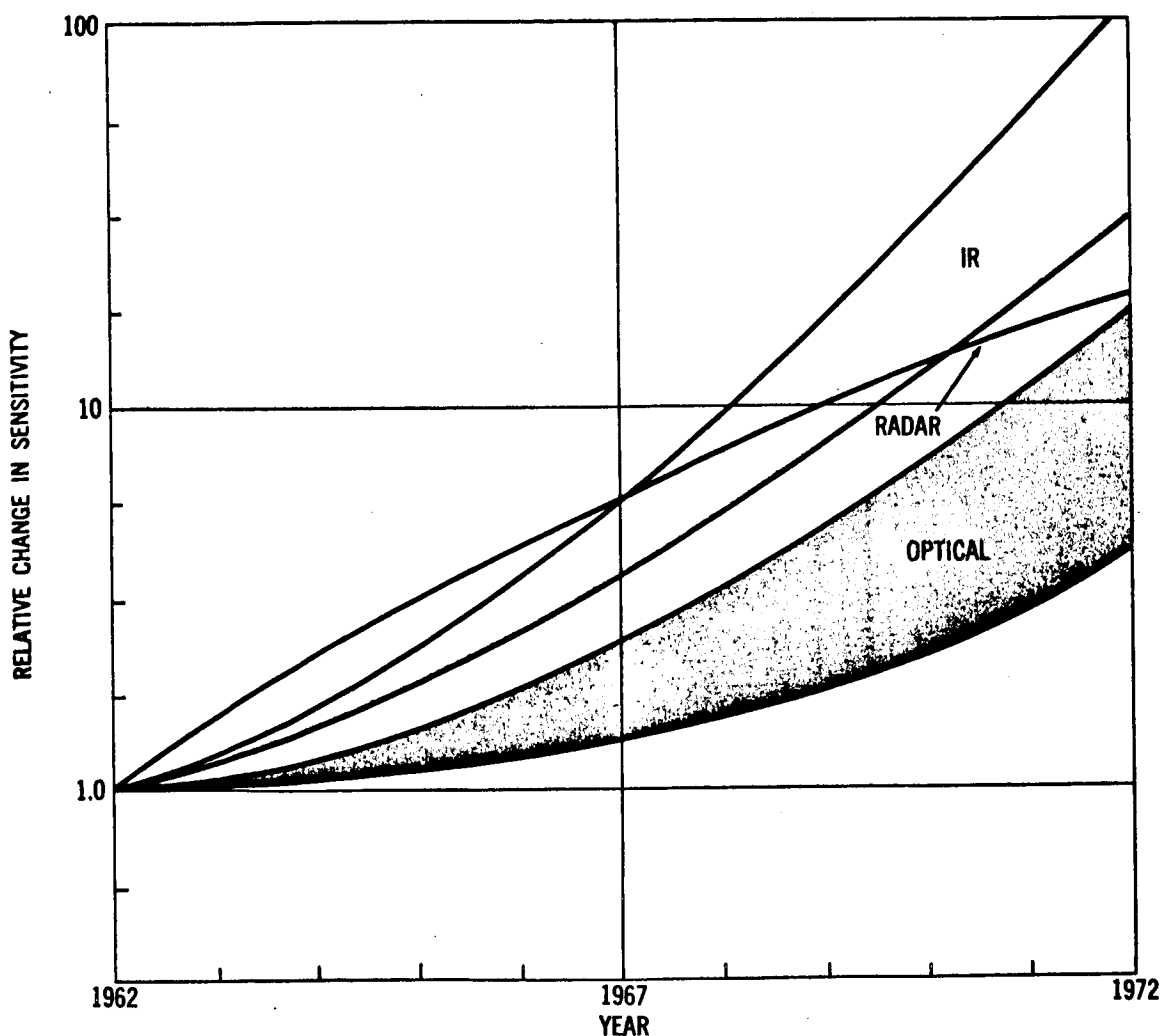


FIGURE 8-1. TRENDS IN SURVEILLANCE AND TRACKING SENSITIVITY

SOURCE: Aerospace Industries Association (Ref 8.1)

Current radome and irdome techniques are not satisfactory for use in high speed aircraft and low altitude satellites. A means of locating and tracking submarines from aircraft and/or space vehicles remains an urgent requirement.

8.2 SURVEILLANCE AND TRACKING - RESOLUTION

The anticipated growth in space vehicle size will have a significant effect upon surveillance and tracking system requirements. Figure 8-2 illustrates the expected effects on resolution requirements. Optical systems with aperture limited resolution of one foot at 300 nautical miles are desired. Even higher resolutions would be useful if atmospheric distortions permit. Achievement of such a capability calls for improved optical elements, optical system mechanical design, image motion compensation, higher resolution films and higher resolution electro-optical devices.

IR sensors have less inherent resolution than optical sensors. However, they have many valuable uses in tracking, weather observation and as supplements to optical mappers. Thus, similar improvements in IR resolution are needed.

The unique all-weather capability of radars calls for continuous emphasis on developing higher resolution systems for tracking and for mapping. Development of data processing methods and techniques which will extract all of the information in the return signal is a critical problem. Better recording and data storage methods and more effective displays are other needed improvements. Exploitation of millimeter through optical wave lengths for space tracking is an urgent problem.

REFERENCE LIST FOR SECTION 8

- 8.1 "Aerospace Technical Forecast 1962-1972," Aerospace Industries Association, 1725 DeSales Street N.W., Washington 6, D. C.

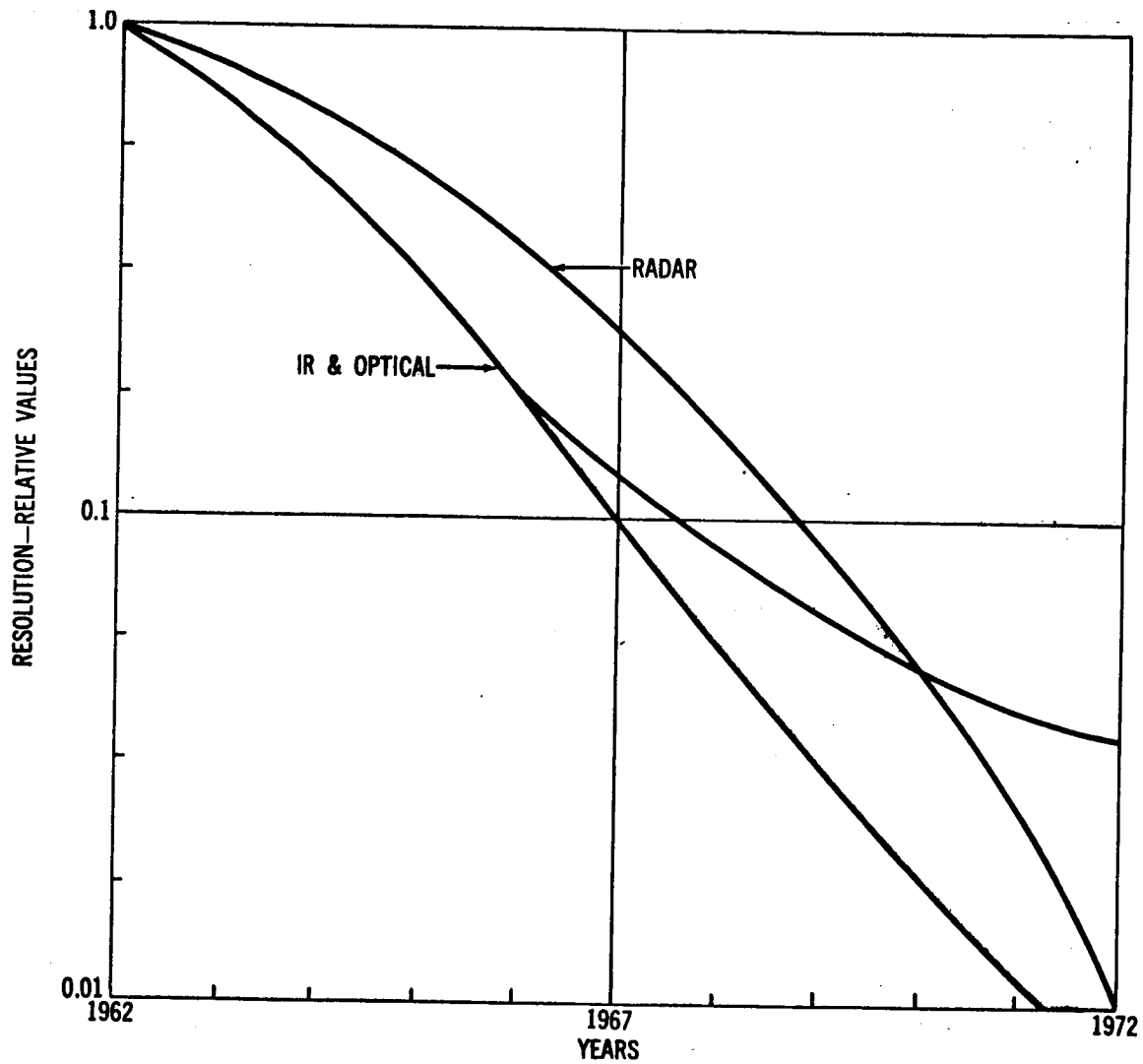


FIGURE 8-2. TRENDS IN SURVEILLANCE AND TRACKING RESOLUTION

SOURCE: Aerospace Industries Association (Ref 8.1)

Section 9

PROPULSION

9.0 GENERAL COMMENTS

Propulsion encompasses all thrust generation functions used to accelerate or propel the vehicle. Included are all functional elements such as rocket motors, propellant, propellant tankage, storage and thrust control devices, and associated instrumentation. Specifically excluded are thrust generators such as reaction jets designed to provide the stability and control function.

Propulsion system improvement is the fundamental requirement for progress in all types of flight vehicles. Propulsion improvement may be categorized in three major groupings: larger payloads to Earth orbit, maneuvering capability in space, and unconventional propulsion techniques. Future propulsion requirements were estimated from References 20 through 28 and from other similar NASA documents supplied by Walter H. Stafford of NASA Marshall Space Flight Center.

In general, it is desirable to improve our capability to launch heavier payloads into Earth orbit. There are five general methods by which this might be accomplished:

Boost Upgrading

Larger liquid rocket boosters (NOVA, SEA DRAGON, etc.)

Thrust augmentation of present boosters by means of solid rocket strap-ons.

Air breathing (turbo-ramjet-scamjet). Recoverable boosters or horizontal take-off boosters.

Catapult launch of large spacecraft/booster assemblies horizontally.

Rendezvous in Earth orbit of many smaller payloads (assembly of interplanetary boosters in Earth orbit, for example).

In addition, it is desirable to upgrade our capability for maneuvering larger payloads throughout space. There are six general approaches for accomplishing this:

Upper Stage Upgrading

High energy upper stages or "kick stages".

Nuclear fission rocket upper stages.

Nuclear pulse upper stages.

Use of the moon as a rendezvous station for assembly and
"LUNATRON" Catapult launch of large spacecraft from a lunar
"railway".

Electric and plasma accelerator upper stages with very high
specific impulses.

Solar sails.

Research into other (far out, or super "blue sky") areas of propulsion and lift technology may conceivably unlock natural keys to easier space flight. The areas with some remote promise are:

Far Out Concepts

Natural force fields (electric, magnetic, undefined F, "flying saucer" research, spirit levitation),

Photon rockets (bright idea with a dim promise),

"Ionocraft" electric levitation within the stratosphere (Seversky),

Contragravity inertial accelerators (Davis "law" machines),

Bi-location, time control or "teleportation" (angel flight),

Fusion rockets (Sanger's concepts).

In all cases (except the ionocraft), there is a basic lack of understanding of the nature of time, space, matter, energy, and motion and their interactions with "field forces", particularly "gravity". Breakthroughs in basic physics are needed before work can proceed.

The purpose of this section is to provide background material to assist the voter in assigning relevance numbers to the various approaches actually chosen for the concepts.

The following charts (Figures 9-1 through 9-10) are taken from discussions with V. Gradecak, NASA, Marshall Space Flight Center. Figures 9-11 through 9-13 are self-explanatory, and are included for general reference.

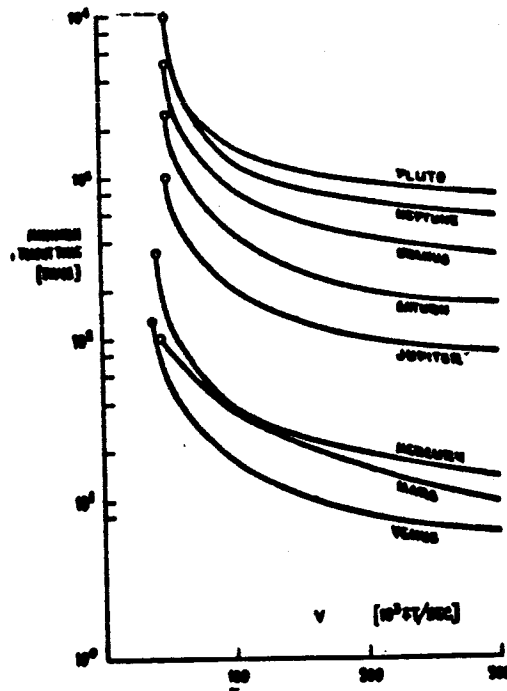


FIGURE 9-1. EARTH TO PLANETS ESTIMATED MINIMUM FLIGHT TIMES, BALLISTIC INTERORBITAL TRANSFER
(Source: V. Gradecak)

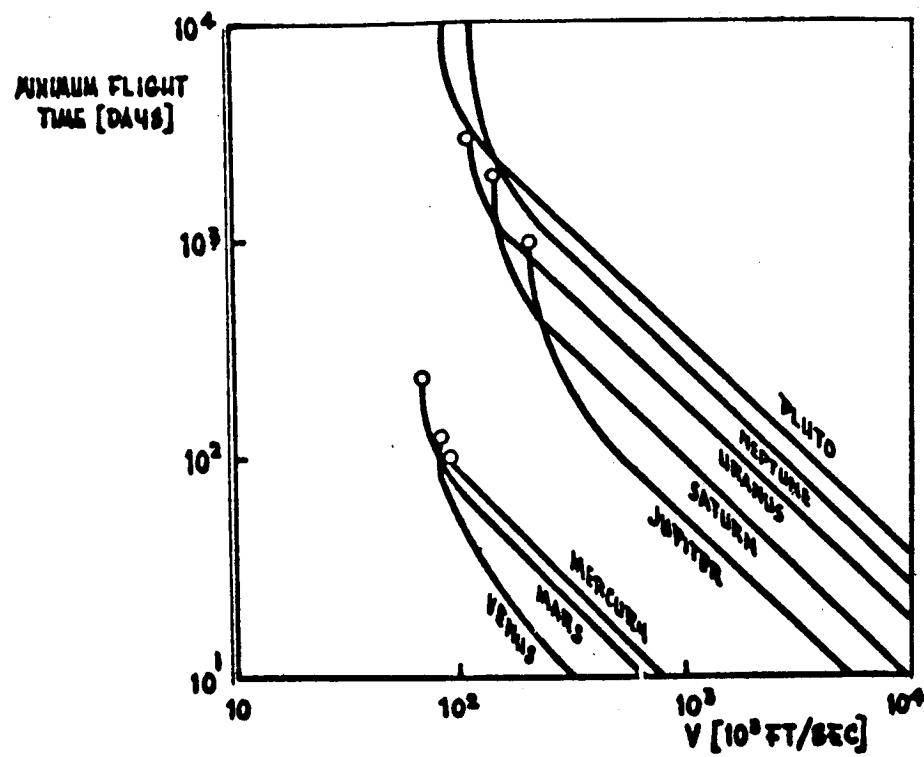


FIGURE 9-2. EARTH TO PLANETS ESTIMATED MINIMUM FLIGHT TIMES, BALLISTIC SURFACE TO SURFACE TRANSFER
(Source: V. Gradecak)

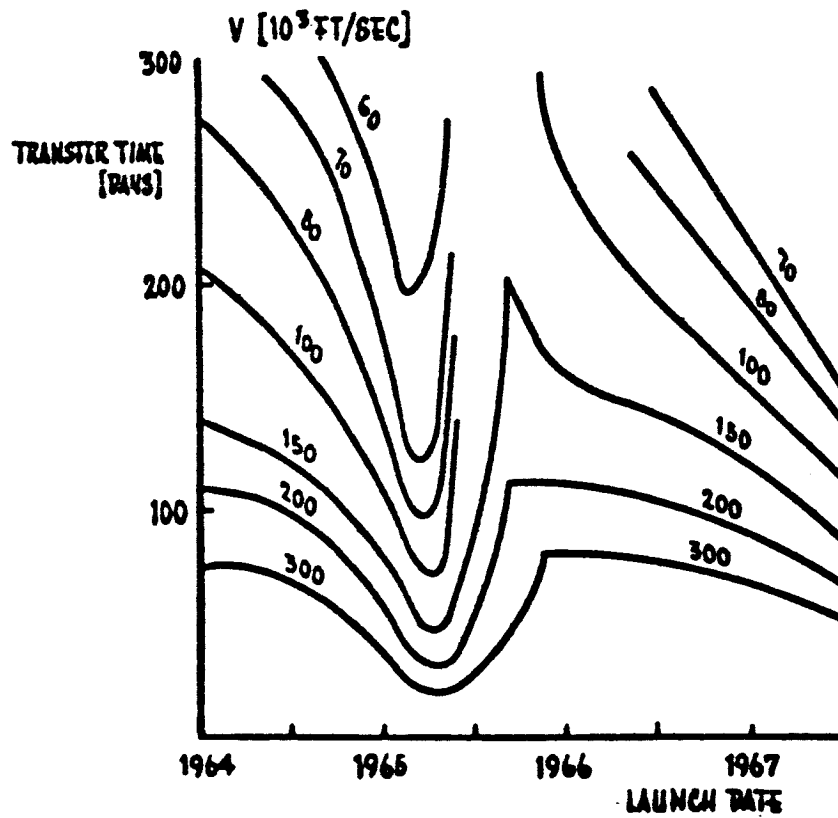


FIGURE 9-3. EARTH TO MARS MINIMUM FLIGHT TIMES,
IMPULSIVE VELOCITY REQUIREMENTS BALLISTIC
SURFACE TO SURFACE TRANSFER
(Source: V. Gradecak)

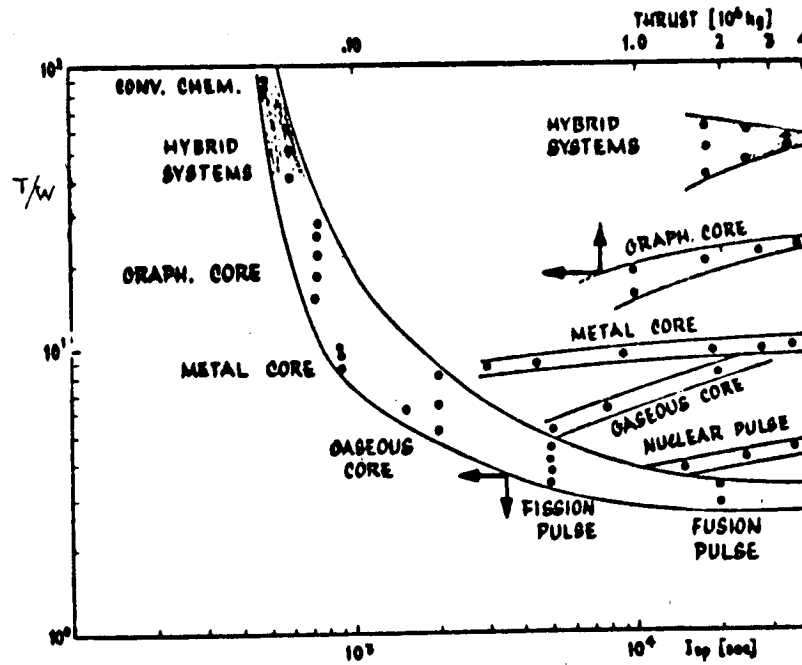


FIGURE 9-4. T/W RATIO VERSUS I_{sp} FOR HIGH THRUST PROPULSION SYSTEMS
(Source: V. Gradecak)

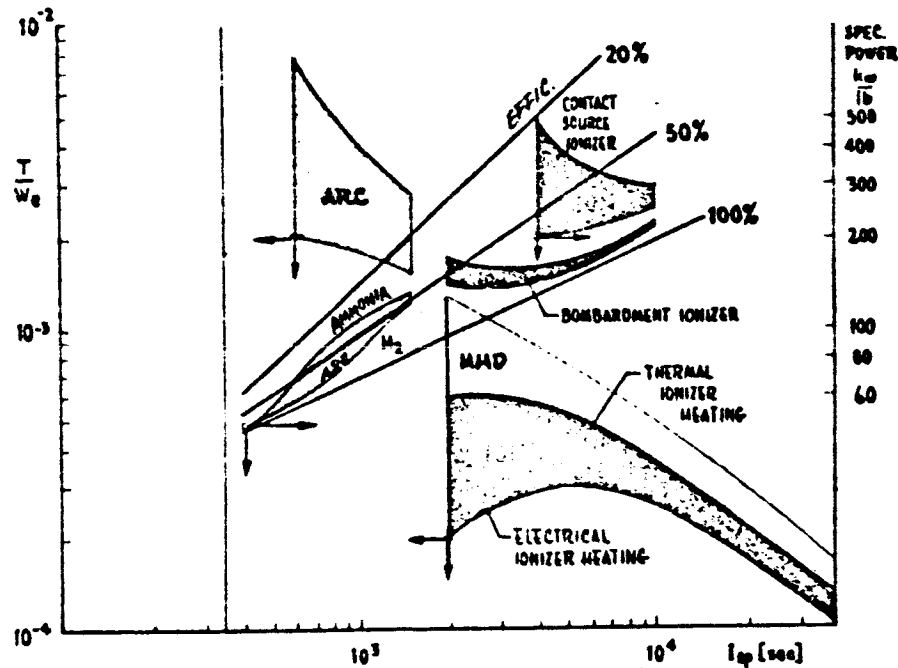


FIGURE 9-5. T/W_e VERSUS I_{sp} FOR LOW THRUST PROPULSION SYSTEMS
(Source: V. Gradecak)

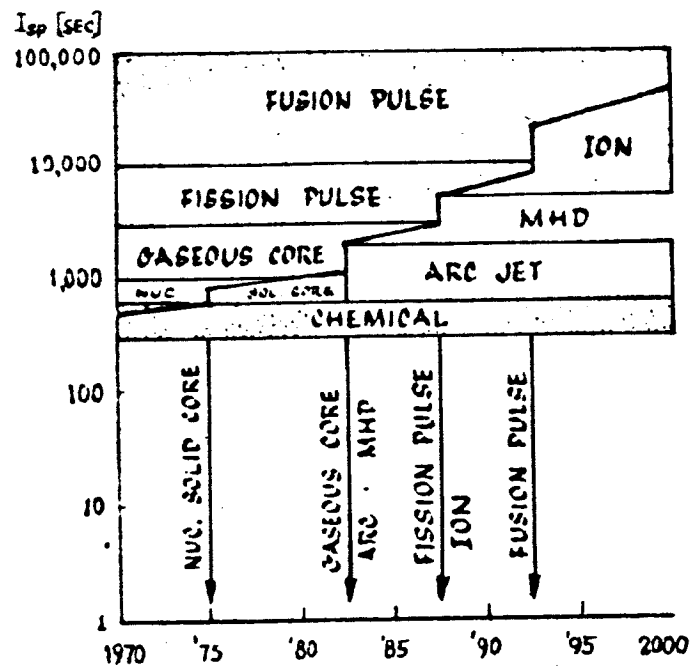


FIGURE 9-6. ASSUMED OPERATIONAL DATE OF ADVANCED PROPULSION SYSTEMS
(Source: V. Gradecak)

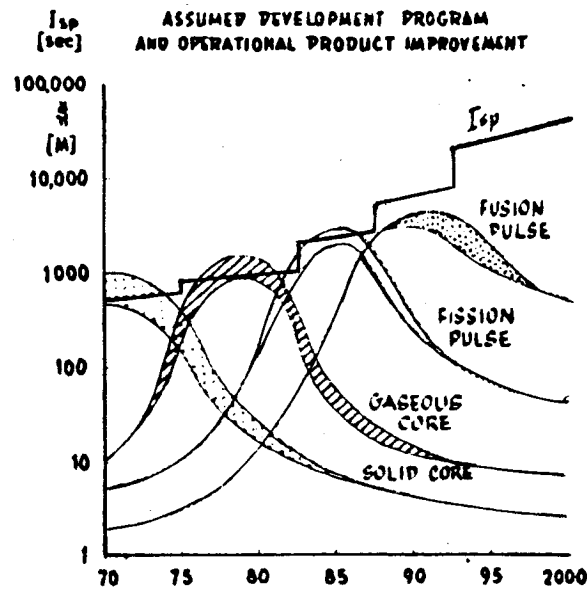


FIGURE 9-7. I_{sp} FORECAST FOR HIGH THRUST
PROPULSION SYSTEMS
(Source: V. Gradecak)

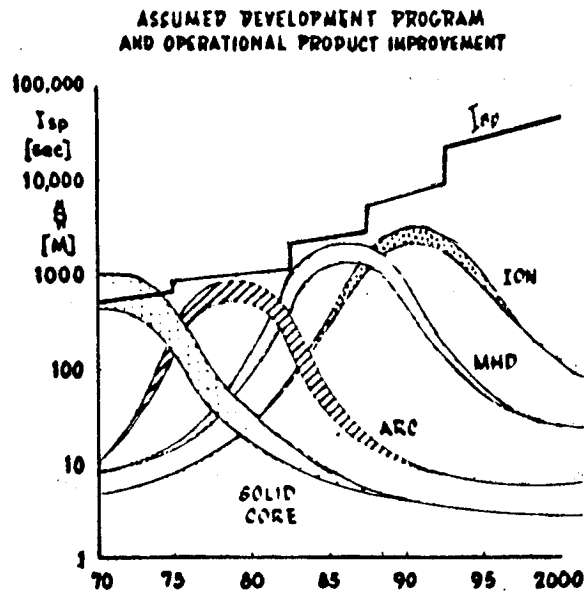


FIGURE 9-8. I_{sp} FORECAST FOR LOW THRUST
PROPULSION SYSTEMS
(Source: V. Gradecak)

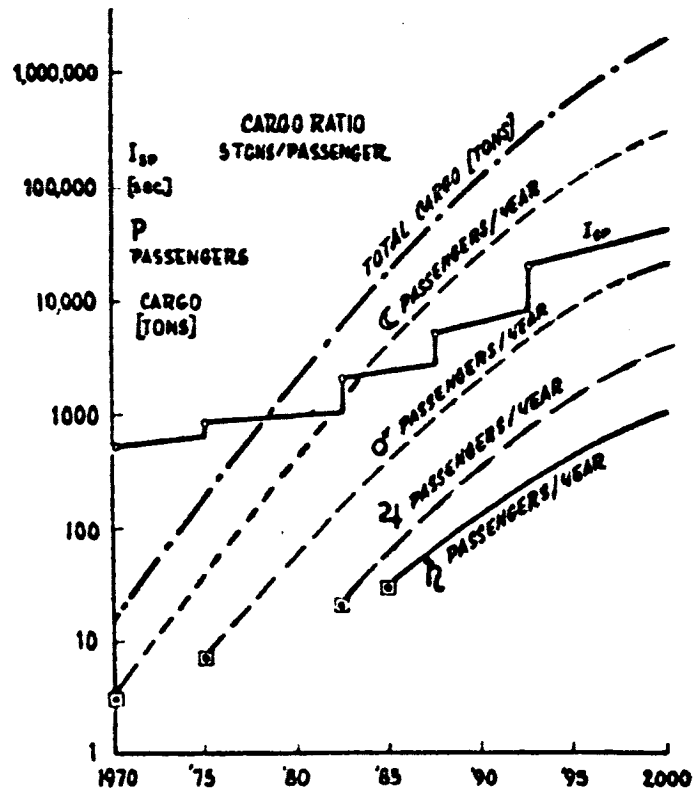


FIGURE 9-9. ESTIMATED FUTURE FEASIBLE TRANSPORT VOLUME
OF INTERPLANETARY PASSENGERS
(Source: V. Gradecak)

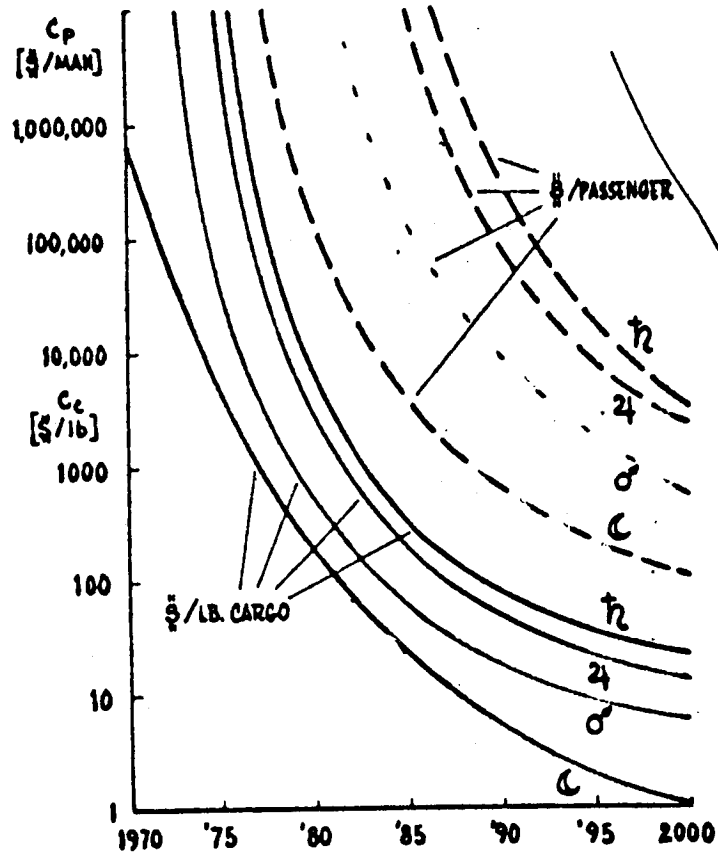


FIGURE 9-10. ESTIMATED OPERATING COSTS FOR
FUTURE INTERPLANETARY TRANSPORTATION
(Source: V. Gradecak)

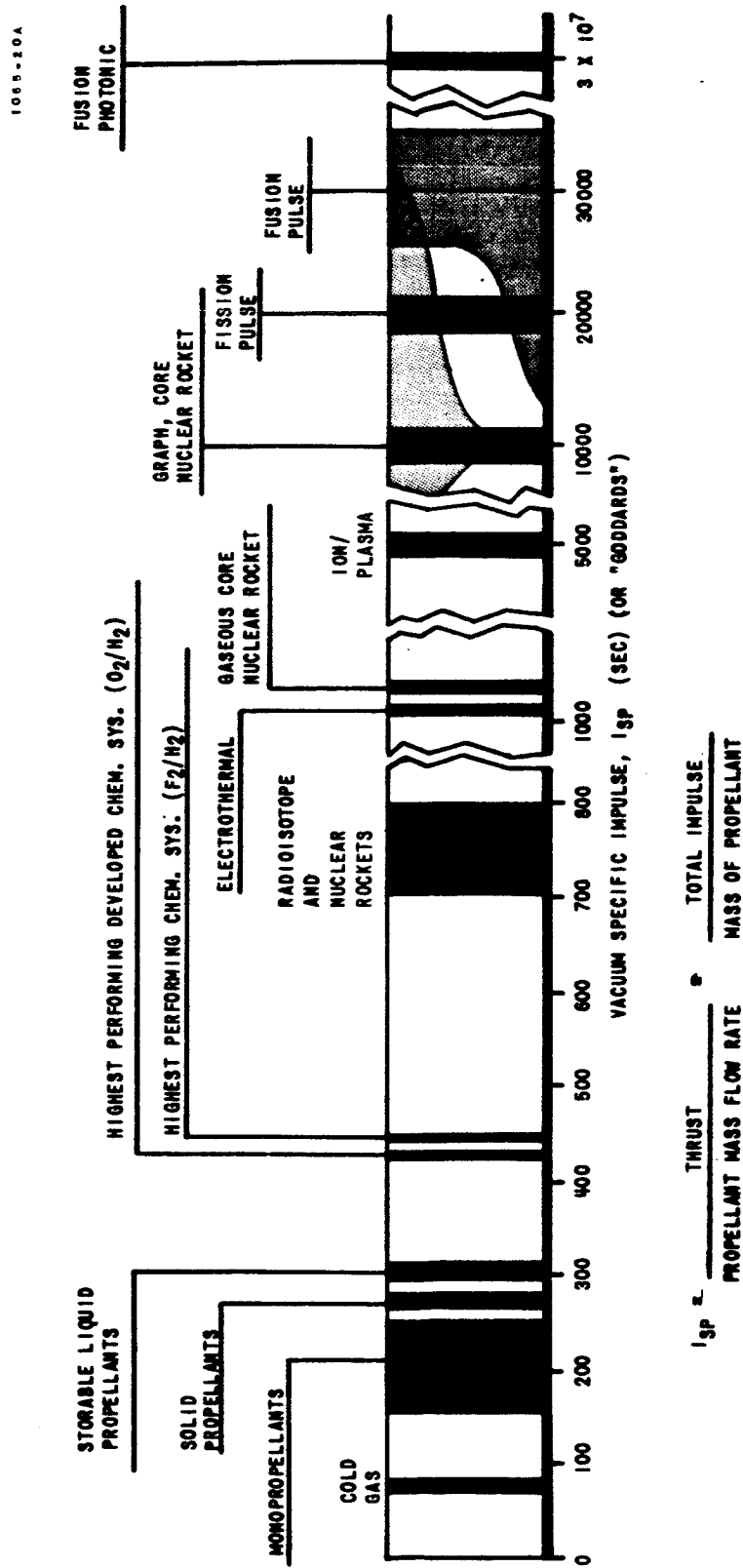


FIGURE 9-11. APPROXIMATE SPECTRUM OF ROCKET I_{sp}
(Sources: C. Benfield, W. R. Corliss)

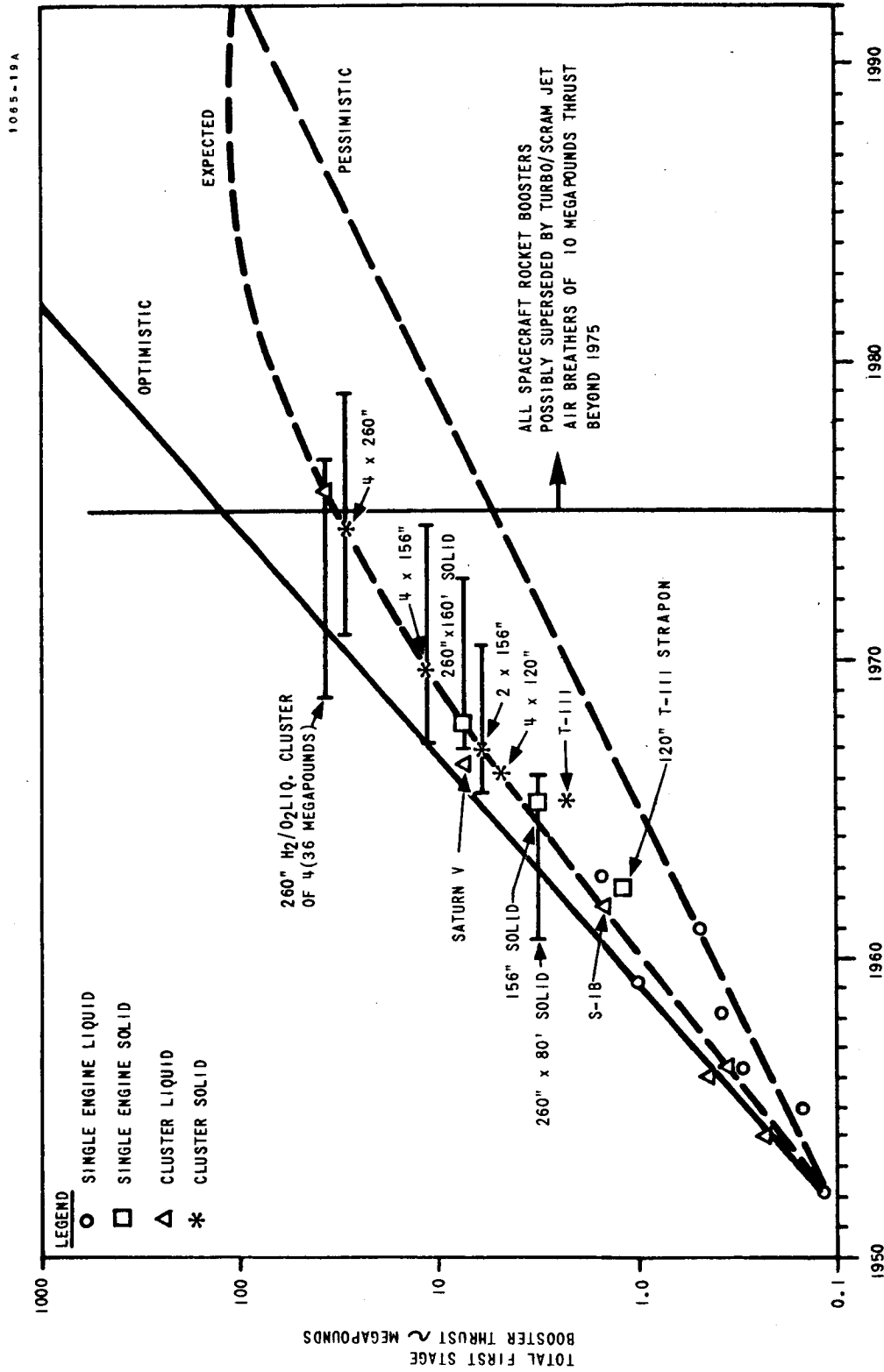


FIGURE 9-12. BOOSTER THRUST FORECAST
(Sources: Ref. 3, 4, 5 & 6)

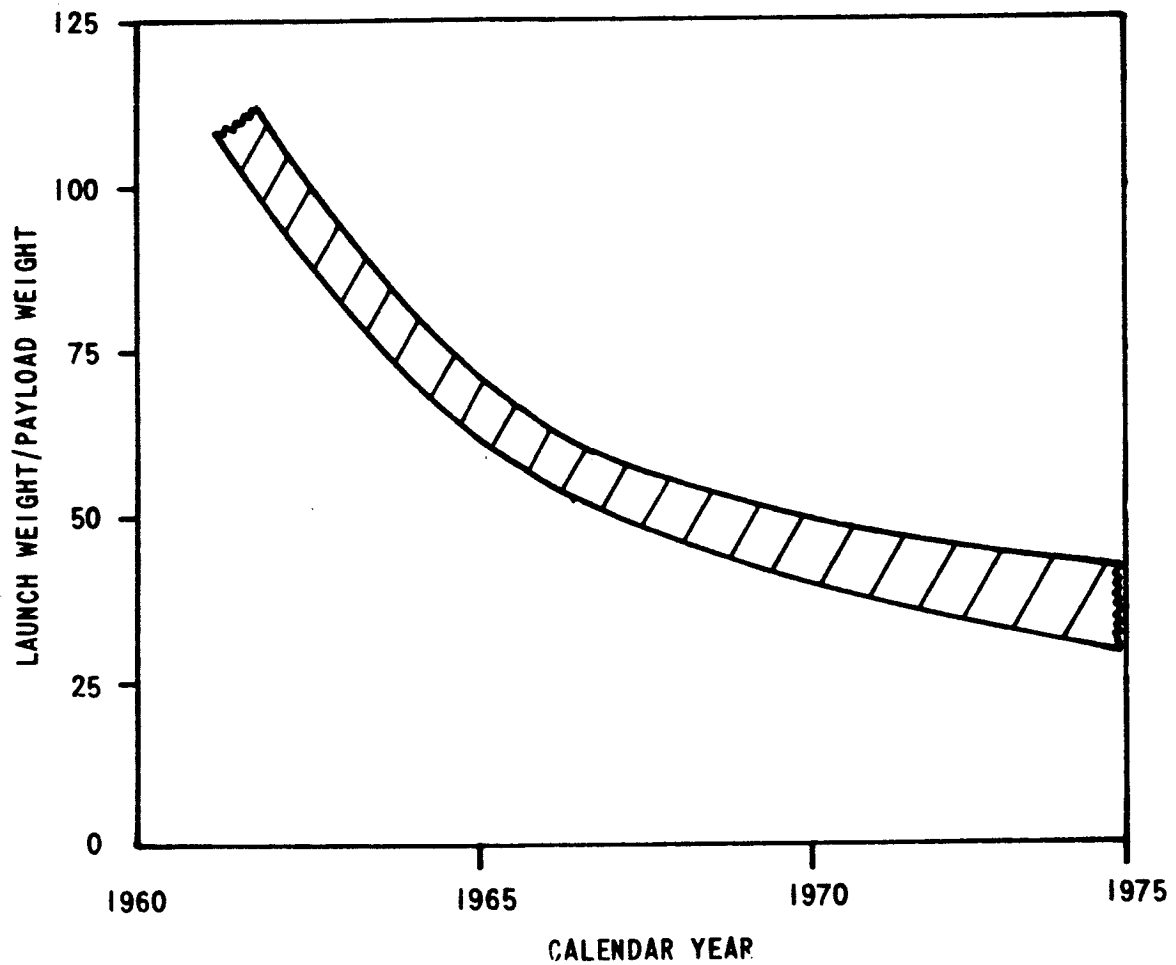


FIGURE 9-13. FORECAST OF LAUNCH VEHICLE
WEIGHT PER POUND OF PAYLOAD IN ORBIT
(Internal Honeywell Source Unidentified)

The following is quoted from Reference 3 (Space/Aeronautics, Paul Roman), referring to Figures 9-1 and the color insert following page 9-17.

The United States will soon have a fairly broad-gaged space transportation system. It wasn't really planned that way, but that's the way it is turning out. And the fact that it is being looked at more as an integral system than as a collection of launch vehicles helps explain the cautious, possibly reluctant approach toward making any new changes or additions to it.

At the top of the system is Saturn 5, and for some \$115 million or so (hardware but not R&D costs of the launch vehicle itself, plus software costs associated with launch) it will send 120 tons into low Earth orbit, 45 tons to the moon, 35 tons to Mars, or two tons right out of the solar system. A much less ambitious example is Atlas/Centaur, which for a cost of about \$12 million will send five tons into Earth orbit, more than a ton to the moon, or half a ton to Mars. At the bottom is Scout, which for a mere \$1 million or so will send a few hundred pounds into low orbit.

The system of ten basic launch vehicles (Figure 9-14) will not be completed until 1968 or 1969-- that's when Saturn 5 will become operational. And before then, Atlas/Centaur, Titan 3C and Saturn 1B will have joined the existing fold of Titan 2, Atlas/- and Thor/Agenas, Thor/Ablestar and Delta, and Scout.

With this transportation system now well defined and almost in hand, the question facing NASA and Air Force space booster planners is: what next? For a transportation system is never completed.

There are many options from which to choose: should some of today's boosters be refined to a high point of reliability and perhaps a lower point in cost? Should new techniques be used to uprate existing vehicles? Should gaps in our present capability be filled by the development of new stages? And should some existing boosters be dropped?

The question, then, really is: what order of priority should be given to these various courses? A clearcut answer does not appear to be forthcoming either from NASA, which has a number of unfinished missions for its developing vehicle stable, or from the Air Force, which is still searching for missions for its showpiece, Titan 3C. At NASA, particularly, the impression given is of intentionally not placing any bets, but at the same time not doing anything that would rule out any bets (excepting a de-emphasis on a post-Saturn vehicle). Another way of looking at it, according to one

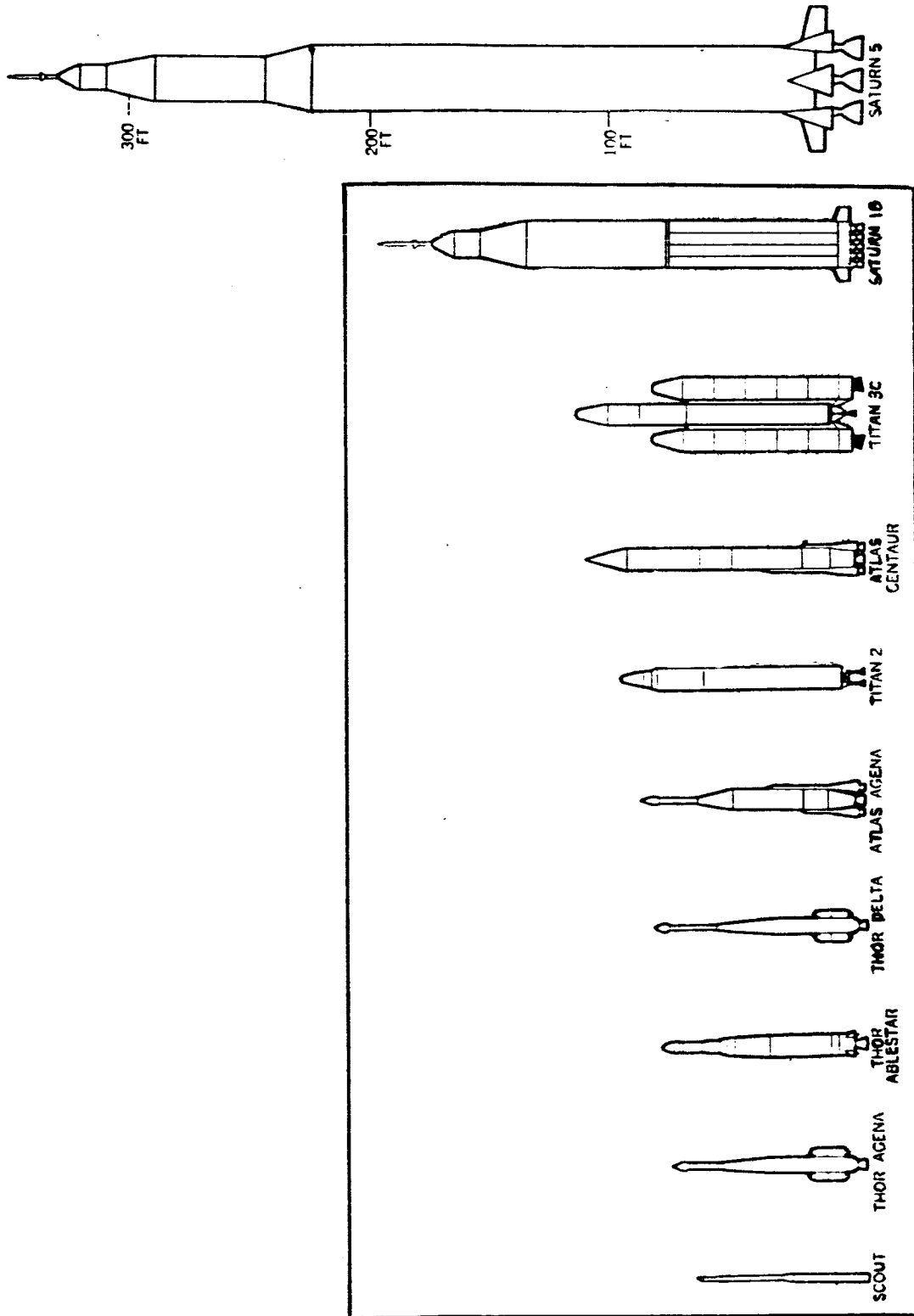


FIGURE 9-14. U. S. LAUNCH VEHICLES
(Source: Ref. 3)

NASA official, is that booster people for the first time have gained breathing space. The stable that goes up through Saturn 5 can accomplish a wider assortment of missions than is now on the books, and there is no committed mission for which a booster does not exist, as there was when Apollo was decided upon.

This breathing space most surely will be used to place more emphasis on costs. For with an adequate stable already in hand, future modifications can be eyed more critically, especially if there are competing ways of accomplishing the same objective. At the same time, there will be greater reliance on the building-block concept. For example, if a new kick stage is to be developed, as seems likely, it undoubtedly will have to be adaptable to a wide range of vehicles, and not tailor-made for just one vehicle.

The results of increased cost-consciousness can be illustrated in another manner. At NASA's Office of Space Science and Applications, there is not much doubt that the small kick stage would be a highly useful addition to the launch stable. Placed on top of an Atlas/Centaur, say, it would increase that booster's lunar transfer capabilities by 50 percent or so, or would give it an entirely new capability-- probes to the far planets.

Of the several options open in the evolution of this space transportation system, the one that is least a choice and most a necessity is that of increasing reliability of service. On a 20-point moving average (the last 20 launches of a particular vehicle), Thor/Delta now has an 85-percent reliability record (it had 100 percent for a while), Thor/Agena stands at 95 percent, Atlas/Agena at 90 percent and Scout at 75 percent. How much effort should be made to improve reliability? There is a certain merit in the view that as launchings become more frequent and routine, the standard of excellence demanded of them will also rise. Will reliability keep pace with this rising standard, or are extra development efforts required?

Another way of improving service is to increase the performance of existing boosters. Thor, Atlas, and the Saturns could have their payload capability increased by up to 30 percent through enriching the oxygen with fluorine-- so-call floxing. The Titans could gain a similar increase in payload capability by using high-energy thixotropic fuels. Should these experimental techniques be advanced to the hardware stage?

The use of solid strap-on motors on the first stage is another method for significantly improving the performance of existing boosters, as the Thrust-Augmented Thor and Delta have successfully demonstrated. Titan 3C capability could be increased up to 40 percent by substituting 156-inch strap-ons for the present 120-inch ones. Even more significant increases in capability could be given to the Saturns by addition of strap-ons. But again, is this the best way to improve performance?

Other ways of filling the gaps are to use new combinations of existing vehicles. Agena could be put on top of a beefed-up Titan 2 (a configuration now referred to as Titan 3X/Agena). Centaur might be used in the same way, as a substitute for the Transtage in Titan 3C, or on top of Saturn 1B.

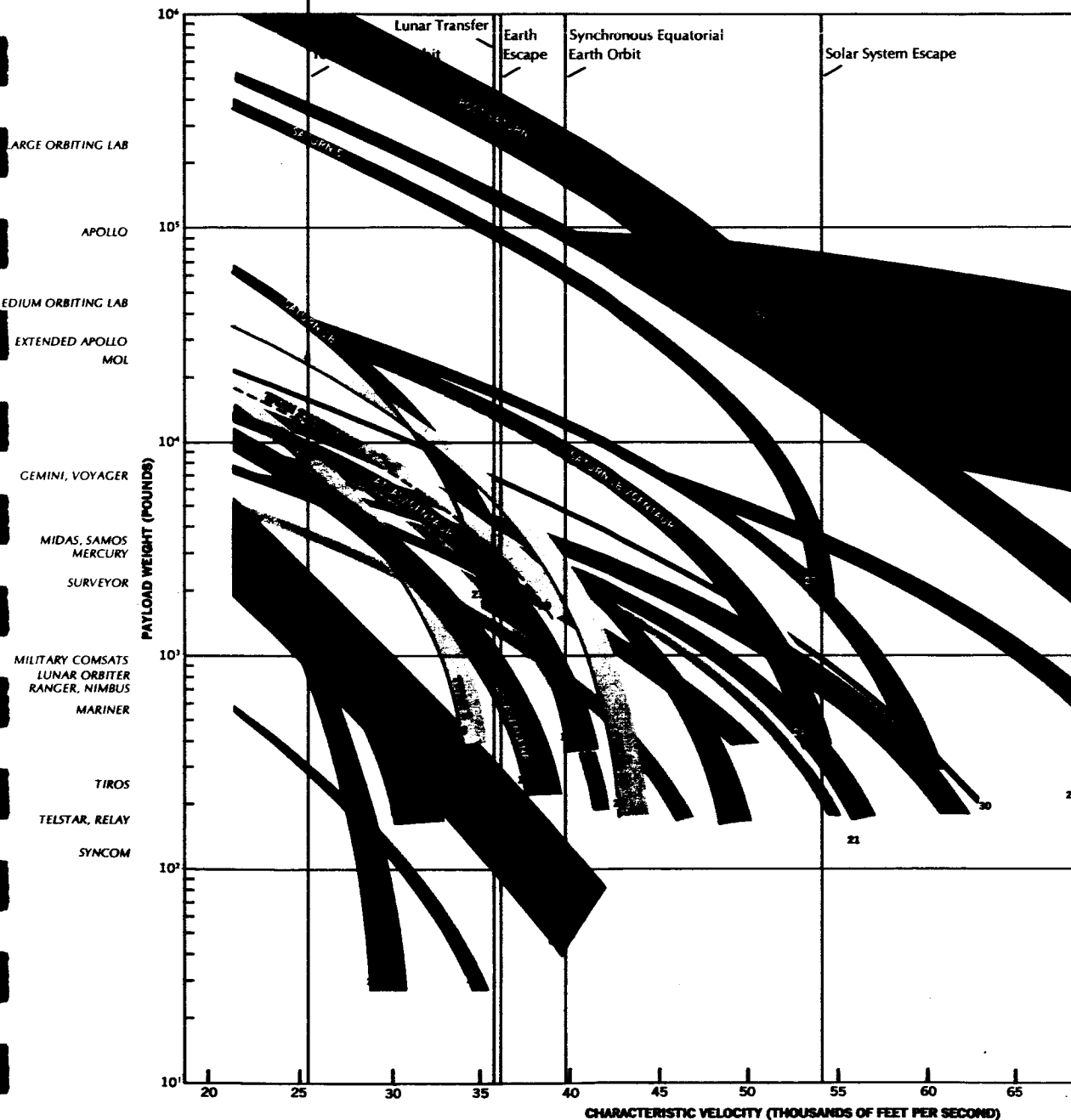
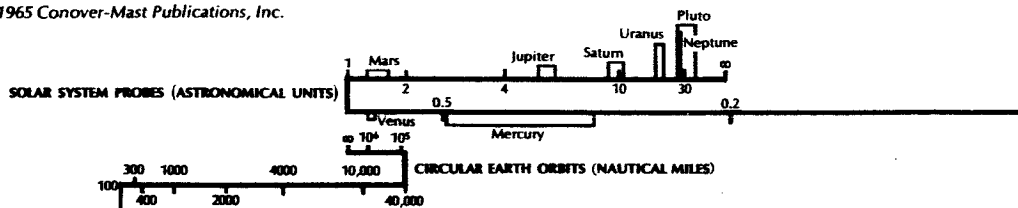
Although the hydrogen-burning Centaur is the highest-performing stage in the present booster stable, there is already talk of a post-Centaur stage having thrust in the range of 40,000 pounds (as against 30,000 for Centaur). With liquid fluorine and hydrogen as the propellants, specific impulse would be about 455 seconds, compared to Centaur's present 433 and anticipated 440-444 seconds.

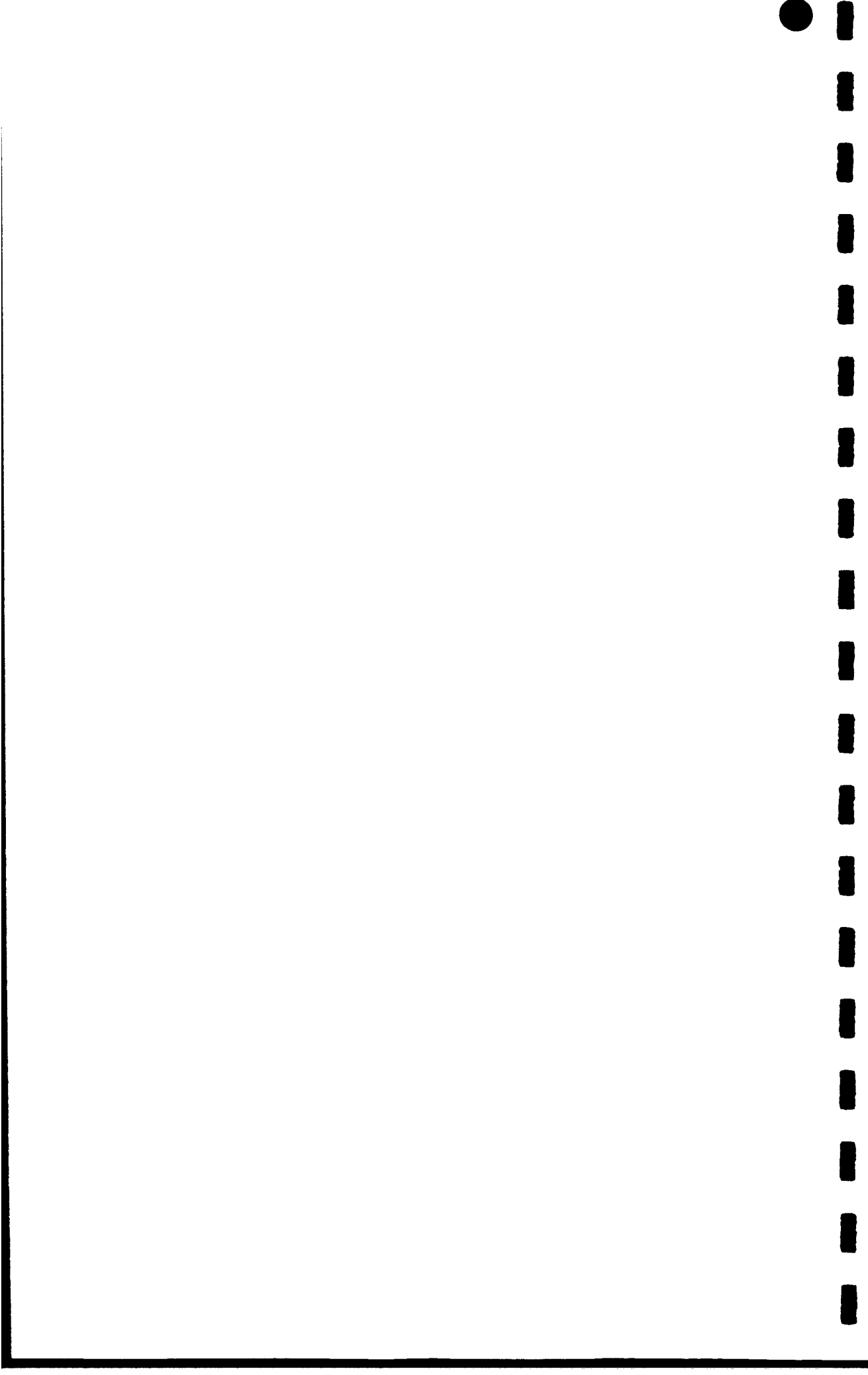
Another new stage being talked about, in effect a post-Atlas vehicle, is being advocated not so much to fill a gap, but as a much lower-cost vehicle to replace Atlas and Titan 2. The assumption is that boosters roughly have the same launch cost (including cost of booster) per pound of lift-off weight. Therefore, a high-performance vehicle burning liquid fluorine and hydrogen would weigh less (and cost less) for the same performance.

Finally, there is the question of post-Saturn. No one has the answer here, for it lies beyond another large question mark: the nature of post-Apollo missions. Nevertheless a post-Saturn launch vehicle remains one of the many options facing the managers of the nation's newest transportation system as they cautiously explore ways of improving it. (End of quote from Reference 3.)

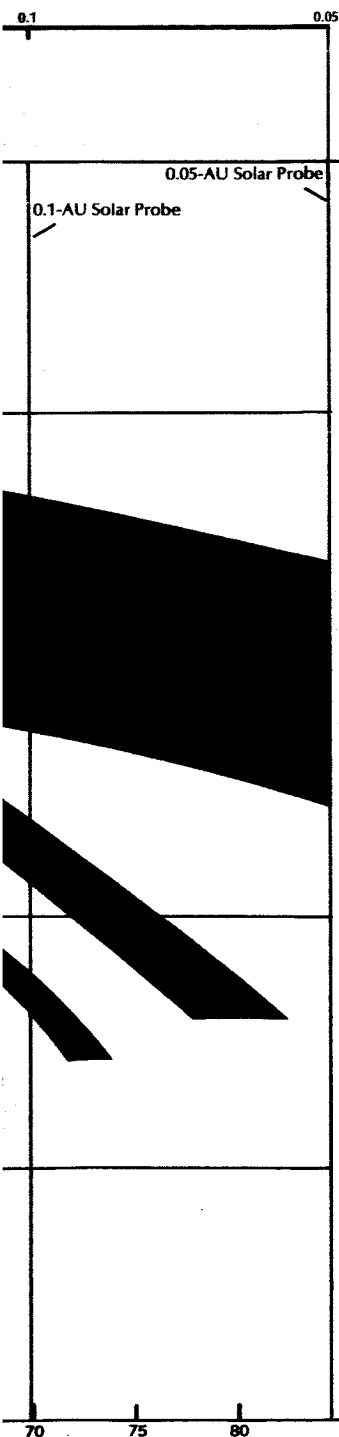
SPACE/AERONAUTICS LAUNCH VEHICLE CAPABILITIES

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-PAYLOADS & DESTINATIONS



LEGEND

Numbered curves at left show capabilities of various combinations of booster stages as defined below. Three hypothetical stages are frequently referred to under the notation of 7K, 40K, and 235K. These numbers denote the thrust in thousands of pounds that these liquid fluorine and hydrogen-fueled stages would generate.

1—Scout.

2—Thor/Ablestar; 3—Thrust-Augmented Thor/Ablestar; 4—Thor/Agena; 5—Thrust-Augmented Thor/Agena; 6—Thor/Delta; 7—Thor/Improved Delta and Thrust-Augmented Thor/Delta; 8—Thrust-Augmented Thor/Improved Delta; 9—Thrust-Augmented Thor/40K stage; 10—Thrust-Augmented Thor/Agena/7K stage; 11—Thrust-Augmented Thor/40K stage/7K stage.

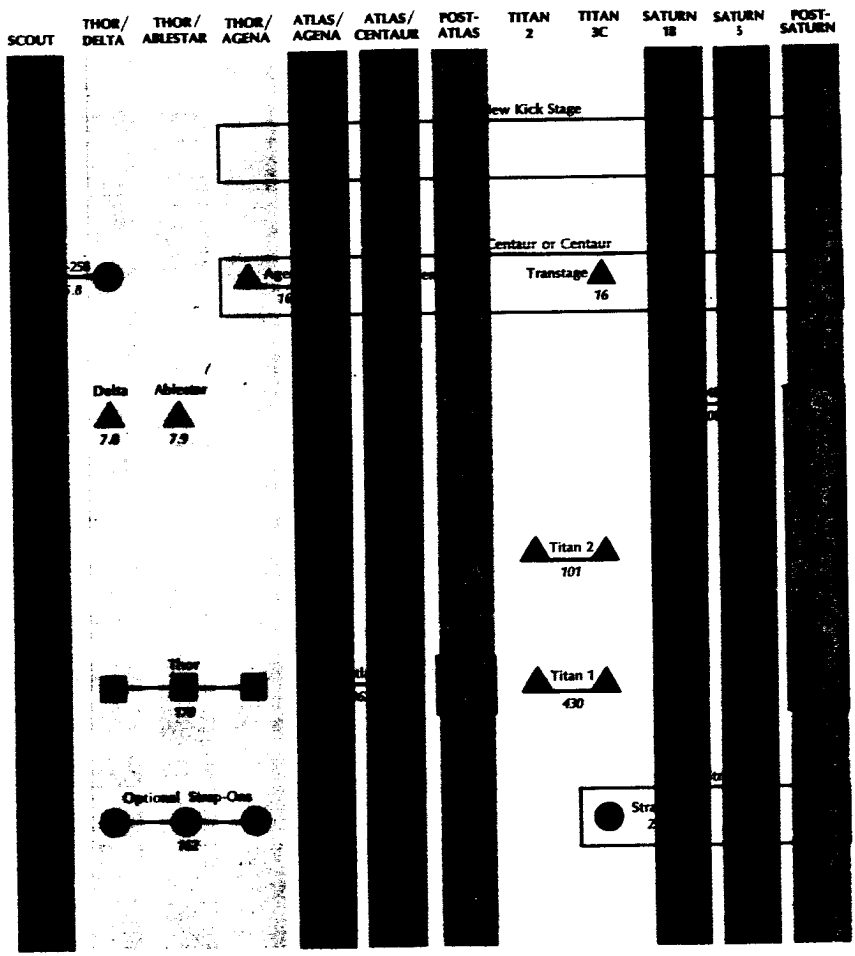
12—Atlas/Agena; 13—Atlas/Centaur; 14—Atlas/40K stage; 15—Atlas/Agena/7K stage; 16—Atlas/Centaur/7K stage; 17—Atlas/40K stage/7K stage.

18—Titan 2; 19—Titan 2/Centaur; 20—Titan 3C; 21—Titan 3/40K stage; 22—Titan 3/40K stage/7K stage.

23—Saturn 1B; 24—Saturn 1B/Centaur; 25—Saturn 1B/40K stage; 26—Saturn 1B/40K stage/7K stage; 27—Saturn 5; 28—Saturn 5/nuclear upper stage; 29—Saturn 5/nuclear-electric upper stage.

30—Post-Atlas 235K stage/40K stage/7K stage; 31—Post-Saturn.

LAUNCH VEHICLES BY STAGES



Legend—Vertical bars represent U.S. launch vehicles, color coded by families, the same as in chart at left. Gray represents two possible future vehicles: post-Atlas and post-Saturn. Individual stages of each vehicle are represented by solid geometric symbols whose shapes denote propellant used in those stages: ● is solid propellant; ■ is liquid oxygen and RP-1; ◆ is liquid oxygen and hydrogen; and ▲ is earth-storable propellant—Aerozine-50 and N₂O₄ for the Titan stages, and UDMH and IRFNA on all other stages. Where the symbols of two or more vehicles are connected by a horizontal black line, such as ■—■, essentially the same stage is used. Numbers in italics give the thrust of the stage in thousands of pounds. Large outline rectangles denote further applicability of existing stages or of possible new stages.

The U.S. launch vehicle family is made up of a relatively small number of building-block-like stages (see chart above), but range of payload weight and destination (or mission) is quite broad, as the chart at left shows. The solid black curves there show actual capabilities of existing stage combinations or those under active development. Dashed lines refer to new combinations of these committed upper and lower stages. Dark-colored bands (coded according to families) show minimal potential improvement in capabilities, e.g. from 30% flexing. Maximum improvements could triple width of bands. Light colored bands show capabilities achievable with entirely new stages, sometimes on top of existing lower stages. They are hypothetical ones used only for comparative purposes. The nuclear and electric stages are those that might logically result from current research programs. The post-Saturn is defined only by capability likely to be desired. The horizontal characteristic velocity scale shows velocity a vehicle needs at 100-nm altitude (not orbit) to reach destination, assuming a minimum energy criterion.

TABLE 9-1. PAYLOAD CAPABILITIES
(Reference 7)

	Van- guard	Jupiter-C	Thor, Jupiter	Atlas, Titan	Saturn- class rockets†
Gross mass:					
Kilograms.....	10,000	22,700	45,400	90,800	
Pounds.....	22,000	50,000	100,000	200,000	
Thrust:					
Newtons.....	125,000	335,000	670,000	1,330,000	6,700,000
Pounds.....	28,000	75,000	150,000	300,000	1,500,000
Mission:					
480-km orbit:					
Kilograms.....	9	14	910	3,600	17,000
Pounds.....	20	30	2,000	8,000	37,000
Moon impact:					
Kilograms.....			360	1,400	5,000
Pounds.....			800	3,000	11,000
Moon satellite:					
Kilograms.....			220	680	2,700
Pounds.....			500	1,500	6,000
Moon soft landing:					
Kilograms.....			140	450	1,800
Pounds.....			300	1,000	4,000
Mars or Venus probe:					
Kilograms.....			270	1,100	5,500
Pounds.....			600	2,500	12,000
Jupiter probe:					
Kilograms.....			180	680	1,100
Pounds.....			400	1,500	2,500

† Nova class rockets would multiply these figures by approximately a factor of 4.

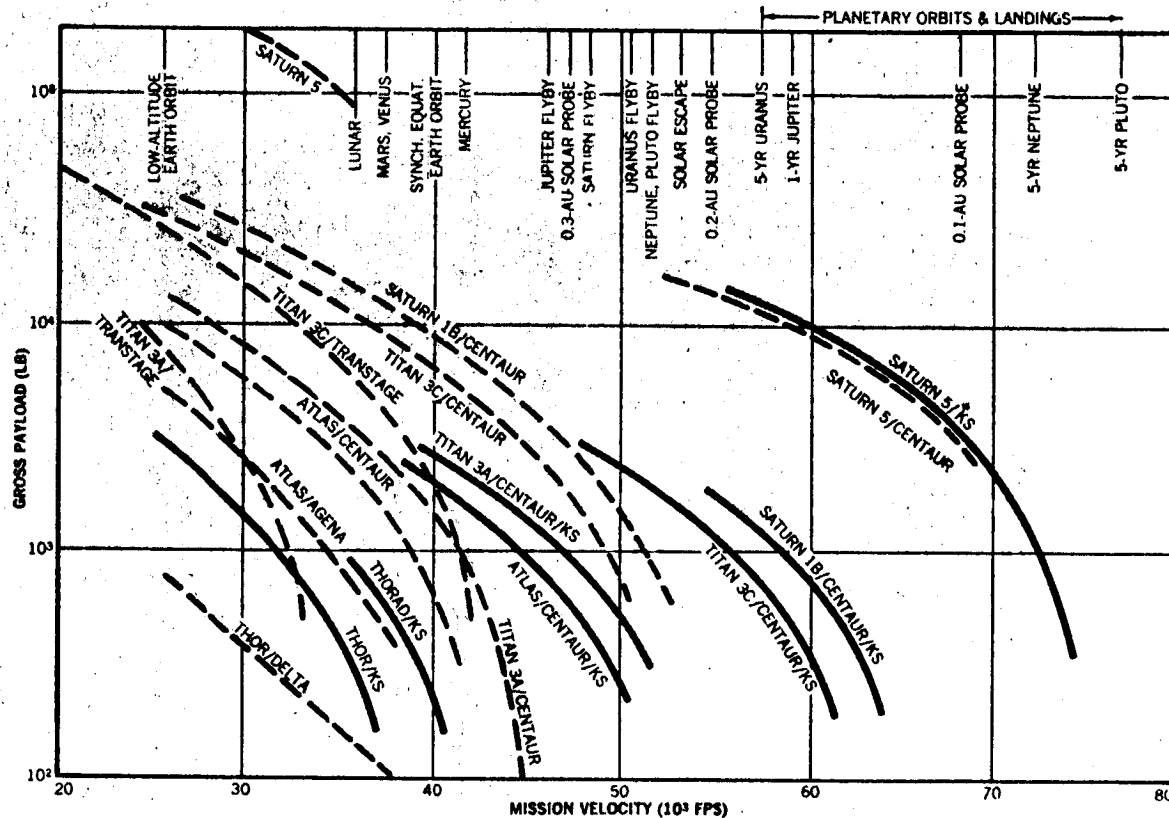


FIGURE 9-15. PERFORMANCE PROFILES
(Source: Reference 12)

Performance profiles of various launch vehicle combinations using existing and proposed high-energy upper stages. In virtually all cases, the use of a small (7,000 pounds), high-energy kick stage (ks) is a most effective approach. For example, simply adding the small fluorine/hydrogen stage to, say, the Atlas/Centaur extends the latter's ΔV capability well into the 40,000 to 50,000 fps range. At the same time, it allows the booster to place a significant experimental payload (generally pegged at around a minimum of 1,200 pounds) into the difficult synchronous equatorial orbit and into Mars, Venus, and Mercury probe trajectories-- a capability it doesn't otherwise enjoy. Short of using nuclear upper stages, rather far downstream in future technology, it's clear that the advanced planetary missions in the 50,000 to 70,000 fps class will require use of high-energy upper stages with the huge Saturn 5 booster.

TABLE 9-2. POTENTIAL HIGH-ENERGY MISSIONS
(Source: Reference 12)

Mission	Trajectory and Approximate Ideal Injection Velocity (fps)	Approximate Payload Capability (pounds) ^a			
		SLV-3X/ Centaur	SLV-3X/ Centaur/ Kick Stage ^{bc}	Saturn 1B/ Centaur	Saturn 1B/ Centaur/ Kick Stage ^c
Surveyor Follow-On	Direct-ascent launch; injected on lunar impact trajectory.	3,300	5,000	13,600	14,000
Voyager Mars	Earth escape; 42,000-48,000.	700/---	2,500/1,000	7,100/3,700	8,500/4,400
Voyager Venus	Earth escape; 42,000-43,000.	700/650	2,500/2,200	7,100/6,400	8,500/7,500
Voyager Test - Mars ('69)	Earth escape (parking orbit prior to injection); 42,000.	700	2,500	7,100	8,500
Planetary Explorer (comet intercept)	Earth escape; 40,500-46,000.	1,275/---	2,900/1,400	8,200/4,600	9,300/5,500
Planetary Explorer (asteroid flybys)	Earth escape; 42,000-44,000 (for Eros).	700/300	2,500/1,900	7,100/5,800	8,500/6,800
	42,000-46,000 (for Vesta).	700/---	2,500/1,400	7,100/4,600	8,500/5,500
	47,000 (for Ceres).	----	1,200	4,100	4,900
Jupiter Flyby	Earth escape; 52,500-55,000.	----	350/---	800/---	2,600/1,900
Mercury Flyby	Earth escape; 46,000-48,000.	----	1,400/1,000	4,600/3,700	5,500/4,400
Advanced Pioneer	Solar probe to 0.2-0.3 AU (anti-solar probe to 3.0 AU).	---- (500)	900 (2,300)	1,000/3,100 (7,000)	1,950/4,500 (7,900)
Synchronous Meteorological Satellite	Synchronous equatorial orbits, at approximately 90 degrees W viewing U. S.	1,450	3,200	8,900	10,000
Interplanetary Meteoroid	Earth escape; 42,000-48,000.	700/---	2,500/1,000	7,100/3,700	8,500/4,400
Communication & Navigation (CN-3) Satellite	Synchronous equatorial orbits; also establish and remain on station in deep cislunar space.	1,450/1,800	3,200/3,500	8,900/9,800	10,000/11,000
Direct Broadcast Satellite - FM Radio or TV	Synchronous equatorial orbits.	1,450	3,200	8,900	10,000
Circumlunar Meteoroid Detection Satellite	Circumlunar and/or highly elliptical (approximately 200,000-240,000 nm).	3,400	5,200	14,000	14,200
Optical Technology Satellite	Synchronous orbit.	1,450	3,200	8,900	10,000
Applications Technology Satellite (ATS-5)	Synchronous orbits.	1,850	3,550	9,900	11,000
Applications Technology Satellite (ATS-6)	Synchronous equatorial orbits.	1,450	3,200	8,900	10,000

^a Where two numbers are given, they represent the minimum and maximum payload corresponding to minimum and maximum escape velocities given in "Trajectory" column.

^b SLV-3X is the designation used for the standard launch vehicle based on the uprated Atlas.

^c The kick stage assumed for this payload capability is an 8,500-pound oxygen/hydrogen, 7,000-pound thrust vehicle (I_{sp} 450 seconds).

TABLE 9-3. PERFORMANCES OF CHEMICAL ROCKET FUELS

<u>Liquids</u>		<u>Chamber at 34 atm</u>		
<u>Oxidizer</u>	<u>Fuel</u>	<u>T_o, °K</u>	<u>M</u>	<u>I_{sp}, sec</u>
Hydrogen peroxide	Gasoline	2,930	21	248
Nitric acid	Gasoline	3,120	25	240
Oxygen	Alcohol	3,340	22	259
Oxygen	Hydrazine	3,240	18	280
Oxygen	Hydrogen	2,760	9	364
Fluorine	Ammonia	4,260	19	306
Fluorine	Hydrogen	3,090	9	373

<u>Solids</u>			
<u>Propellant System</u>	<u>Flame temp, °K</u>	<u>M</u>	<u>I_{sp}, sec</u>
KClO ₄ and C ₂ H ₄	1,800-3,030	25-35	165-210
NH ₄ ClO ₄ and C ₂ H ₄ O	1,800-2,150	22-25	175-240
Nitrocellulose and nitroglycerine, misc.	2,360-3,140	22-28	205-230
Asphalt and perchlorate	2,360-2,640	30	180-195

(from Ref. 7.)



**FIGURE 9-16. ESTIMATED PAYLOAD INTO EARTH ORBIT EACH YEAR
(Internal Honeywell Source Unidentified)**

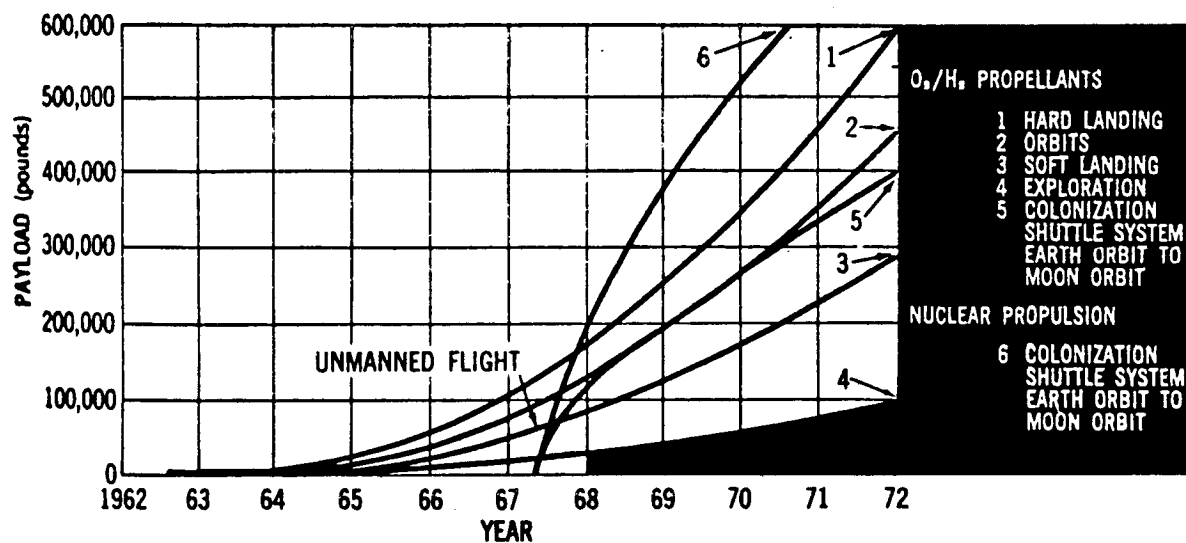


FIGURE 9-17. MOON MISSION PAYLOAD CAPABILITIES
(Source: Reference 1)

1085-1004

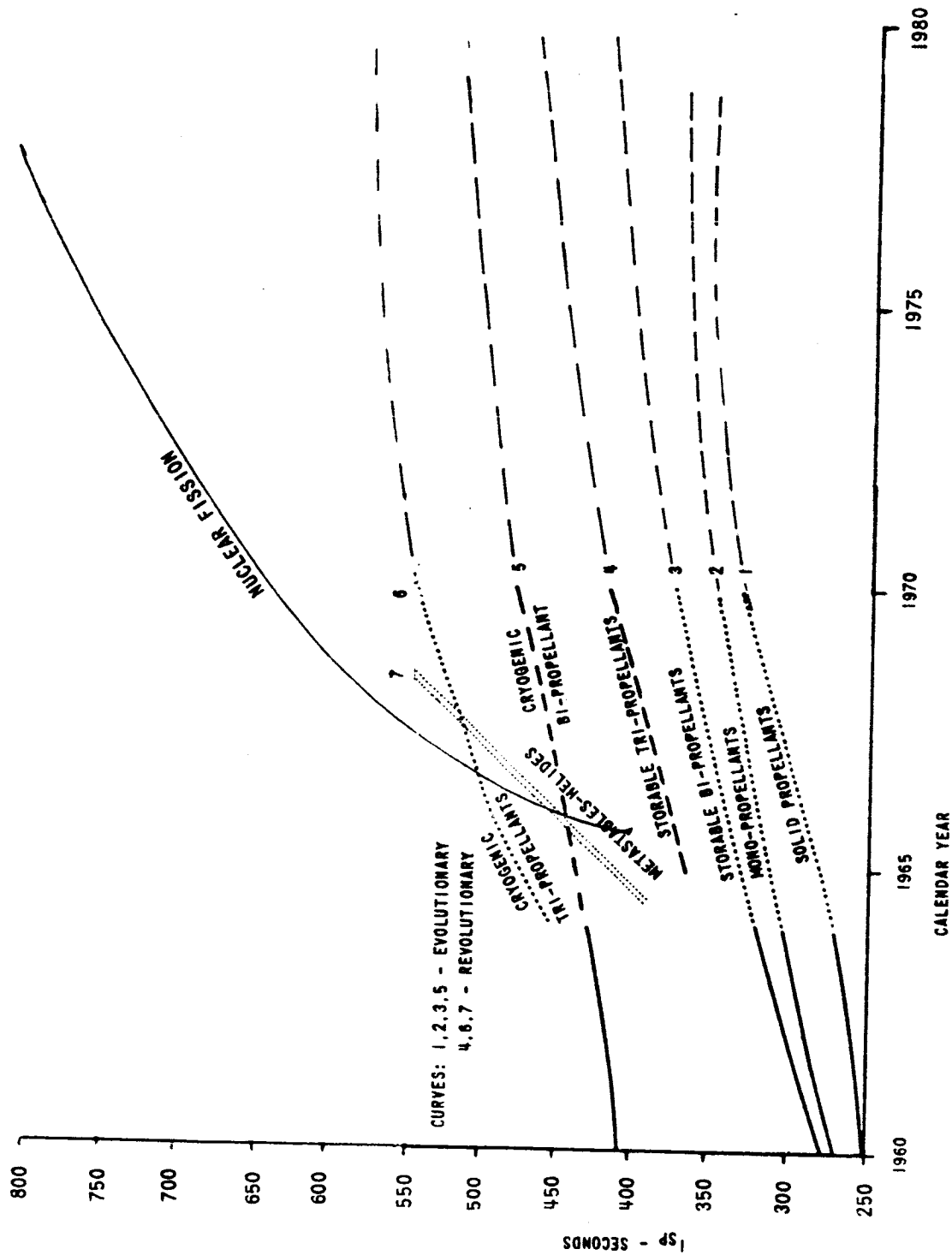


FIGURE 9-18. FORECAST OF SPECIFIC IMPULSE GROWTH
(Internal Honeywell Source Unidentified)

1065-101A

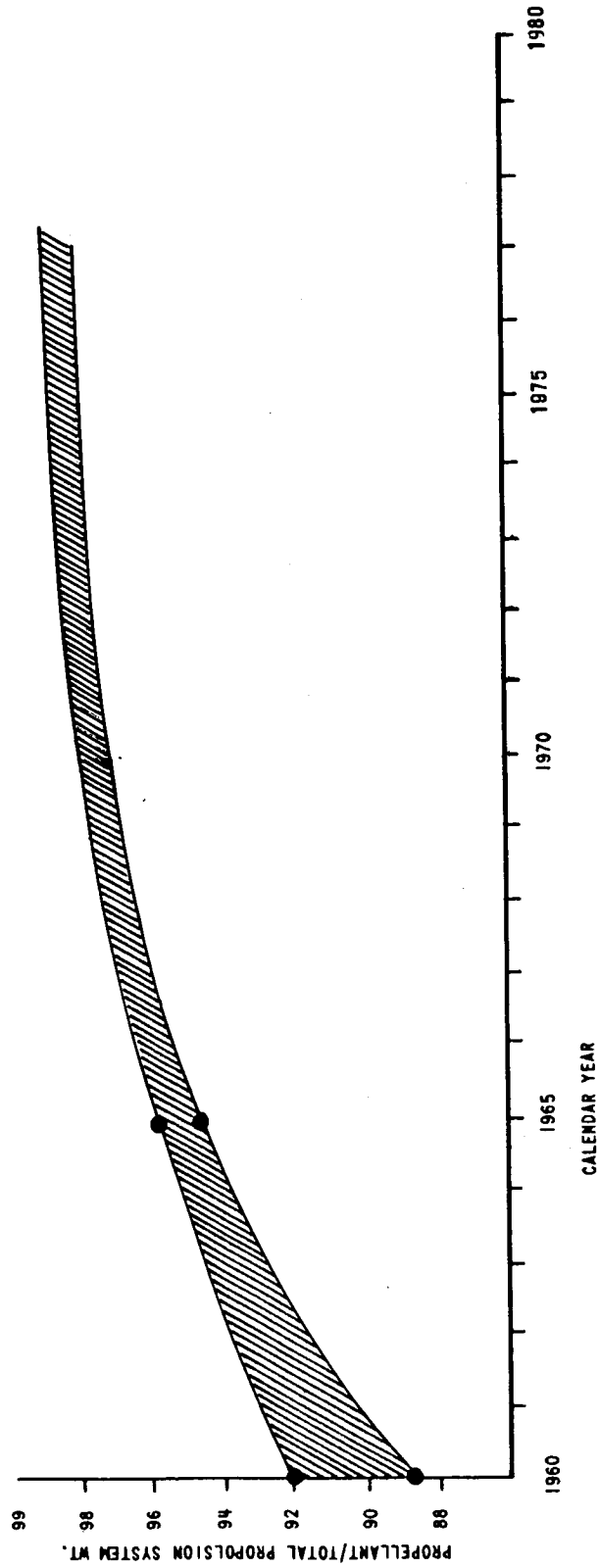


FIGURE 9-19. PROPELLANT MASS FRACTION - CHEMICAL ROCKETS
(Internal Honeywell Source Unidentified)

1065-102A

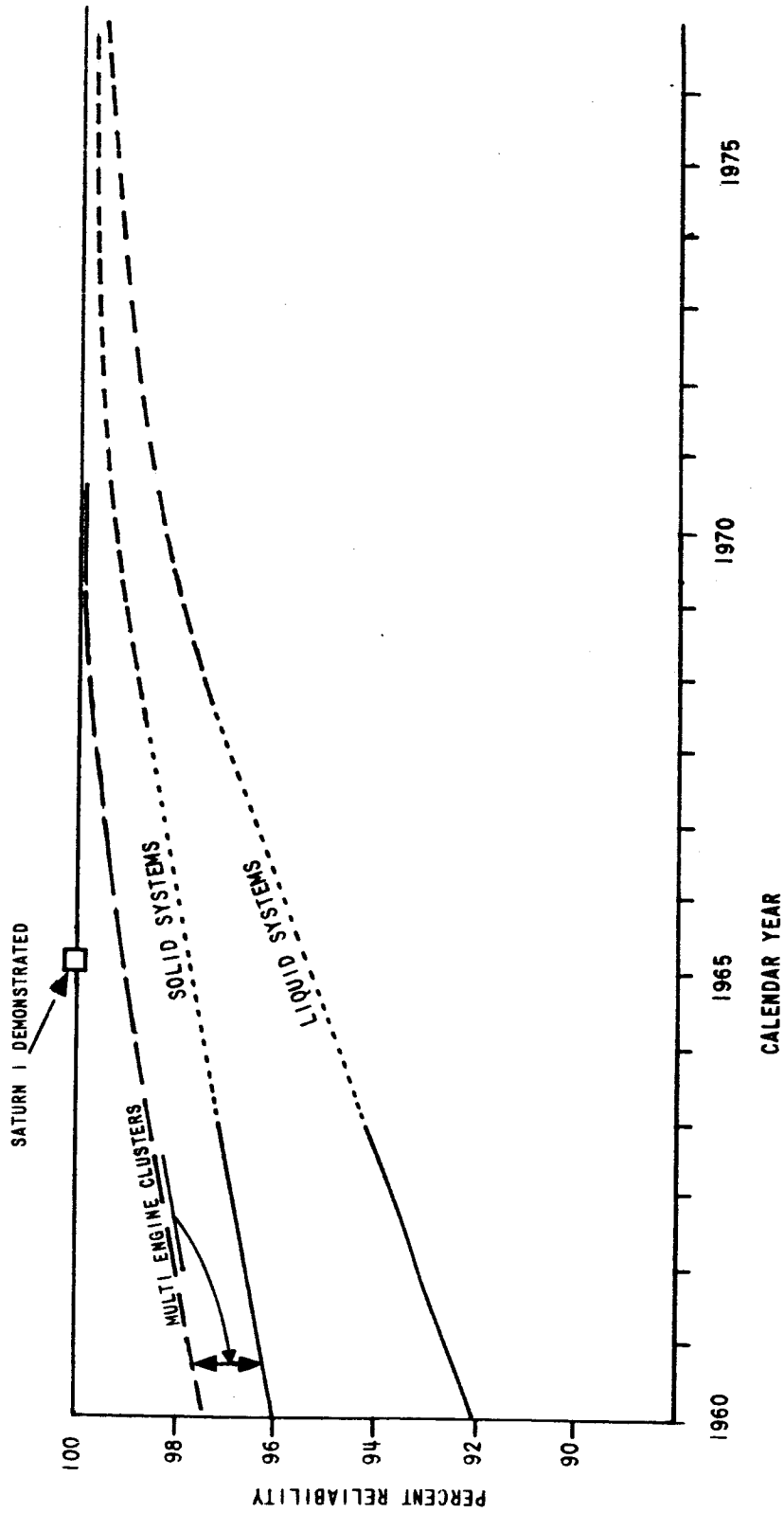


FIGURE 9-20. FORECAST OF CHEMICAL ROCKET RELIABILITY
(Internal Honeywell Source Unidentified)

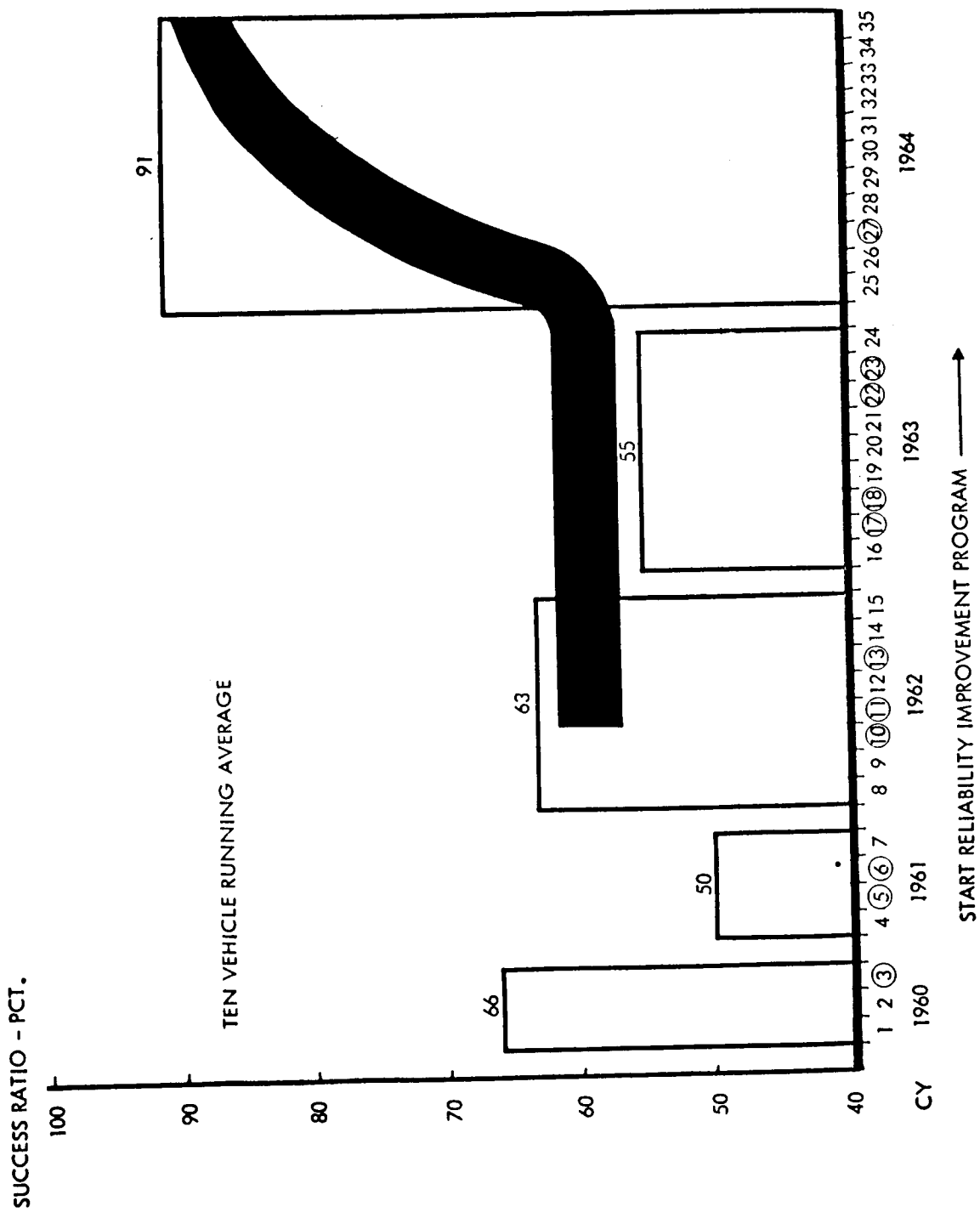


FIGURE 9-21. NASA/DOD SCOUT SUCCESS RATIO BY CALENDAR YEAR

1065-08A

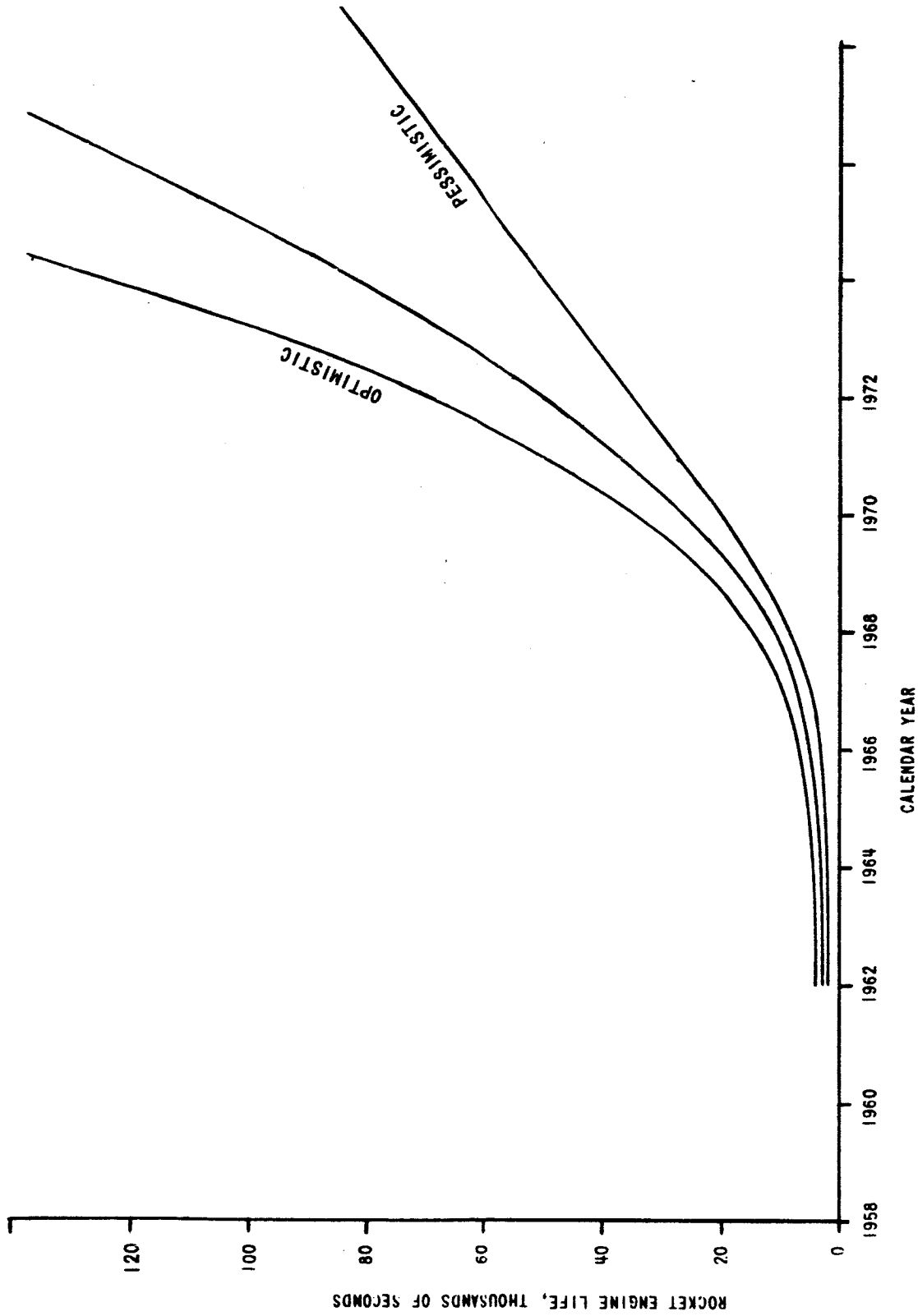


FIGURE 9-22. CHEMICAL ROCKET ENGINE LIFETIME, DEVELOPMENT
(Internal Honeywell Source Unidentified)

9.1 LIQUID ROCKETS

Apollo Engines

Existing liquid rockets of primary interest in this study are those used for Apollo and Centaur. They are summarized in Table 9-4.

9.1.1 Storable Liquid Rockets

There are three main propulsion systems contained within the spacecraft portion of the Apollo vehicle. These are the service module propulsion engine, the lunar excursion module (LEM) descent and LEM ascent engines. There are major differences in the launch vehicle propulsion systems and in the spacecraft propulsion systems. These differences are, of course, related to the particular requirements, mission and environments of the stage. All three of the spacecraft propulsion systems are pressure-fed bi-propellant systems, utilizing storable hypergolic propellants and have redundant control valves. The propellants used are nitrogen tetroxide and a 50/50-percent blend of unsymmetrical dimethyl hydrazine and anhydrous hydrazine.

Cutoff Precision. Trajectory errors can result when the variability of "tail off" thrusting adds more or less ΔV than the guidance computation had predicted. The guidance computer calculates an early cut-off signal based upon an assumed value of "tail off" thrust (due to burning of residual propellant in the combustion chamber, and the fact that a finite time is required for valves and pumps to shut down). Deviation of actual tail off impulse from the assumed impulse becomes a trajectory velocity error. Inertial systems can measure this error and can take it into account during subsequent flight.

Reliability, Long Storage. This refers to the reliability of first firing after long storage in space. As an example of its importance, the ascent engine of the Mars excursion module (MEM) will have been in dormant storage for several months since boost from Earth. Its failure to fire would leave the astronauts marooned on Mars.

TABLE 9-4. APOLLO ENGINES SUMMARY
(Data From Reference 5)

Type	Propellant	Thrust $\left(\frac{\text{lbs}}{\text{Kg}}\right)$	I_{sp} "sec"	Thrust Duration sec	Dry Weight lbs (Kg)	Burnout Weight lbs (Kg)	Exit-To Throat Area Ratio	Propellant Flow Rate $\frac{\text{lb/sec}}{(\text{Kg/sec})}$	Length Overall in (cm)	Exit Dia. in (cm)	Chamber Pressure psia	Contractor	Use
H-1	RP-1/LOX	200,000 (90,000) Sea Level	257	155	1,938 (880)	2,101 (956)	8:1	775 (356)	105 (267)	49 (124)	598	NAA	8 on Saturn I first stage
F-1	RP-1/LOX	1,500,000 (682,000) Sea Level		150	16,900 (7,700)	1,900 (864)	18:1		222 (564)	148 (378)		NAA	5 on Saturn V first stage
RL-10 A-3	LH ₂ /LOX	15,000 (6,820) at Altitude	433	470	295 (134)		40:1		88 (173)	37.8 (96)	300	Pratt Whitney	8 on Saturn IV stage of Saturn I
RL-10 A3-3	LH ₂ /LOX	15,000 (6,820) at Altitude	444	470	300 (137)		57:1		70.6 (179)	38.9 (98.8)	400	Pratt Whitney	8 on Saturn IV stage of Saturn I
J-2	LH ₂ /LOX 1:5	200,000 (90,000) vacuum		500	3,170 (1,440)	3,299 (1,500)	27.5:1		116 (295)	81 (208)	632	NAA	Sat IV/S IVB (1) Sat V/S II (5) Sat V/S IVB (1)
Service Module	Hydrazine UMDH/ N ₂ O ₄ 2.0 mix	21,900 (~10,000)							157.8 (402)	98.4 (250)	100	Aero-Jet	One on Apollo SM
LEM Descent	11 1.6 mix	Variable 1,050 to 10,500 lbs					47:1		85 (216)	60 (152)	110	TRW/STL	Land LEM on moon, 1 engine
LEM Ascent	11 1.6 mix	3,500 (1,590)					45.6:1				120	Bell Aero System	Ascend to rendez- vous in lunar orbit
CM Launch Escape	Polysulfide/ NH ₄ ClO ₄ (solid)	155,000 (~70,000)			4,700 (full) 2,140				180 (457)	28 (68)		Lockheed	Boost abort of CM

TABLE 9-5. HIGH-ENERGY LIQUID PROPELLANTS
(From Ref. 12)

HIGH-ENERGY LIQUID PROPELLANTS					
Oxidizer	Fuel ^a	O/F Ratio ^b	T _c (F)	I _{sp} , Sec ^c	Bulk Sp. Gr
Chlorine trifluoride	Ammonia*	3.93	2619	240	1.43
	Hydrazine*	0.677	5840	251	1.22
Fluorine	Ammonia*	2.90	7512	295	1.017
	Hydrazine*	1.98	7692	300	1.070
	Hydrogen*	9.42	8072	371	0.457
		3.77	4469	356	0.239
	Lithium	1.83	8319	309	0.785
	Methyl alcohol*	2.37	7472	298	0.991
Flox	JP-4	3.78	7370	278	1.21
Nitrogen tetroxide	Hydrazine*	1.00	4905	249	1.193
	Hydrogen	11.5	5610	279	0.565
	Propane	4.15	5121	240	1.128
Oxygen	Acetylene	1.23	6012	266	0.828
	Borohydride	1.32	6000	276	0.634
	Ammonia	1.25	4834	250	0.969
	Ethane	2.30	5308	254	0.867
	Ethyl alcohol	1.50	5297	242	0.970
	Ethylene	1.86	5538	264	0.773
	Gasoline	2.26	5660	252	0.955
	Hydrazine	0.83	5382	263	1.065
	Hydrogen	2.89	3886	345	0.23
	Kerosene	2.28	5702	249	1.02
	Lithium*	1.15	13,000	318	0.747
	Li borohydride*	1.47	8300	306	0.846
	Lithium hydride*	1.34	6400	268	0.975
	Methane	2.33	4874	263	0.747
	Methylamine	2.06	5600	252	1.07
	n-Octane	2.20	5498	248	0.975
Oxygen bifluoride	Hydrazine	1.33	6290	298, S.E.	1.28
	Ammonia	2.13	6065	292, S.E.	1.12
WFNA (white fuming nitric acid)	Hydrazine*	1.22	4681	246, S.E.	1.23
	Hydrogen	12.6	5360	298	0.60

*Asterisk denotes that propellant combination is hypergolic. ^bMixture ratio given is generally optimum. ^cSpecific impulse figures given are based on a stable propellant composition.

Specific Impulse. Nitrogen tetroxide and UDMH propellant has an ISP theoretical of 340 (313 achieved). Compared with OF_2/H_2 of 473 theoretical (433 achieved). For long term storage in space, however, the advantage of the cryogenic propellants is offset by the boiloff requirement (evaporate some to keep the rest cold). The hypergolic (self-igniting) feature of $\text{N}_2\text{O}_4/\text{UDMH}$ (hydrazine) gives better reliability of startup (see subparagraph 9.1.1).

Reliability, Restart. This refers to probability of successful restarts after shutdown in space. Both hypergolic and cryogenic liquid rockets have been successfully restarted in space. Repeated thermal cycling of critical hardware elements (due to firing and cooldown) perhaps can reduce the probability of each successive restart. Feasibility of restart is well established, but numerical data on reliability has not yet been found in the literature.

9.1.2 Hypergolic Storable Liquid Rocket

This is really the same category as subparagraph 9.1.1.

Specific Impulse. See paragraph 9.1.1.

Restart Reliability. See paragraph 9.1.1.

Throttleability. Both the LEM descent engines and the RL-10 engines have a throttle ratio of 10:1 (thrust variation of 10 percent to 100 percent). Finer throttleability would pose a difficult design problem. The lower thrust is obtained at a lower efficiency.

Throttling requirement for the LEM descent engine was probably the single most stringent requirement for the engine. To assure high performance and stable combustion of all thrust settings, it was necessary to maintain uniform injector patterns over the entire range of propellant flow rates. Initially, two parallel approaches were undertaken to insure the success of the program. One approach employed the injection of an inert gas (helium) into the propellant manifolds to sustain propellant injection velocities. A second method uses a mechanically throttled engine which has a variable area injector. Both approaches were tested and evaluated, and the mechanically throttled (10:1) engine selected for development.

Impulse Accuracy. Variation of total impulse (see Figure 9-23). The rated thrust of an engine is only a nominal value. Actual thrust depends upon many environmental and design factors, and variations of ± 10 percent

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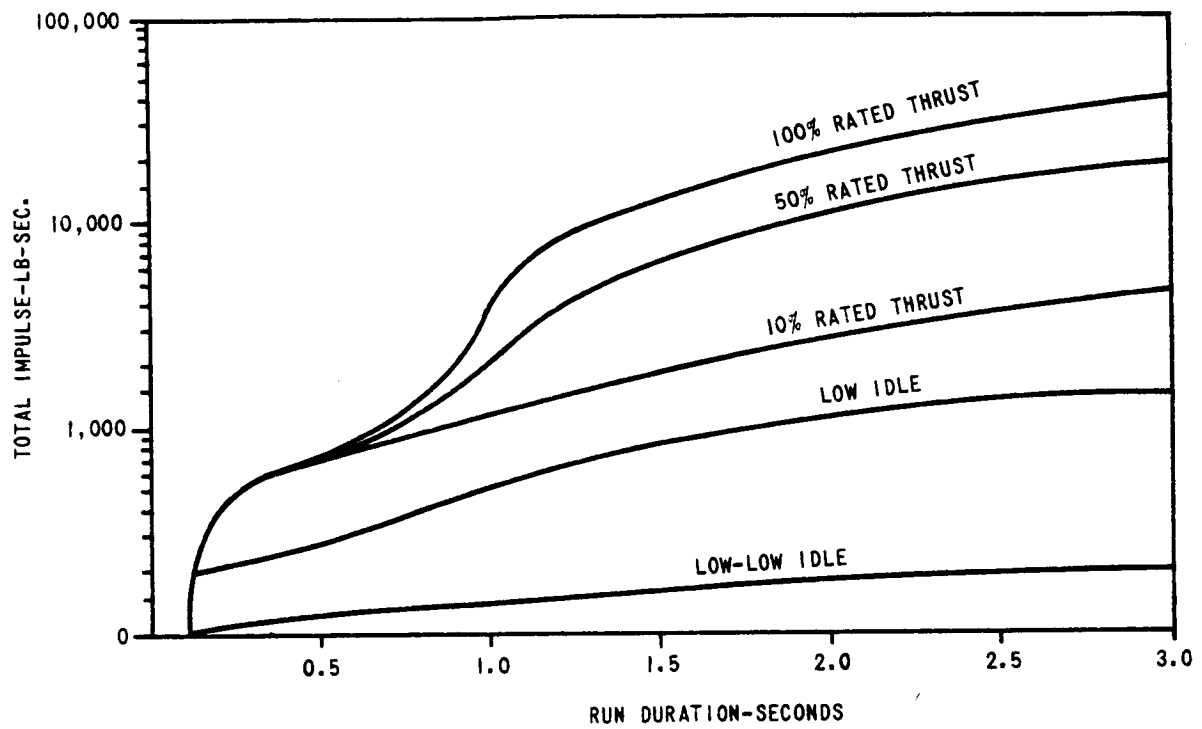


FIGURE 9-23. RL10 ENGINE ESTIMATED TOTAL IMPULSE
(Source: Reference 5)

can exist. In combination with the cutoff precision problem (paragraph 9.1.1) this variability makes it generally impractical to meter ΔV magnitude by simple open loop timing of thrust. Accelerometer control of ΔV cutoff is preferable.

Reliability. Refers to the overall reliability of the rocket engine as a consideration of MTBFs of individual, functional hardware elements.

Reliability, Long Storage. See paragraph 9.1.1.

9.1.3 Cryogenic Liquid Rocket

Rockets which use liquified propellants whose boiling temperatures are a few degrees above absolute zero ("cryogenic" temperatures) are called cryogenic rockets. The liquid oxygen (LOX)/liquid hydrogen (LH₂) rocket is of primary interest. Typical members of this family are the RL-10 engine on Saturn I (S IV stage) and Centaur, the J2 engine on Saturn S IB (S IVB), Saturn V (S II), and Saturn V (S IVB).

Figure 9-24 shows the freezing to boiling range of several liquid propellants (see also Figure 9-25).

Cryogenic Storage. The amount of propellant lost due to boiloff from vented tanks is mission dependent as well as time dependent. A general idea can be gained from the specific mission analysis results shown in Figure 9-26, where the remaining ΔV capability is plotted versus coasting time for several propellant combinations. Note the slope of the O₂/H₂ curve showing a 25 percent loss in six months. (See also paragraph 9.1.4.)

Specific Impulse. The primary reason for using cryogenic propellants is their high specific impulse. Flox (30:70 F₂/O₂)/hydrogen is one of the best (443 to 473).

However, all chemical rockets are severely limited to specific impulses less than this. Higher Isp rockets are possible (nuclear, electric, pulse, arc jet, MHD, etc.)

Reliability, Restart. (See paragraph 9.1.1.) If water or other high boiling contaminants exist in cryogenic propellants, icing of valves in space can reduce probability of successful restart.

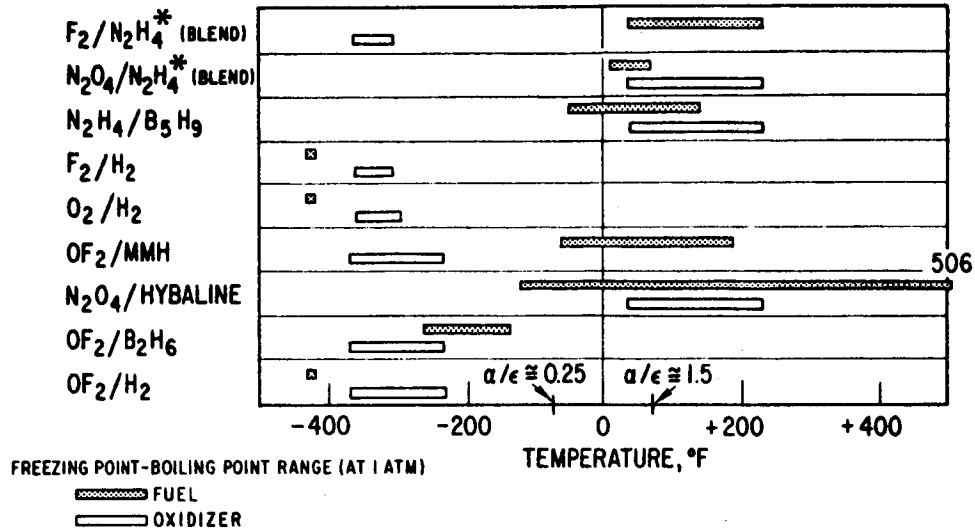


FIGURE 9-24. PROPELLANT STORAGE IN SPACE
 PROPELLANT TEMPERATURE OVERLAP
 (Source: Reference 11)

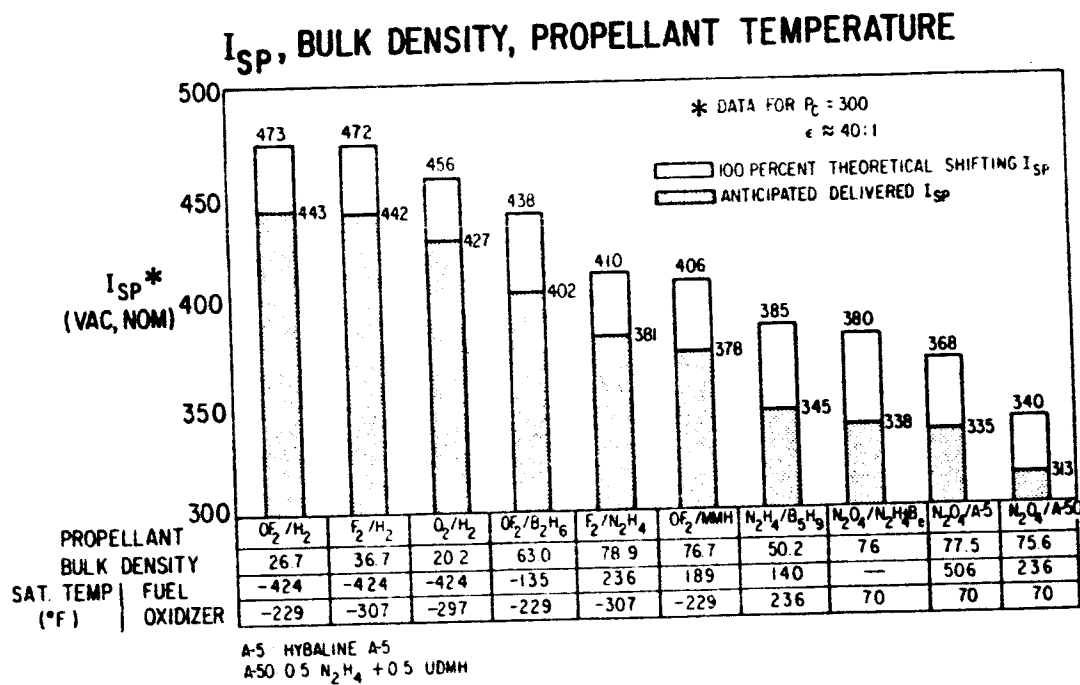


FIGURE 9-25. PROPELLANT CHARACTERISTICS
(Source: Reference 11)

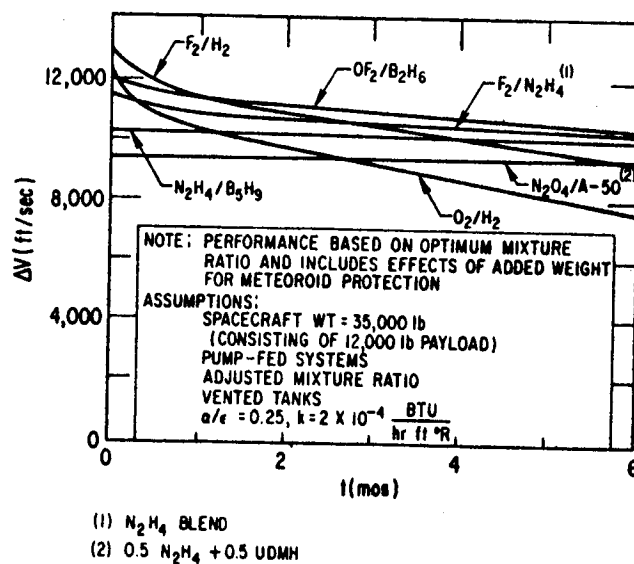


FIGURE 9-26. SPACE MANEUVERING SYSTEMS
 VELOCITY INCREMENT VS COAST TIME (mos)
 (Source: Reference 11)

Throttleability. The RL-10, LOX/LH₂ engine has a ten-to-one throttle ratio. To provide this on other engines presents a redesign problem, with additional controls. Total impulse of the RL-10 at various thrust levels is estimated in Figure 9-23.

Reliability, Long Storage. (See paragraph 9.1.1). Additives to the LOX can permit cryogenic rockets to start hypergolically. Extreme cold of propellant, applied over long periods may adversely affect materials and reduce probability of successful start.

Cutoff Precision. See paragraph 9.1.1.

9.1.4 High Energy Upper Stage

Table 9-6 lists four of the previous "high energy upper stage" programs. Table 9-7 shows the results of the optimization study of Reference 11. Table 9-2 lists some high energy missions. Table 9-5 lists some high energy propellant combinations.

Two high energy stages presently are being studied, a 7 k "kick stage" and a 40 k "optimized upper stage". Figure 9-15 shows how the addition of a 7 k stage can upgrade the mission capability of various boosters. The color insert following page 9-17 shows how the 40 k plus 7 k stages can upgrade standard boosters.

R&D Requirement. The following is quoted from Reference 12.

"To meet the high ΔV requirement for advanced missions, the broad alternatives are two: either to concentrate on uprating the existing stages by such measures as flexing (using a roughly 30:70 mix of fluorine and oxygen as the oxidizer) and strap-on rockets, or to develop a completely new final or kick stage, a vehicle optimized for its job of applying the necessary "kick" after the regular boosters have done their job.

A typical precursor of the kick stage is the Transtage used on the Titan 3C. This vehicle fires briefly after having been placed in, say, a 300-mile orbit, coasts until it reaches the desired altitude, and then fires again to circularize the new orbit at the higher altitude. In this manner, it might, for instance, inject a synchronous comsat.

Flying a kick stage on top of the various vehicles in our already-committed stable of space boosters-- such as Centaur-mated versions of Atlas, Titan 2, Titan 3C and Saturn 1B-- can handle the majority of the contemplated high- ΔV missions to Venus, Mars, asteroid belts, comets, and so forth (see

TABLE 9-6. HIGH ENERGY UPPER STAGE PROGRAMS
(Source: Ref. 11)

	NOMAD	HYDRA	CHARIOT	CENTAUR**
CONTRACTOR	ROCKETDYNE	AEROJET	BELL	P & W/GDA
DATE INITIATED	≈MAR 1958	APRIL 1959	≈NOV 1960	OCT 1958
PROPELLANTS	$F_2/N_2H_4^{(1)}$	O_2/H_2	$F_2/4N_2H_4-H_2O-MMH$	O_2/H_2
FEED SYSTEM	PRESSURE	PRESSURE	PUMP	PUMP
THRUST (VAC,NOM) lb	12,000	15,000	35,000	30,000
MR	1.6	5.0	2.24	5.0
ϵ	20	20	45	40
P_c (PSIA)	150	80	350	300
I_{sp} (VAC,NOM) sec	357	410	390	433
STAGE WEIGHT	12,656*	12,456	34,000	32,531
BURNOUT WEIGHT	1,324*	2,164	2,184	4,002

* DOES NOT INCLUDE GUIDANCE

** TWIN CHAMBERS

(1) N_2H_4 BLEND

TABLE 9-7. HIGH ENERGY UPPER STAGE
OPTIMIZATION STUDY
(Source: Reference 11)

STAGE SIZE AND THRUST LEVEL IN THOUSANDS OF lb

PROPELLANT	BEST UPPER STAGE FOR INDIVIDUAL BOOSTER						STANDARD	
	TITAN III A		TITAN III C		SATURN- I B		UPPER STAGE	
	STAGE WT	THRUST	STAGE WT	THRUST	STAGE WT	THRUST	STAGE WT	THRUST
F ₂ /H ₂	26	20	53	40	37	20	40	35
F ₂ /N ₂ H ₄ (BLEND)	28	20	41	25	34	20	37	25
O ₂ /H ₂	24	20	41	23	36	20	32	23
N ₂ H ₄ /B ₅ H ₉	22	20	38	20	30	20	30	20
N ₂ O ₄ /A-50 ⁽¹⁾	21	20	34	20	32	20	28	20
OF ₂ /H ₂	27	20	49	40	37	20	39	35
OF ₂ /MMH	28	20	41	25	36	20	37	25
OF ₂ /B ₂ H ₆	26	20	41	25	36	20	32	20
N ₂ O ₄ /0.8N ₂ H ₄ +0.2B ₆	18	20	37	20	31	20	26	20
H ₂ O ₄ /HYBALINE (A5)	23	20	36	20	32	20	29	20

(1) 0.5 N₂H₄ + 0.5 UDMH

Figure 9-15). Moreover, these missions would not be the relatively simple, experimental payloads that we've been lifting but truly sophisticated ones, weighing probably 1,350 pounds at a minimum (in comparison, Mariner 4 weighed in at roughly 575 pounds)." (End of quote from Reference 12.)

Bell Aerosystem's 40 k thrust engine has been fired repeatedly with flourine/hydrogen at over 95 percent combustion efficiency. However, at the present time, development of the 40 k upper stage is somewhat in doubt.

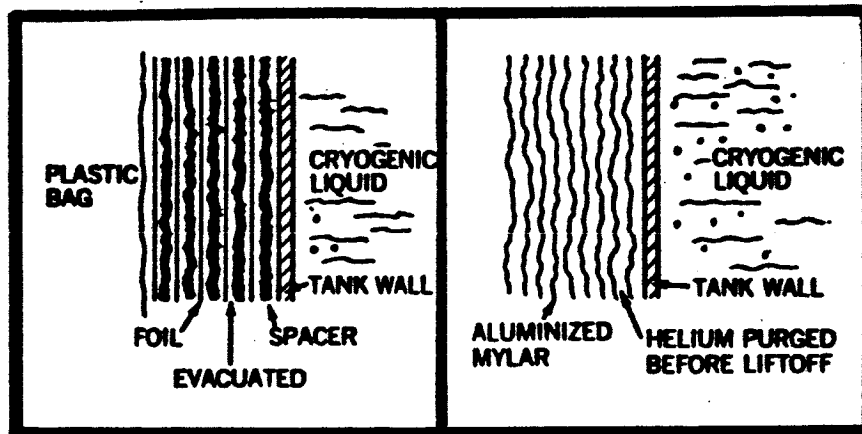
R&D is required in propellant selection, engine design, throttleability, storage and restart combustion stability, etc. for whatever advanced kick stages are funded.

Cryogenic Storage. The following is quoted from Reference 12. See Figure 9-27.

Superinsulation (spaced foil laminates that act like a thermos bottle in space) works especially well on the surface of pressure-stabilized tankage that is enclosed in a load-bearing structure. Problems arise when this necessarily lightweight material must provide external protection during the atmospheric phase of the launch. Also, there is the problem of wrapping this fragile layering around the multicurved and inset surfaces of the stage. The same problem is met by the alternative layer-insulation technique of using pairs of crinkled aluminum sheets to give less contact, and thus less thermal conduction between insulation surfaces and tank wall.

Any comparative listing of available high-energy chemical bipropellants shows the most potent to be the liquid cryogenics, in particular the fluorine/hydrogen combinations with a theoretical maximum ISP of 477 seconds. And close on their heels are the oxygen/hydrogen mixtures with ISPs of around 458 seconds. Of the more complex tripropellants, the lithium/fluorine/hydrogen combination is the most potent (theoretical ISP = 531 seconds).

The problem with cryogenic propellants comes when you want to have them remain intact and quiescent during extended coast periods on longer missions. The real culprit in any O₂/H₂ or F₂/H₂ combination is the hydrogen. Aside from its notoriously low density (which means it requires larger tanks and thus more tank surface to insulate), liquid hydrogen is constantly evaporating, and this gas must be continuously vented overboard-- or recondensed by a liquefier-- during the mission to prevent over-pressuring the tanks. Then,



Two alternative approaches for foil laminate insulation on cryogenic stage tankage. Thermos bottle concept at left is perhaps the more common; it uses two foil laminates, separated by thin sheets of glass fiber paper or thin cloth, covered by a plastic bag and evacuated. Drawbacks include the difficulty in applying it to a tank that has many discontinuous surfaces and extruding supports, and the rarity of effective sealants that can tolerate cryogenic temperatures (e.g., the -239°C of liquid hydrogen). The approach shown at right uses the natural separation of layers afforded by tightly crinkled aluminized Mylar, but this material, too, is difficult to apply to a complex tank surface.

FIGURE 9-27. TANK INSULATION
(Source: Ref. 12)

too, insulating it from the lox or liquid fluorine (which are stored at temperatures some 110°F higher than hydrogen) is a very tough proposition even on Earth, to say nothing of space soak effects.

Despite these finicky properties, oxygen/hydrogen is proving to be quite a respectable high-energy propellant (it is used repeatedly in the operational Saturn upper stages and in the Centaur, all powered by Pratt & Whitney RL-10 engines), vindicating NASA's earlier disputed decision to use O₂/H₂ in upper stages. Fluorine/hydrogen also shows promise of fulfilling kick-stage propellant requirements, having been fired repeatedly with over 95 percent combustion efficiency in Bell Aerosystem's 40,000-pound-thrust engine and in modified 15,000-pound-thrust RL-10A3s.

Corrosive Fuel Handling and Storage. Fluorine, N₂O₄, nitric acid and other high energy oxidizers tend to attack whatever reactive material they contact. Nevertheless, means have been developed for storing, pumping, and handling all of them reliably. Proper safety procedures and equipment have been developed also. According to Dr. Lucas at MSFC, there are difficulties but no insurmountable problems. Surprisingly enough, fluorine can be handled in anodized aluminum containers.

Restart Reliability. See paragraph 9.1.1.

Specific Impulse. The following is quoted from Reference 12.

"Moderate vs Deep Cryogenics

As an alternative to the deep cryogenics, NASA is also sponsoring study of the less energetic (up to 380-second Isp) moderate cryogenics, such as fluorine or flox plus hydrocarbons (light petroleum gases such as methane and propane). The advantage of this class of propellants derives from the common boiling temperature range of both the fuel and oxidizer, greatly easing the space storability problem. Other promising moderate cryogenics under study include OF₂/diborane (which Reaction Motors, in particular, has been working on) and OF₂/monomethyl hydrazine. At present, emphasis is on getting the specific impulse above 360 seconds.

Metallized propellants also are candidates for the high-energy kick stage, offering even higher theoretical Isp than the F₂/H₂ combinations. However, at today's state of the art, their combustion properties are even trickier than the latter's, so it's impossible to predict what the operational Isp is going to be when you start up the engine. For instance, the promising approach of burning 52 percent beryllium and 47 percent hydrogen with

oxygen (or, similarly, the lithium/hydrogen/oxygen combination mentioned earlier) has been hard-pressed even to reach 400 seconds, not to mention the theoretical ISP of 536 seconds.

While the pursuit of alternatives to the liquid cryogenic bipropellants is a promising one, the latter are far and away the strongest contenders for the kick stage." (End quote) (See also paragraph 9.1.3.)

9.2 SOLID ROCKETS

Quoting Reference 1: "The steady growth in solid propellant rocket size, thrust, payload efficiency and burning times during the past ten years is expected to continue for the next decade. The most significant development in the last several years is the development of castable, case-bonded, internal burning propellant grains. It eliminated the need for large extrusion presses for forming of propellant grains and has also made it possible to reduce inert component weight.

Improvement in construction materials is a key element in the advancement of solid rocket performance as it is in most other areas of aerospace technology. Another vital requirement is better cooling techniques. Unless both cooling and materials advancements are achieved it will not be possible for the case and aft-closure-nozzle combination to withstand the higher operating temperatures and pressures required to produce an overall performance gain in these engines. Future requirements include more stringent directional control of the thrust vector, closer control of thrust magnitude, thrust termination and center of gravity location as the propellant burns."

Quoting Reference 12: "Bulk density of solid propellants is, of course, unexcelled (which is largely why hybrids evolved), and propellant storability, variable thrust, restartability, instant cutoff and many other requirements are apparently being met. Where the solids fall down in competing with the other types of chemical engines is in providing both throttling and stop-start operation.

Stop-start solids of fixed impulse increments have, of course, become state of the art hardware; there would seem to be no limit to the number of stop-start cycles obtainable with the multiple-wafer approach. With the solid-core propellant engines, water quenching has also shown stop-start capability, but the quantity of liquid that must be carried along for this purpose is no help.

Perhaps the greatest promise for giving solid motors the same stop-start and throttling flexibility as the liquids is shown by annular plug techniques combined with advanced nozzle geometry and ignition techniques. Currently satisfactory engine thrust cutoff is also being achieved by means of blowout ports and thrust vector control techniques, particularly secondary-liquid and hot-gas injection and gimballed nozzles. Clearly, the time when solids will be able to be controlled instantly and at will is not far off."

An outline of the interrelationship between the basic missions, design goals, and technical features of solid rockets is shown in Table 9-8.

In addition to the 120 inch million pound thrust strap-on "stage zero" engine for Titan III, and the Minuteman solid rocket, a 156 inch solid motor with 3.2 megapound thrust has been static test fired. The 260 inch solid has been test fired in its half length (3.7 megapound) 260-SL-1 configuration, which is 80 feet 8 inches long. The full length (162 feet), 260 inch motor will deliver 7.5 million pounds thrust for 120 seconds; however, its future funding is in doubt.

Small solid rockets for reliable deboost from Earth orbit are state of the art.

Various sizes and staging combinations of small solid rockets for research are available commercially as catalog items.

Figures 9-28 through 9-35 (from Ref. 1) show parametric forecasts of solid rockets.

One of the principle applications for solid rockets is in thrust augmentation of standard boosters. As an example, Figure 9-36 shows the increase in payload with various solid strap-ons to the Saturn V. Figure 9-37 shows uprating possibilities for the Saturn 1B using a 260 inch strap-on.

To illustrate further applications of the 260 inch motors, Figures 9-38 and 9-39 show a hypothesized solid-boosted core stage to place 1.2 to 2.7 million pounds into a low Earth orbit.

9.2.1 Low Cost

Table 9-9 summarizes a cost-to-pounds-in-orbit-effectiveness-rating study for various Saturn configurations and the large hypothesized core vehicle.

TABLE 9-8. SOLID ROCKET FEATURES
(Source: Ref. 1)

Major Design Aim	Missions	Technical Features		
		Propellant Types	Materials	Design
Cost	Sounding, Tactical Ordnance, Boosters	Double Base Ammonium Perchlorate Ammonium Nitrate	Steel and some refractories	Conventional
Performance	Space, Strategic, Defensive	Ammonium Perchlorate with high energy additives	Special alloys, Titanium Reinforced Plastics	Sophisticated
Reliability	Man-Rated Defensive Strategic	Ammonium Nitrate, Ammonium Perchlorate plus additives	Special alloys, Titanium	Conservative
Availability	Defensive, Space, Research	Ammonium Nitrate Double Base, Ammonium Perchlorate	Steel Special alloys	Conventional

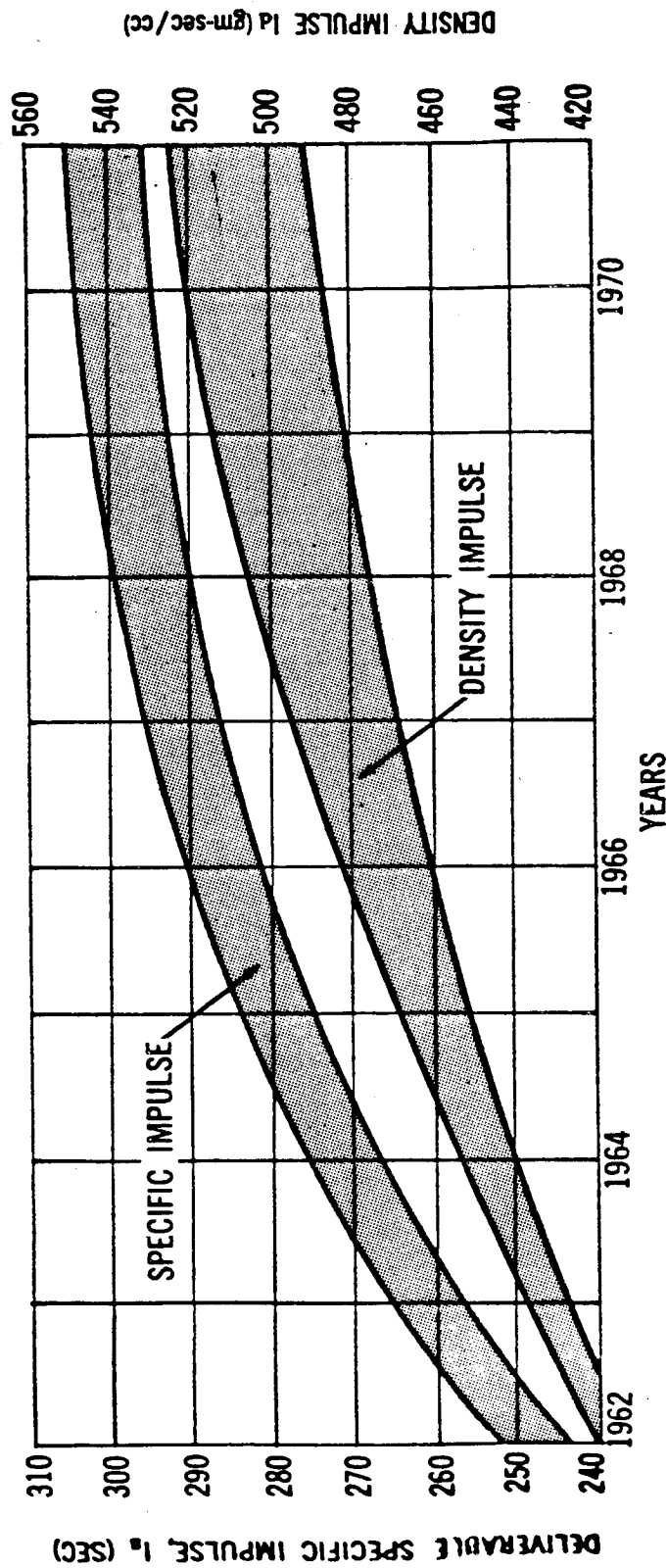


FIGURE 9-28. SOLID MOTOR IMPULSE
(Source: Ref. 1)

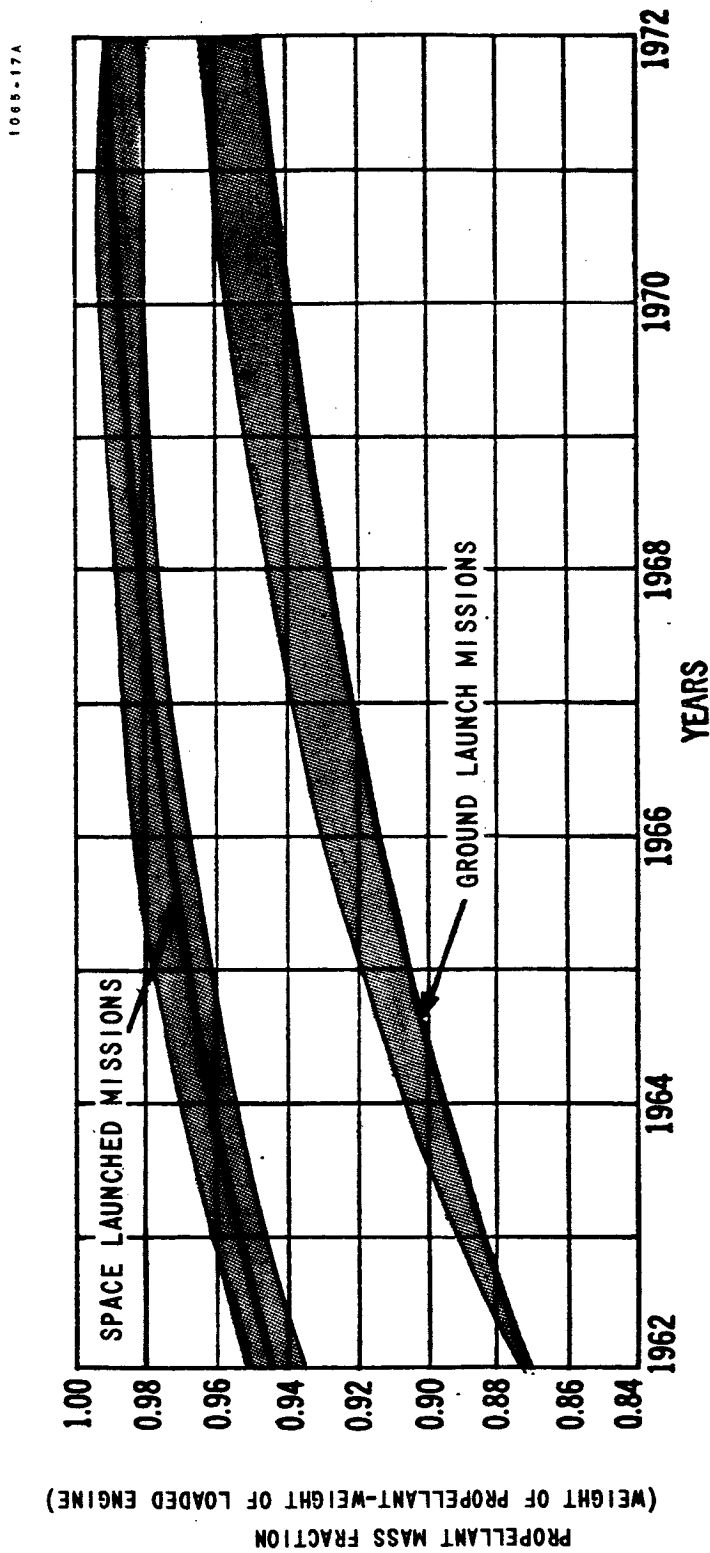


FIGURE 9-29. SOLID ROCKET MASS FRACTION
(Source: Ref. 1)

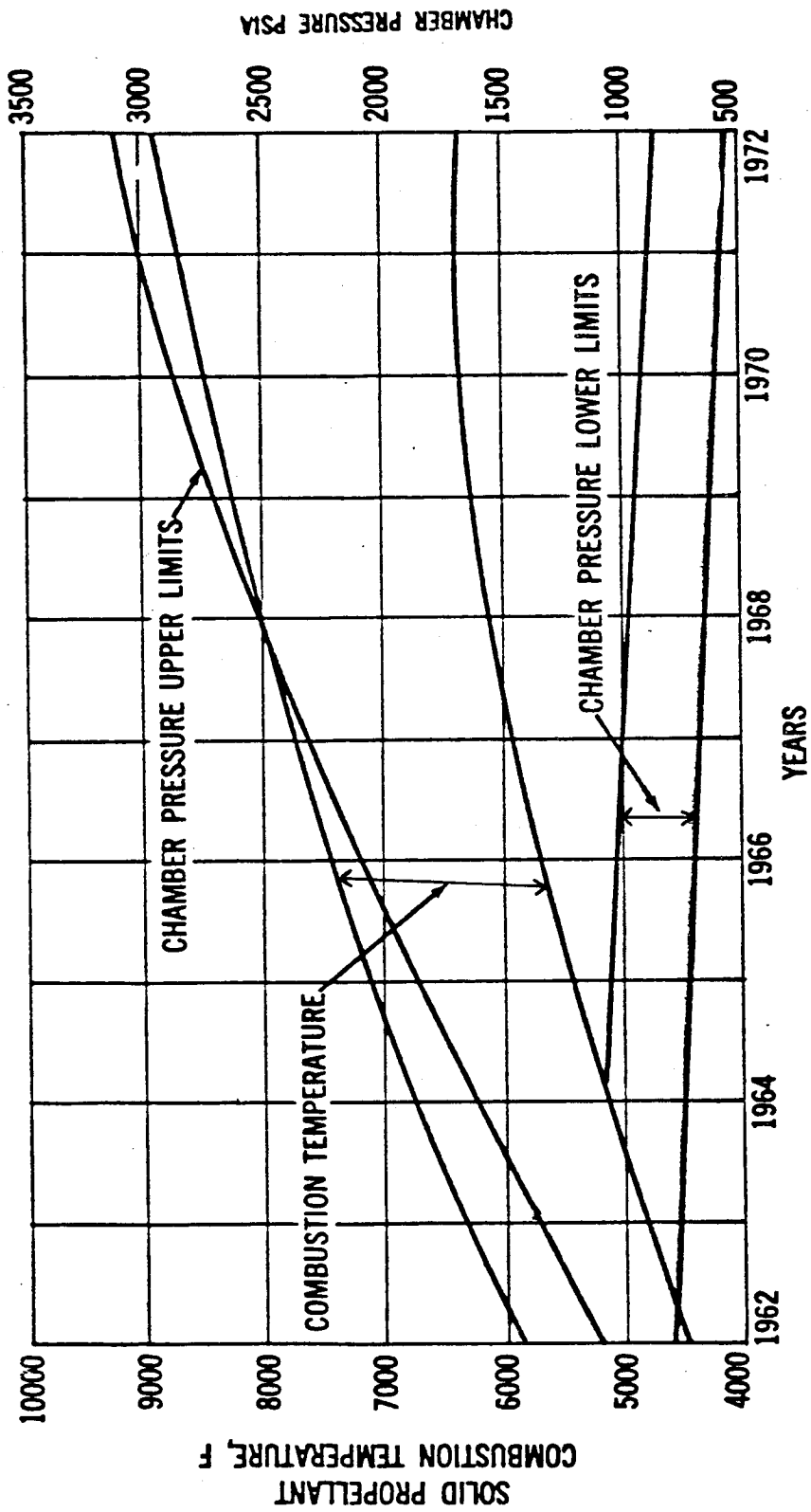


FIGURE 9-30. SOLID ROCKET PRESSURE AND TEMPERATURE
(Source: Ref. 1)

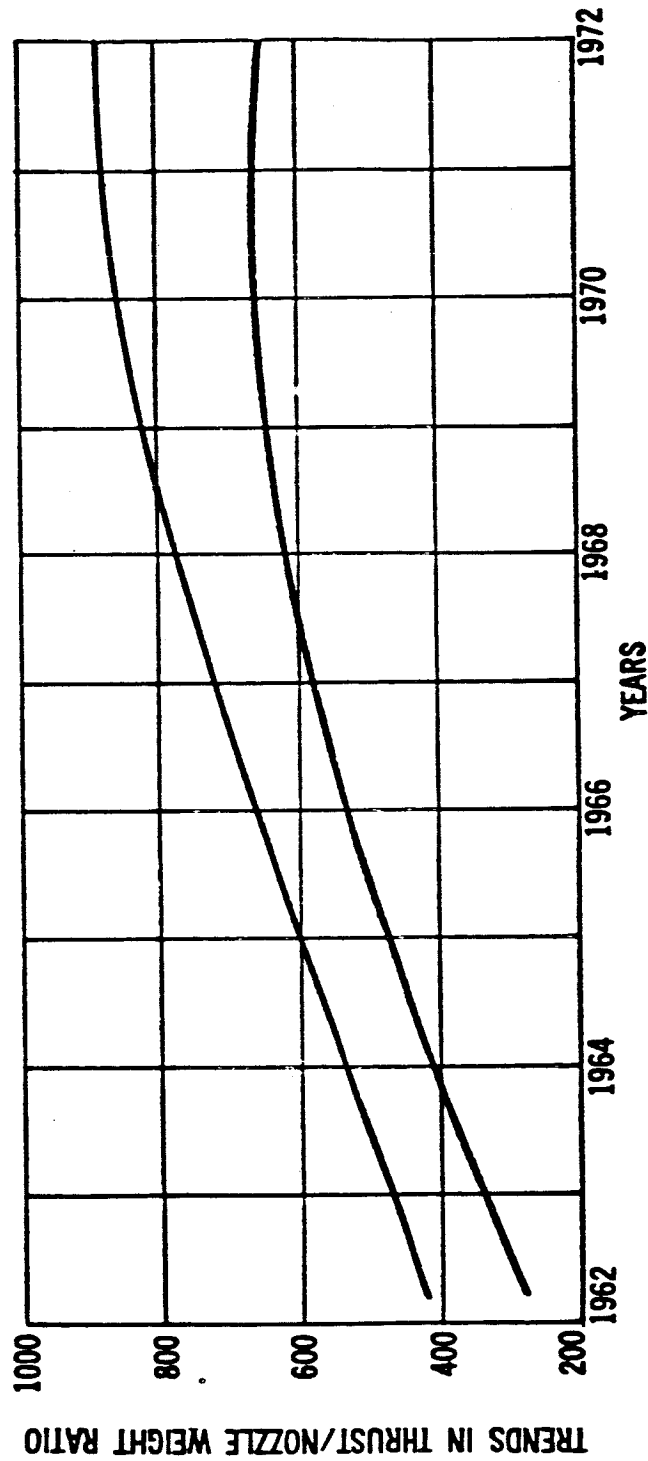


FIGURE 9-31. SOLID ROCKET THRUST/NOZZLE WEIGHT RATIO
(Source: Ref. 1)

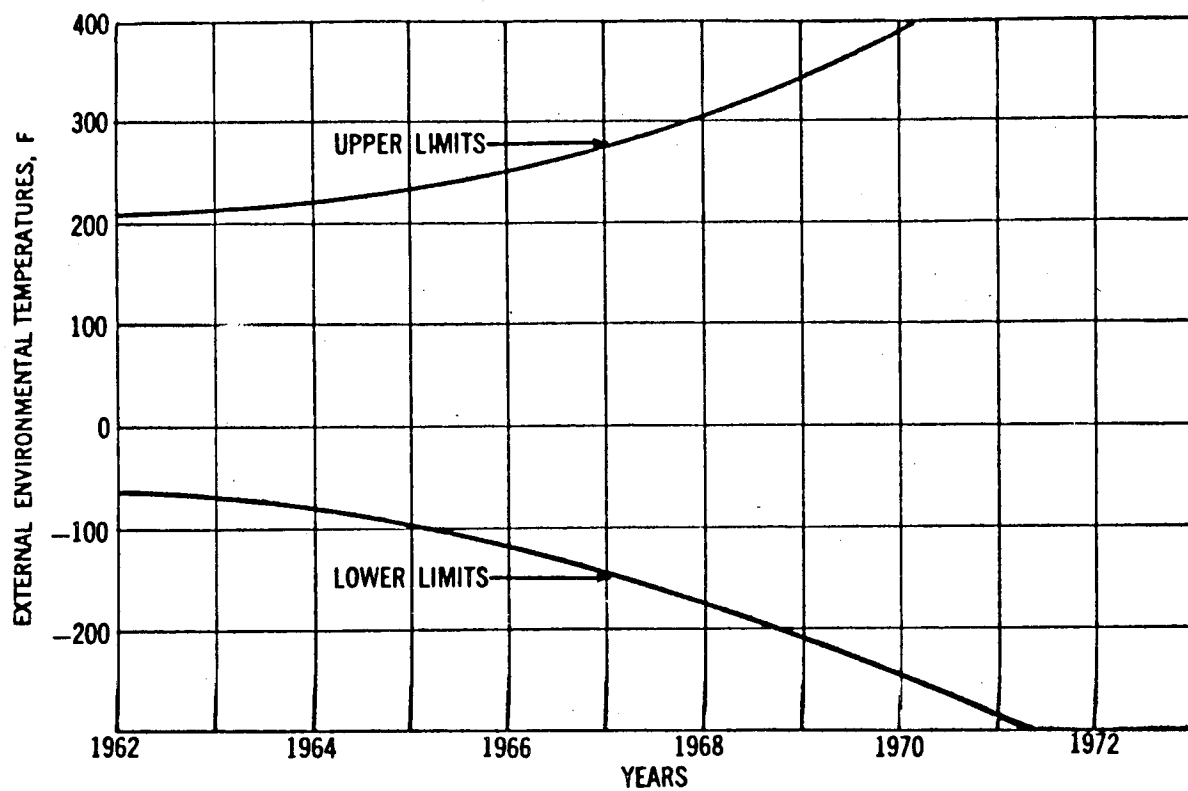


FIGURE 9-32. ENVIRONMENTAL TEMPERATURE RANGE -
SOLID ROCKETS
(Source: Ref. 1)

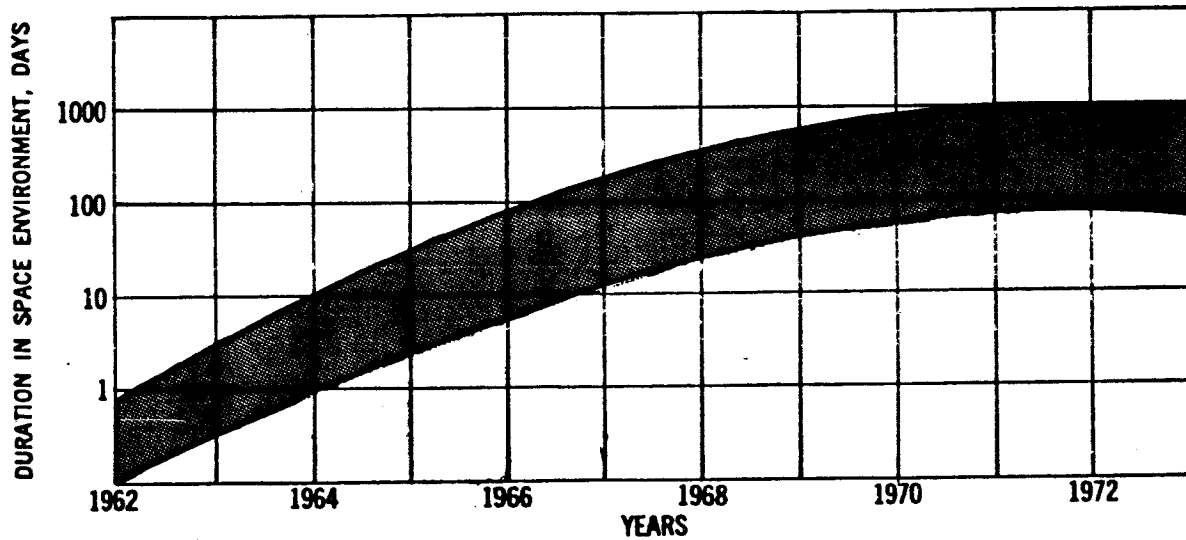


FIGURE 9-33. SOLID ROCKET DURATION IN SPACE ENVIRONMENT
(Source: Ref. 1)

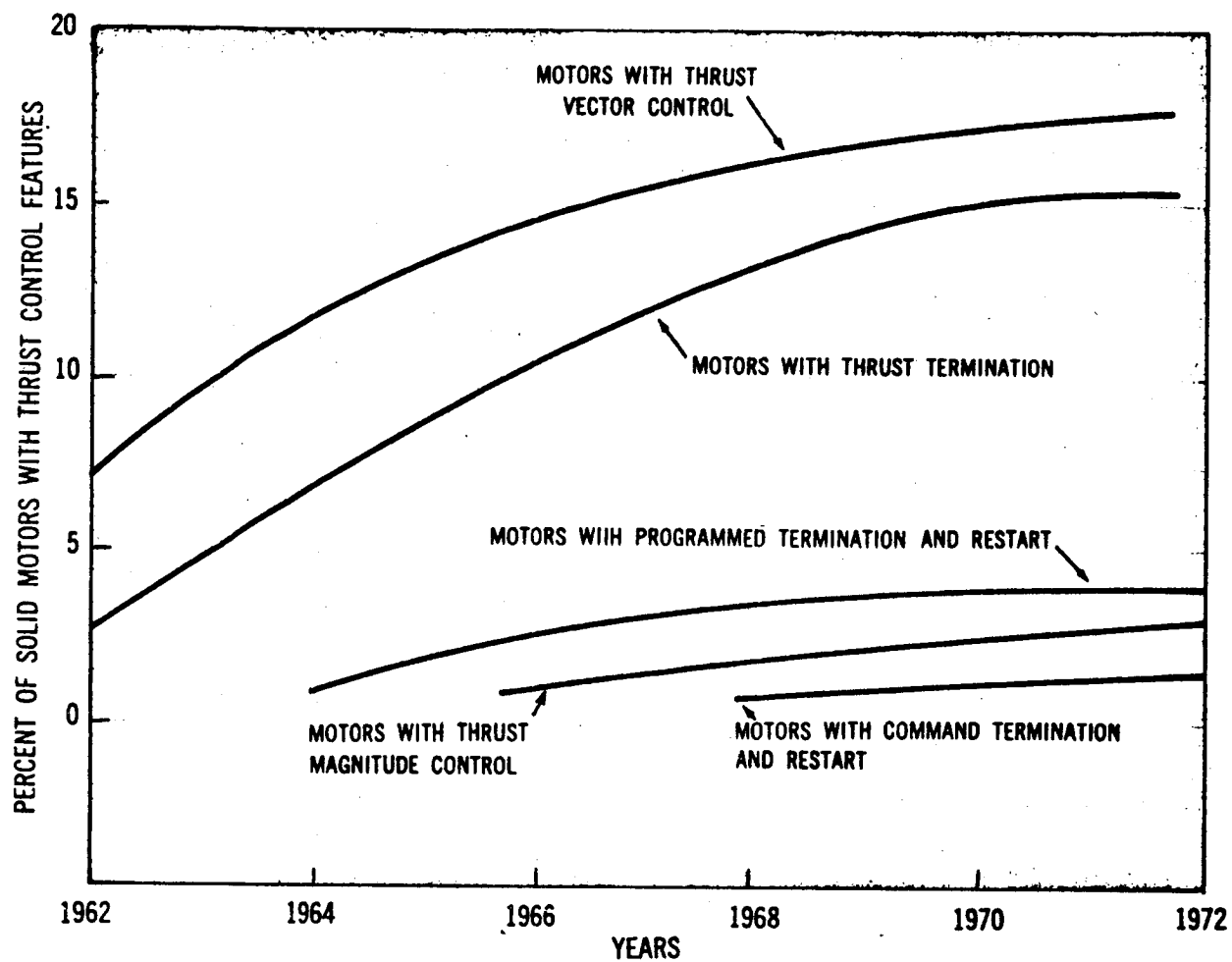


FIGURE 9-34. SOLID MOTOR CONTROL FORECAST
(Source: Ref. 1)

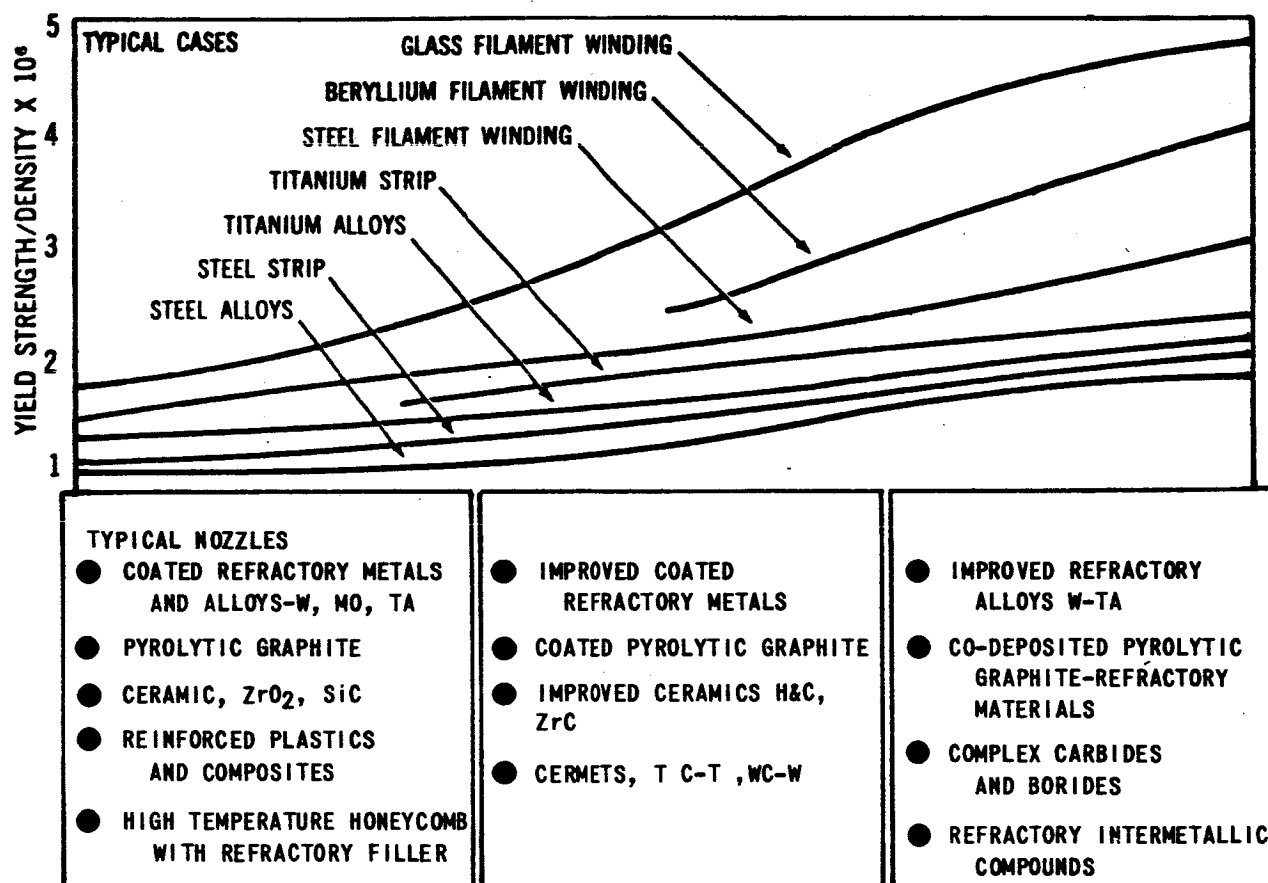


FIGURE 9-35. NOZZLE AND CASE MATERIALS
(Source: Ref. 1)

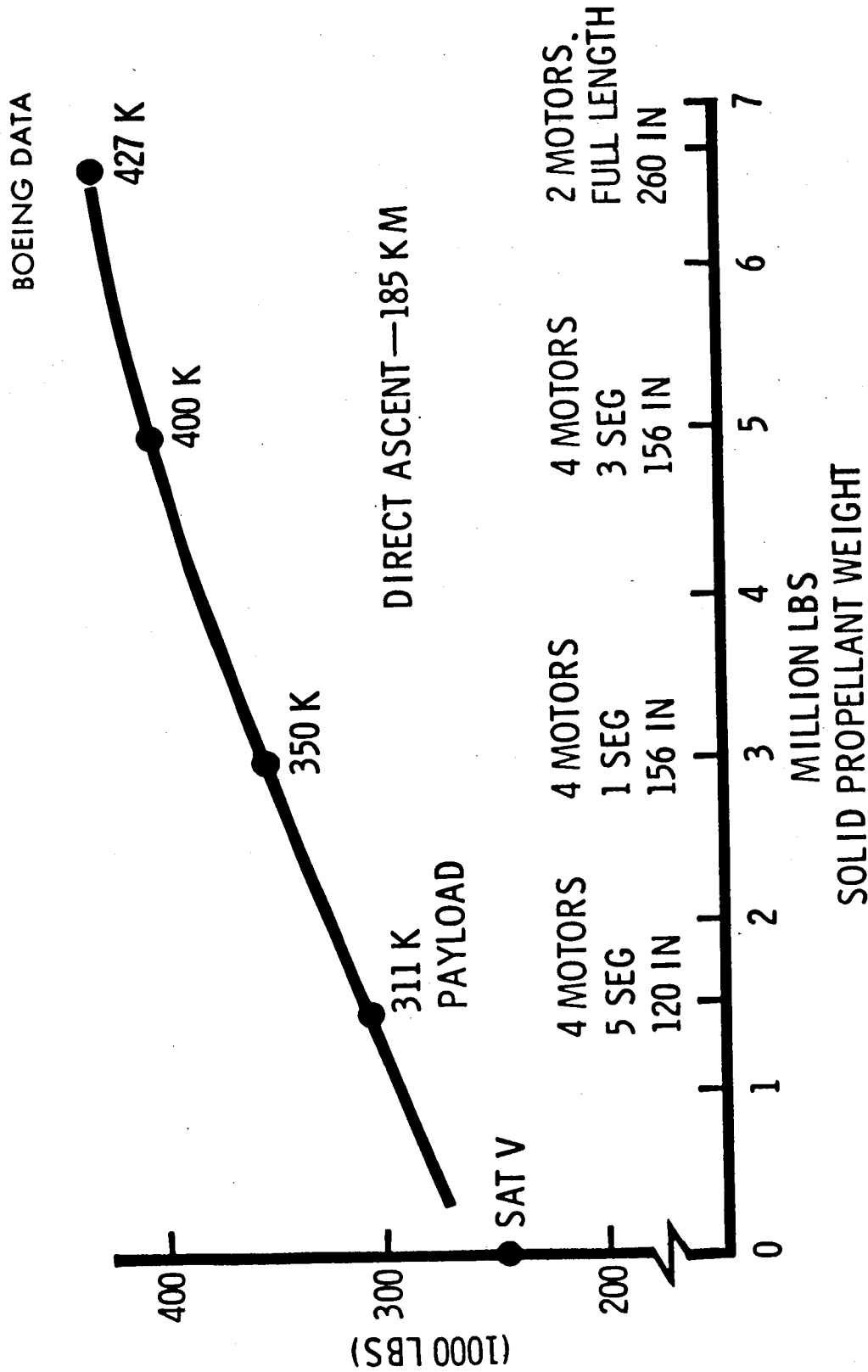


FIGURE 9-36. PAYLOAD INCREASE - SATURN V WITH SOLID BOOST ASSIST
(Source: Ref. 4)

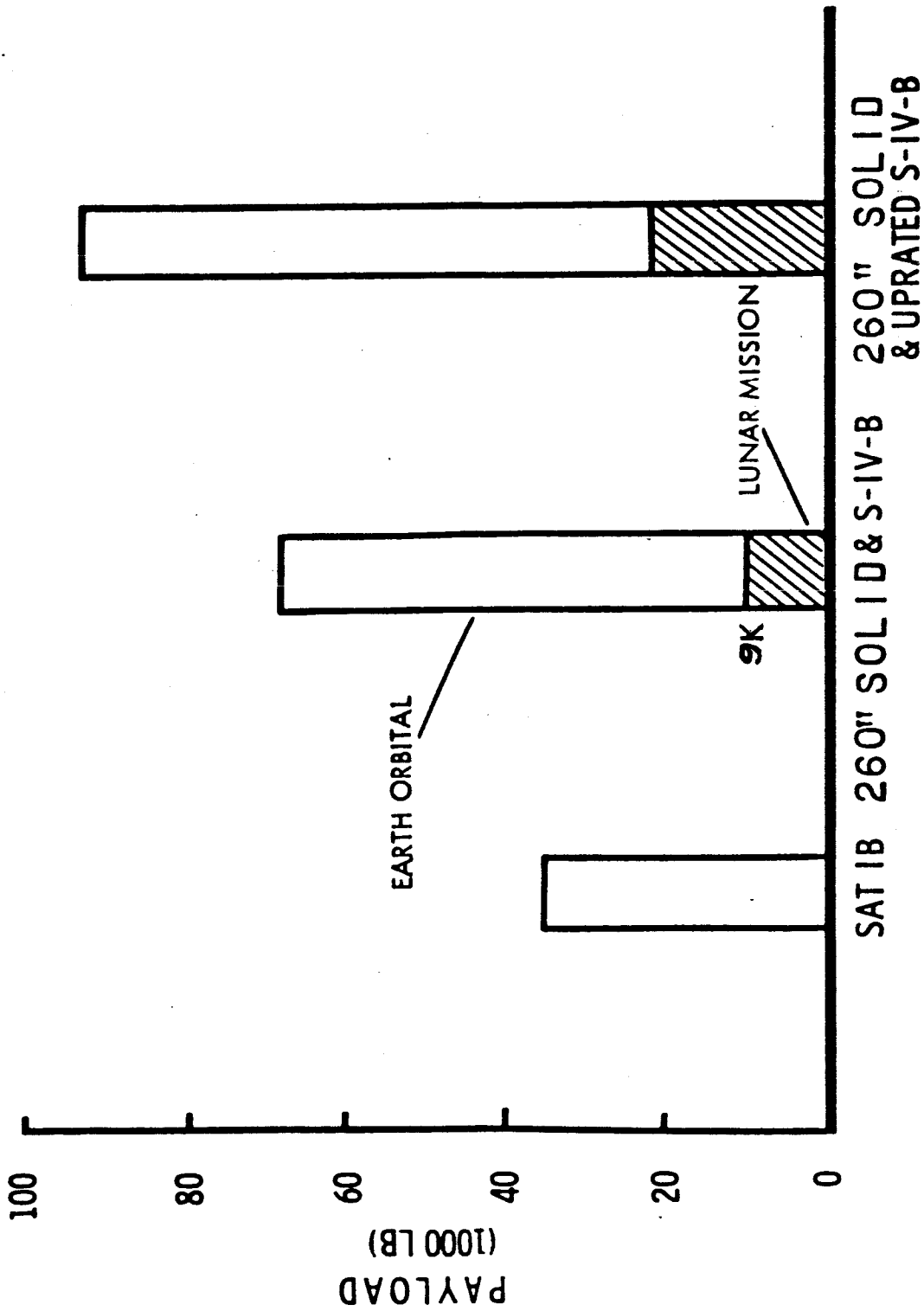
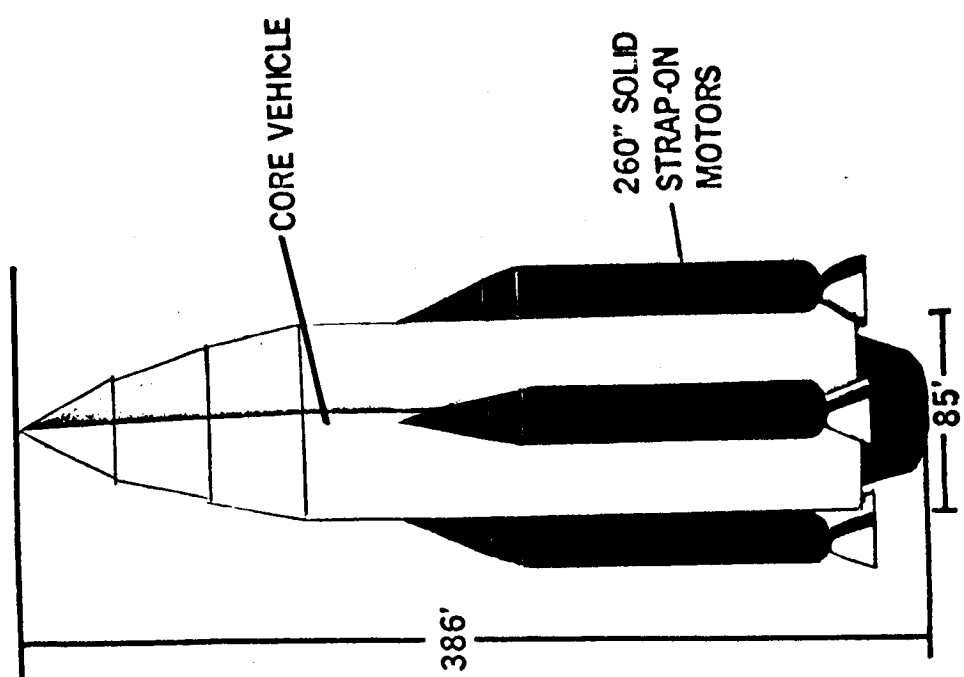


FIGURE 9-37. PAYLOAD CAPABILITIES, TWO-STAGE VEHICLE
(Source: Ref. 4)



ORBITAL PAYLOAD - 1.7 M LBS
LAUNCH WEIGHT - 50.7 M LBS.

BOOST STAGE		CORE ENGINES	
MOTOR OR ENGINE TYPE	260 IN.	CRYO./LIQ.	
NO. OF UNITS	4	H ₂ /O ₂	
PROPELLANTS	SOLID	31 M	
WEIGHT - LBS	18 M	36 M	
THRUST			
STAGE TOTAL	32 M		
PER MOTOR	8 M		

FIGURE 9-38. LARGE LAUNCH VEHICLE STUDY CONFIGURATION
(Source: Ref. 4)

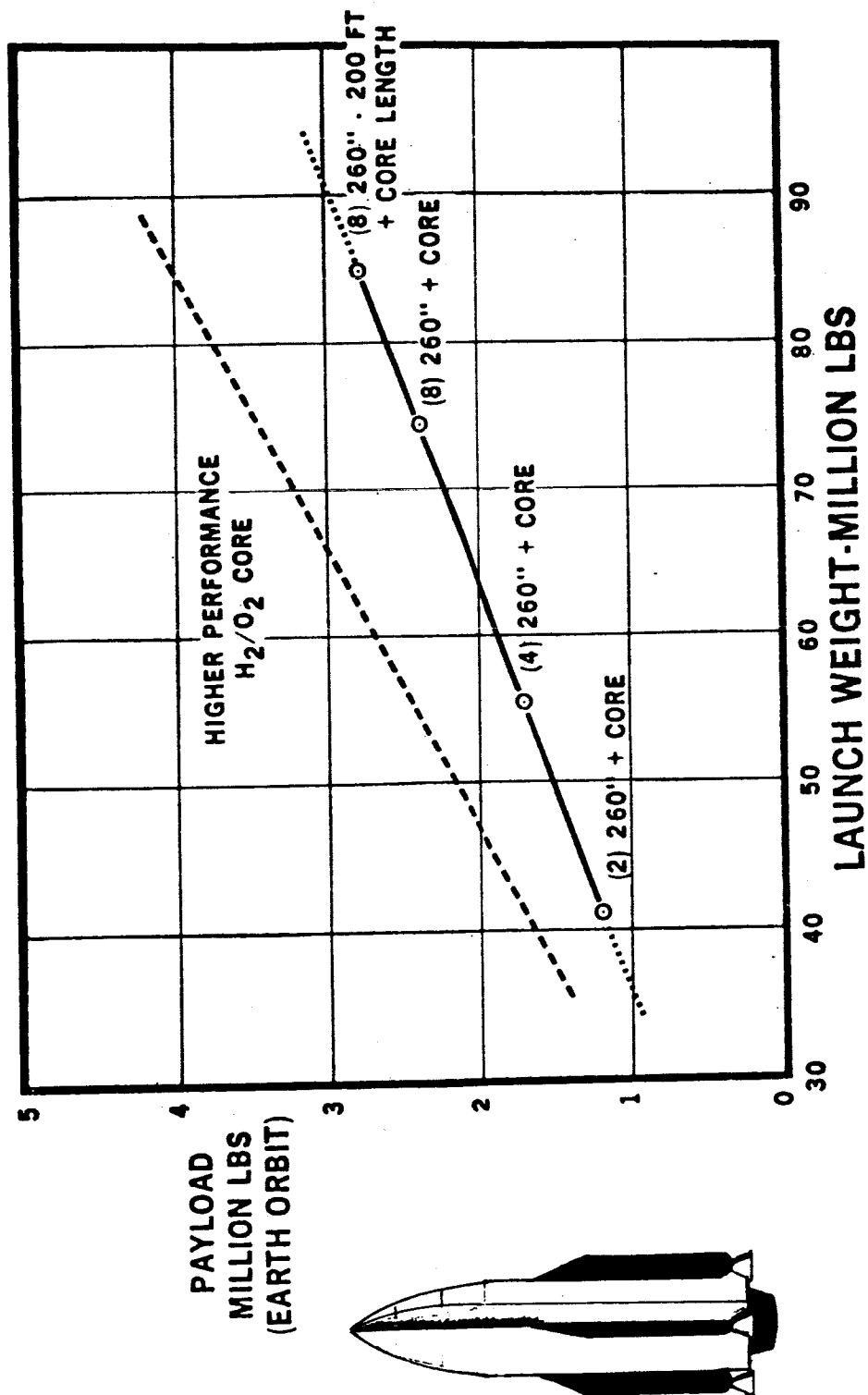


FIGURE 9-39. LARGE LAUNCH VEHICLE STUDY CONFIGURATION PERFORMANCE (Source: Ref. 4)

TABLE 9-9. SOLID BOOSTED VEHICLE COST SUMMARY

Concept	(Source: Ref. 4)			Source
	P/L in Orbit (pounds)	R&D Cost (\$10 ⁹)	Cost-Effectiveness (dollars/pounds)	
Saturn IB	35, 800	----	542(139)	Boeing (corrected)
Saturn IB and Solid Assist (4 x 120-inch Strap Ons)	74, 800 53, 500 68, 660	0. 87 0. 67 0. 28	375(80) 299(80) 263(120)	Boeing Boeing Douglas
All-Solid	64, 500	1. 47	330(100)	Boeing
Saturn V (2-stage)	245, 000	----	160(100)	Boeing
Saturn V and 4 (120-inch)	311, 000	1. 39	169(100)	Boeing
Saturn V and 2 (260-inch)	427, 000	1. 73	137(100)	Boeing
Saturn V (3-stage)	284, 855	----	149(100)	Boeing
Saturn IC Replacement	285, 540	1. 34	135(100)	Boeing
Study Configuration	2, 800, 000	3. 75*	83(25)*	Reference 4

Note: Figures in parenthesis represent numbers of missions on which cost is based.

*Quoting Ref. 4: "Cost estimates for study configuration do not deserve to be in the same table with other estimates -- shown here for completeness" (see Figure 9-38).

It is expected that low cost will be one of the primary reasons for development of solid boosters.

9.2.2 Self-Destruction

When solid boosters for small scientific or meteorological payloads are used over populated areas, it will be necessary to either completely destroy the rocket at apogee or recover the casing with a parachute, in order to avoid danger to life and property from falling debris.

It is possible, in concept, to provide the rocket with a completely combustible casing which will be converted to hot gases near apogee, leaving no debris to fall back (assuming payload separation prior to apogee). When a simple grain structure is designed for this, there is danger of flame-out, if nonuniform burning causes the forward end to burn through early. There also is a problem in disposal of the nozzle (or else there is an efficiency problem if it is omitted). Research and development is required.

9.2.3 Cutoff Precision

(Refer to Subsection 9.2.) Solid rocket thrust can be terminated upon command by blowout forward ports. However, cutoff precision is not comparable to that possible with throttleable liquid rockets.

It has been suggested that a vernier array of small auxiliary rockets arranged in thrust magnitude decades can be fired sequentially by the guidance computer to add extra thrust (after early cutoff) with precision comparable to that of liquid rockets.

9.2.4 Impulse Accuracy

Aside from cutoff precision (paragraph 9.2.3) the thrust level of solid rockets can be tailored to be almost as continuous and constant as that achievable with liquids. However, maintaining a low variability of total impulse will add to total cost by requiring greater care in mixing and casting grains with homogeneity and precision surfaces.

9.3 SCRAMJET

It is becoming increasingly apparent that acceleration to near orbital velocities might be achieved by airbreathing engines in the thin air of the stratosphere. Hybrid and multimode engines have been studied which show exciting promise of eventually providing a truly economical means for near ballistic flight without costly rocket boosters.

Not only could this permit placing large payloads into orbit from jet airstrips, but it could open up the possibility for a profitable one hour passenger service to any major city on Earth.

Figures 9-40 and 9-41 indicate the possibilities.

However, extensive research and development on airfoil cooling, airframe stability, optimum speed-altitude profiles and flight control are required along with development of the scramjet and turbo-scramjet engines.

9.4 NUCLEAR-FISSION ROCKETS (From Reference 7)

The Application of Nuclear Power to Rockets

"The thermal engines based on chemical fuels are ultimately limited by both the weakness of the chemical bond and the high molecular weight of chemical combustion products. By introducing nuclear power into rocket propulsion, both limitations may be overcome. The high energy densities of nuclear-heat sources are limited only by our ability to remove the thermal energy that is generated. In addition, great freedom in the choice of propellant is permitted since chemical combustion is unnecessary. Both of these advantages combine to make nuclear-fission rockets the prime contenders for the roles now played by the chemical rockets in space technology and for new roles in interplanetary flight.

"There are three basic types of nuclear fission rockets. They may be classified by the manner in which the nuclear reaction is contained: (1) The so-called "heat-transfer nuclear rocket" is no more than a nuclear-reactor core through which propellant is passed and heated. Figure 9-43 illustrates this type schematically. (2) The second type is the consumable nuclear rocket. Here the nuclear fissioning occurs directly in the working fluid. Both unfissioned fuel and fission products are released in the exhaust. This type is conceptually midway between the heat-transfer reactor and an atomic bomb. (3) The final type of

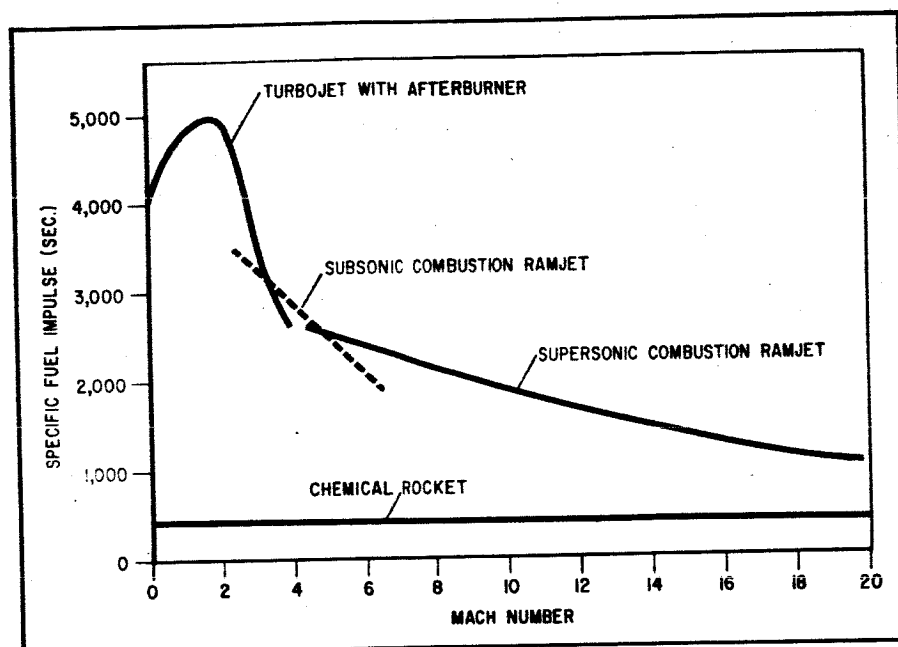


FIGURE 9-40. SUPERIOR PERFORMANCE AREAS

Superior Performance Areas of hydrogen-fueled turbojet with afterburner, subsonic-combustion ramjet and supersonic-combustion ramjet (scramjet) are shown in chart in terms of specific (fuel) impulse as calculated by several researchers. The Air Force Systems Command is pressing for funds to accelerate its scramjet propulsion program.

(From AVIATION WEEK & SPACE TECHNOLOGY, 12 July 1965
Yaffie, Michael L., Page 53.)

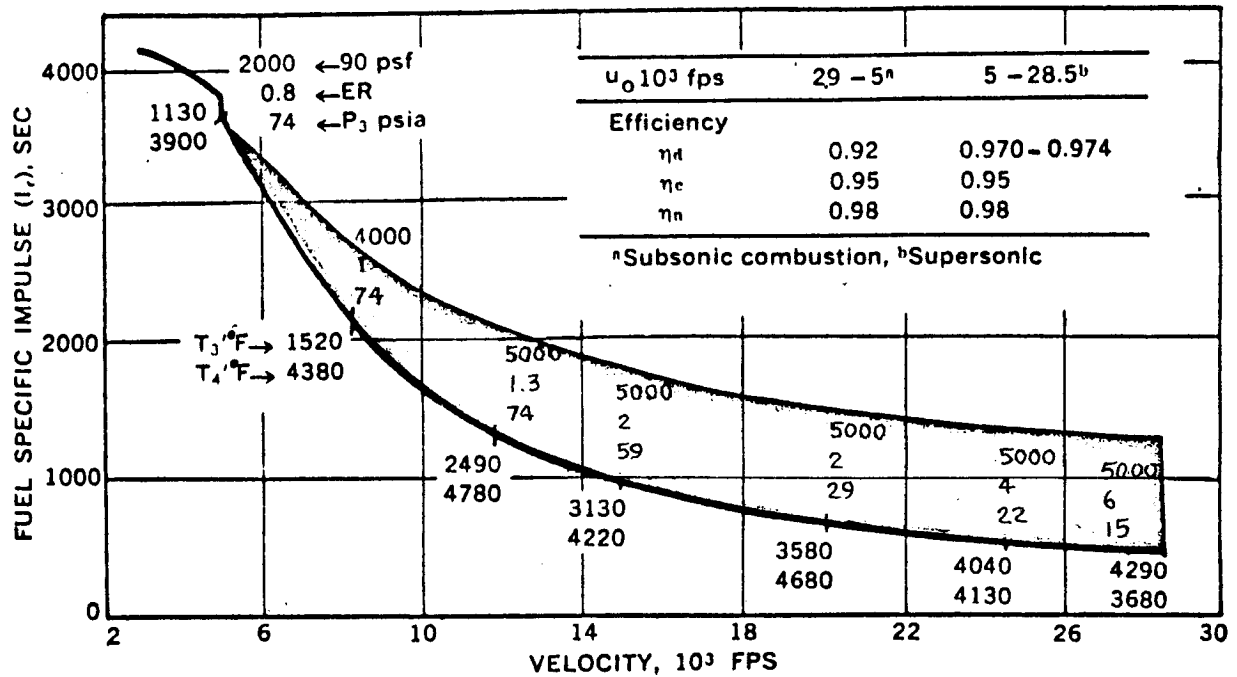


FIGURE 9-41. ESTIMATED HYPERSONIC-RAMJET PERFORMANCE
(Source: Ref. 16)

ESTIMATED EARLIEST AVAILABILITY OF POSSIBLE REUSABLE SPACE VEHICLES

Vehicle type	1970	1980	1990
Earth-orbital transport vehicle:			
10-passenger ROT		▲	
100-passenger global transport			▲
Interorbital ferry vehicle:			
Earth orbit-to-orbit		▲	
Earth-lunar orbit		▲	
Earth-planetary orbit			▲
Lunar surface-orbit shuttle		▲	
Planetary surface-orbit shuttle			▲

ESTIMATED EARLIEST APPLICATION OF RSV PROPULSION SYSTEMS

Propulsion type	1970	1980	1990
Chemical propulsion	▲		
Solid-core nuclear propulsion		▲	
Nuclear-pulse propulsion			▲
Nuclear-electric propulsion			▲
Gas-core nuclear propulsion			▲
Controlled thermonuclear propulsion			▲

POSSIBLE RELATION OF PROPULSION TO REUSABLE SPACE VEHICLES

VEHICLE CLASS \ PROPULSION TYPE	Chemical propulsion	Solid-core nuclear propulsion	Nuclear-pulse propulsion	Nuclear-electric propulsion	Gas-core nuclear propulsion	Controlled thermo-nuclear propulsion
Earth-orbital carriers	■					
Earth interorbital ferries	■	■			■	
Earth-lunar orbit ferries		■	■		■	
Earth-planetary orbit ferries		■	■	■	■	■
Lunar surface-orbit shuttles	■					
Planetary surface-orbit shuttles	■		■		■	

FIGURE 9-42
(Source: Ref. 17)

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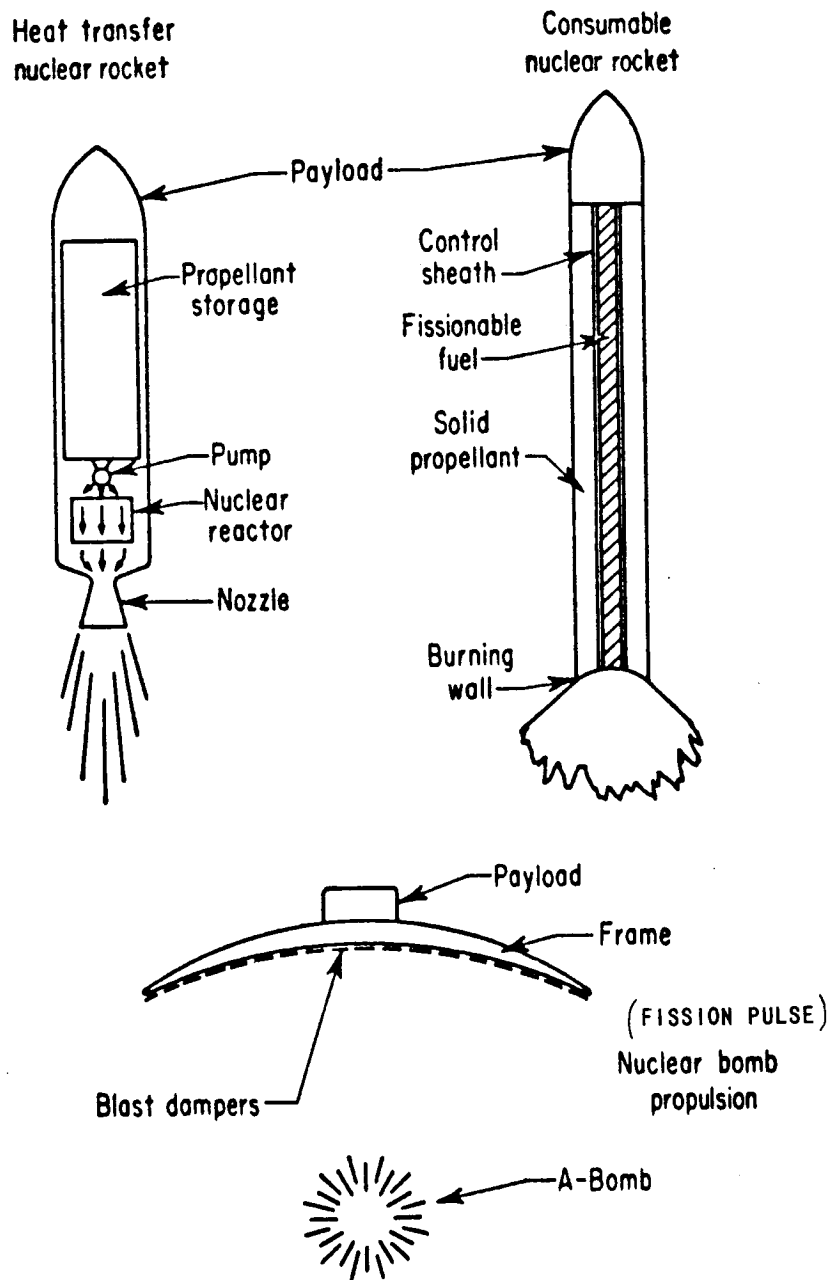


FIGURE 9-43. THREE TYPES OF
NUCLEAR PROPULSION SYSTEMS (ALL THERMAL ENGINES)
(Source: Ref. 7)

nuclear fission rocket is the controlled-explosion type. It is shown in Figure 9-43 in the firecracker-under-can configuration. Each of the three shows promise as an eventual space propulsion system.

Heat-Transfer Nuclear Rockets

"The arrangement of the components in a heat-transfer nuclear rocket is very similar to that in the liquid-chemical rocket. The vehicle is projectile-shaped with the propellant tank occupying much of the vehicle volume. The nuclear reactor occupies a position identical to that of the chemical combustion chamber. Propellant is pumped through the hot core, vaporized, heated to high temperatures, and then expanded through a nozzle to produce thrust. The designation "heat transfer" comes from the fact that the great bulk of the heat generated in the fuel elements is transferred to the propellant by the conventional heat-transfer mechanisms of conduction and convection.

"During the firing of the rocket, turbopumps drive the propellant from the tanks into the core header under high pressure. Pressures up to 100 atm and flow of 1,000 kg/sec are typical of many designs. The heart of the heat-transfer nuclear rocket is the reactor itself. Figure 9-45 illustrates one possible arrangement. Propellant under high pressure from the header enters the many coolant channels drilled in the core matrix material. Heat produced by fission within the matrix is conducted to the surface of the holes and then to the fluid itself by convection. Somewhere along its passage through the core, perhaps even before it leaves the top reflector piece, there will be a propellant phase change. When this occurs, heat-transfer coefficients will also change markedly.

"From this point on, the major problem in core design is the provision of the immense, extended heat-transfer areas needed to convey the heat generated in the elements to the gas. Many thousands of square meters of core area will be required in large rockets. The least blockage of coolant flow will cause hot spots to develop with possible vaporization of core material and the release of hazardous fission products to the exhaust. The reactor core is a region of intense energy production, perhaps 10,000 Mw/m³. It is exceedingly sensitive to the smallest variations in the heat-transfer process.

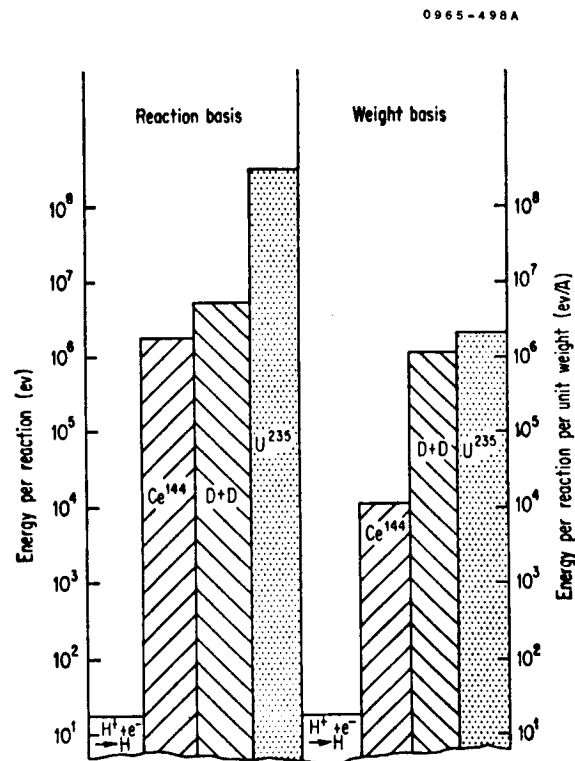


FIGURE 9-44. ENERGY AVAILABLE IN CHEMICAL REACTIONS, RADIOISOTOPE DISINTEGRATIONS, FUSION, AND FISSION
(Source: Ref. 7)

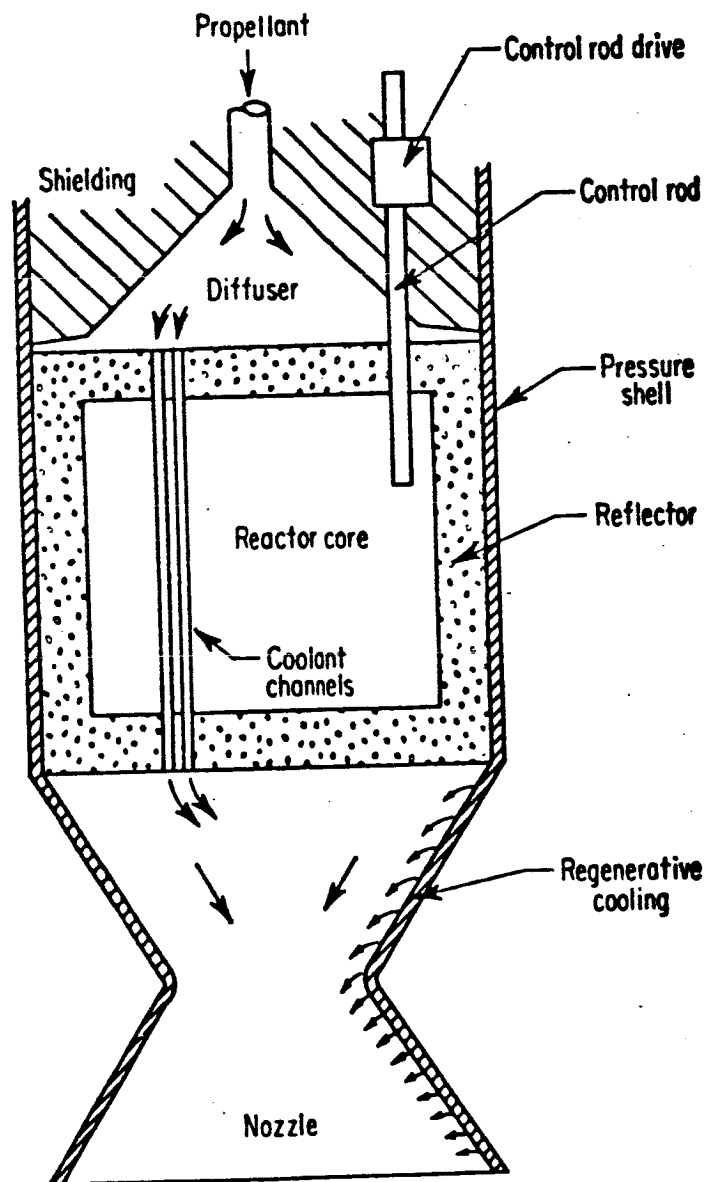


FIGURE 9-45. SCHEMATIC OF THE NUCLEAR-HEAT-TRANSFER
ROCKET ENGINE
(Source: Ref. 7)

Heat is transferred from the solid fuel elements to the working fluid through conduction, radiation, and convection. Propellant turbopumps and piping for regeneratively cooled nozzle are not shown. Shielding keeps propellant from overheating.

"Reactor control can be achieved through the use of axial control rods containing a neutron poison like boron carbide. The axial control rods shown in Figure 9-45 are common in most nuclear reactors. Upon start-up, the rods would be withdrawn, allowing the reactor power to build up at a rate consistent with the thermal stress and shock limitations of the core materials. Of course, propellant flow past the fuel-bearing regions must be programed along with the reactor power changes. The start-up procedure may take as long as several minutes if damage to the core is to be avoided. This single fact constitutes one of the major problems of the heat-transfer nuclear rocket. Ready-to-go capability is limited, and provided from an auxiliary supply to remove heat while the power is being brought up to the design level. Difficulties are also encountered during reactor shutdown because coolant must be directed through the core even after the chain reaction has ceased. Unless this afterheat, a result of radioisotope decay, is removed, the rocket reactor will be destroyed and reuse will be impossible.

9.4.1 Shielding. "Most nuclear rockets will be unmanned, at least during the first years of their use. For this reason, shielding will be a minor problem. Only a few radiation-sensitive components, like transistorized electronic equipment, will have to be protected from the intense burst of radiation occurring during the short operating lifetime of the reactor. It is also essential to place some shielding between the reactor and the propellant for the absorption of nuclear radiation. Small temperature rises in the cryogenic propellant can cause severe cavitation problems in the propellant turbopump. For the eventual nuclear rockets with a human cargo, enough shielding must be placed between the reactor and the crew compartment to bring the total dose integrated over the mission down to perhaps 10 rem. The crew must also be protected from the neutrons and gamma rays that may be scattered into their compartment from the atmosphere. Scattering will be important only during the initial flight phase. It may be that the shielding necessary for human passage through the radiation belt surrounding the Earth will be adequate for air-scattered reactor-produced radiation.

"The direct and scattered radiation consists of two components: the gamma rays, which are best attenuated by high-density substances like lead and tungsten, and the fast neutrons, which should be stopped by light atoms like those present in water, the hydrides, and the organics. Fortunately, the propellant inventory and tankage will "shadow-shield" the crew compartment very effectively during much of the rocket's ascent. The long column of propellant and metallic structure will shield against both radiation components until just before rocket burn-out. As in the case of nuclear turboelectric power supplies, shielding depends so much upon the specific mission and vehicle at hand that generalizations cannot easily be made.

TABLE 9-10. PROPERTIES OF SOME NUCLEAR
ROCKET CORE MATERIALS

Property	BeO	Graphite	W
Melting point.....	2800°K	3900°K, sublimes	3650°K
Density at 20°C.....	2.7	1.7	19.1
Short-time tensile strength, psi.....	1,000-4,000, at 1500°K	3,000-6,000, at 2780°K	5,000-9,000, at 2780°K
Macroscopic thermal absorption cross section, cm ⁻¹	0.0006	0.00037	1.17
Probable rocket ef-flux temperature in 1970.....	1700°K	2300°K	2300°K
Comments.....	Comparatively well developed. Can be hot-pressed into almost any shape	Properties are anisotropic. Easily worked	Difficult to fabricate. Expensive

(Source: Ref. 7)

TABLE 9-11. SPECIFIC IMPULSES
OF NUCLEAR ROCKETS

Propellant	Sea-level thrust chamber specific impulse at the indicated gas temperature for 43 atm		
	1650°K	3300°K	4950°K
Hydrogen.....	627 sec	890 sec	1216 sec
Helium.....	395	540	653
Ammonia.....	307	431	577
Water.....	222		

(Source: Ref. 7)

9.4.2 Cryogenic Storage

"Unhappily, hydrogen and helium, the best nuclear propellants, are cryogenic fluids. Storage, handling, and pumping problems are severe. If instant readiness is a system requirement then it is doubtful whether these two propellants would be useful. The more tractable propellants (H_2O , NH_3) seriously reduce the performance of the nuclear rocket because of their high molecular weights. In the final analysis, the fluid chosen will be dictated by the mission requirements. The actual margin of improvement over chemical rockets will depend upon just what compromises have to be made. Hydrogen is the almost universal choice of propellant for the nuclear heat-transfer rocket."

9.4.3 Research and Development Required

"In spite of the concentrated AEC nuclear-rocket program (the first prototype nuclear rocket, KIWI-A, was successfully tested at the AEC test site in Nevada in 1959), it is doubtful whether nuclear heat-transfer rockets will become operational before 1970.

Table 9-12 illustrates some interesting points about nuclear heat-transfer rockets.

TABLE 9-12. SAMPLE NUCLEAR-ROCKET PARAMETERS

State of the art.....	1975	1975
Core material.....	Graphite	Graphite
Core-efflux temperature.....	3300°K	3300°K
Propellant.....	Hydrogen	Helium
Chamber pressure.....	43 atm	43 atm
Exhaust velocity.....	8,900 m/sec	5,400 m/sec
Specific impulse.....	890 sec	540 sec
Reactor thermal power.....	24,000 Mw	14,600 Mw
Propellant mass flow.....	257 kg/sec	424 kg/sec
Gross mass.....	114,000 kg	114,000 kg
Propellant mass.....	84,000 kg	97,000 kg
Reactor mass.....	7,000 kg	6,000 kg
Structural mass.....	5,000 kg	5,000 kg
Payload into low satellite orbit.....	18,000 kg	6,000 kg
Sea-level thrust.....	228,000 kg	228,000 kg

(Source: Ref. 7)

"The high-thrust nuclear rockets extend the area representing the chemical rockets on the $F/W-I_{sp}$ plane up to about 1,000 sec specific impulse. Their maximum thrust-to-weight ratios will probably be somewhat less than those of chemical rockets due to the desire to keep the reactor power level (and thus the heat-transfer areas) as low as possible.

"There are a great many real barrier problems that must be overcome before this type of propulsion system reaches fruition.

"The fuel elements must maintain their integrity under the intense thermal stresses in spite of the notorious tendencies of ceramics to spall and crack. Any deviation of coolant distribution, because of channel blockage by core fragments, will cause burnout with its attendant release of fission products. The reactor startup and control problem is as difficult as any.

"Similarly, the erosion of the core, resulting in fuel element damage and the release of radioactivity, is an unevaluated hazard. The launching sites of nuclear rockets may be seriously contaminated because of this release of radioactivity."

9.4.4 Gaseous Core Nuclear Fission Rockets

"The liquid type of consumable nuclear rocket (also called plasma core, fizzler, or cavity reactor rocket) makes use of the fact that a critical nuclear assembly may be made from a gaseous core and solid or liquid reflector. Figure 9-46 shows how this concept may be factored into the design of a nuclear rocket. Note that reasonable diameters of 1 to 2 m may be obtained using this principle. The vortex configuration shown is in many respects similar to that employed by some plasma jets. Cold propellant is introduced tangentially along the periphery of a thick-walled pressure vessel and spirals in toward the rocket axis. U-235 fuel is injected at some point to create the core region. In the hot core, the propellant and fuel are heated to perhaps 50,000°K and expelled through an orifice or nozzle. The relative proportions of core materials are kept constant by matching the inflow of fuel and propellant with the flow through the nozzle. The wall of cool propellant surrounding the hot core is at once the neutron reflector, the pressure vessel coolant, and the radiation shield protecting the wall from thermal and nuclear radiation. Very high pressures are needed in this system to reduce the physical size of the core. Nuclear calculations show that core diameters of 2 to 3 m are appropriate for temperatures of 30,000°K and 5,000 atm pressure. These pressures necessitate immense pressure vessels, which in turn control the minimum sizes of consumable rockets."

Research and Development Required, Gaseous Core Nuclear Rocket. "Aside from the question of absolute magnitude, the real technical problems center around:

- A. The attainment of a high-opacity fluid or vapor to protect the pressure vessel wall from the intense thermal and nuclear radiation

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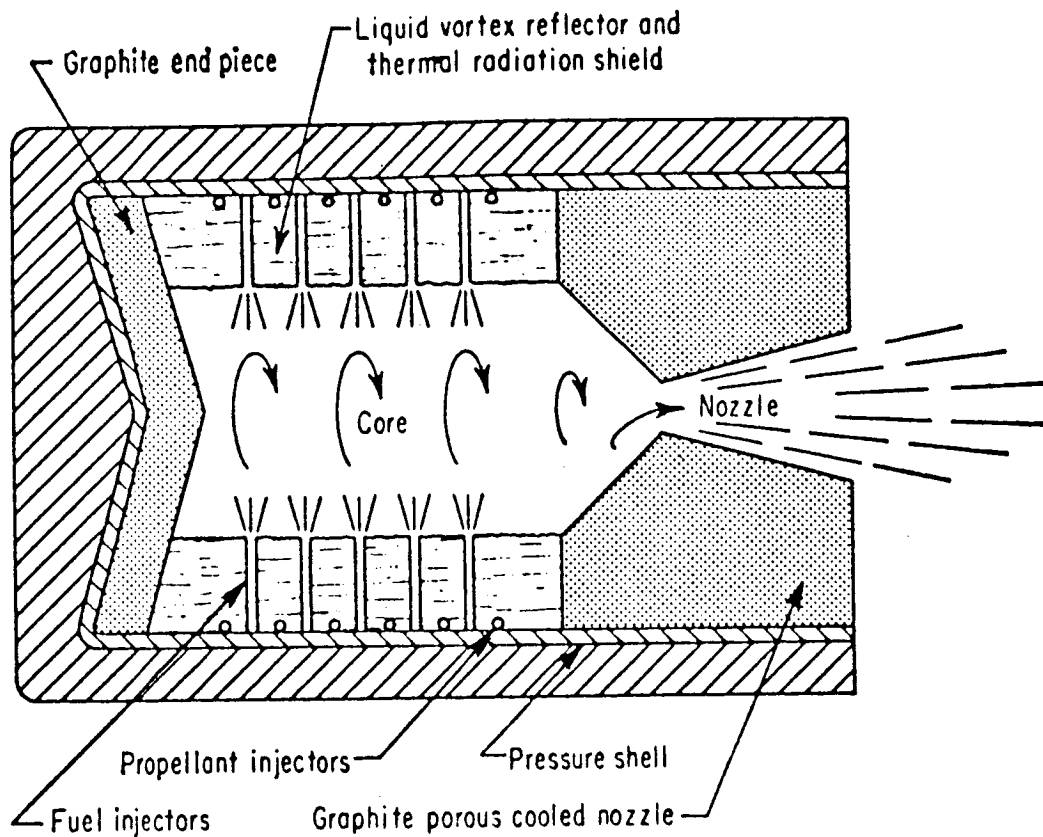


FIGURE 9-46. SCHEMATIC OF A GASEOUS CORE,
CONSUMABLE NUCLEAR ROCKET

(Source: Ref. 7)

The vortex action of the spinning propellant causes some retention of the heavy uranium atoms.

being emitted by the core. Present indications are that the opacity of available materials is at least an order of magnitude too low. Although the thickness of the buffer layer may be increased to compensate for low opacity, it cannot be made larger by a factor of 10 without making the rocket overall dimensions unwieldy. Buffer layer thicknesses of several meters seem necessary to sufficiently attenuate the radiation."

"Both liquid and solid consumable nuclear rockets must deal with two seemingly intractable problems. The first is the release of radioactive fission by-products. Although the energy needed to boost a 100,000 kg gross-mass nuclear rocket out of the gravitational influence of the Earth is only a small portion of that released in a large nuclear explosion, and the total amount of fission products correspondingly less, the political resistance to any dirty nuclear rocket would be difficult to overcome. It is easy to predict that international bans against bomb testing will be made to apply to consumable nuclear rockets." (The problem) "may be alleviated by first boosting them to altitudes so high before firing that most of the radioactivity will have disappeared by the time atmospheric circulation (a 10-year cycle has been measured) brings them back to Earth. This technique is possible in theory, but in practice the start-up and control of the rocket on a mobile launching pad are impressive and probably preclude this kind of solution. The second problem common to all consumable nuclear rockets is the waste of expensive fissionable material. Nuclear bombs, reactors, and rockets generally utilize less than 20 percent of the fissionable fuel mass in the original critical assembly. The rest is dispersed and lost. (In the case of stationary power reactors, it is reprocessed and refabricated into new fuel elements.) In the rocket where retention of fuel is nil, we may expect that approximately 1 kg of U-235 will be expended for each 100 kg of propellant in most designs. The ratio may be made more favorable only by considerably enlarging the size of the core in the liquid type. U-235 costs about \$20,000 per kilogram at the prices set by the Atomic Energy Commission. For the zero-retention case, it turns out that the consumable rocket can be competitive with chemical rockets in terms of dollars per kilogram placed in orbit. We calculate about \$50 to \$200 per kilogram for each kind of rocket. However, the situation is competitive only when the absolute size of the nuclear rocket is very large. Typical designs call for gross masses of 100,000 kg and uranium investments in the billions of dollars." (End quote from Ref. 7).

Thrust Vectoring. Gimbaling a gaseous core reactor appears impractical although not impossible.

Auxiliary thrusting chemical rockets very possibly may have to be relied upon for thrust vectoring unless a breakthrough in refractory materials for spoiler vanes occurs.

Fluid injection amplification also presents a possible solution.

Cooldown/Re-Start. The reactor is not very dangerously "hot" prior to firing. However, after firing, long half-life isotopes are present in the engine, and much more shielding is required than for a new engine. It has been suggested that the rockets be dropped rather than reused (the same propellant tanks and structure are retained, and another engine fired for the next mission ΔV).

Shielding. See paragraph 9.4.1.

Cryogenic Storage. See paragraph 9.1.3.

9.5 ELECTRICAL PROPULSION SYSTEMS (From Reference 7)

Introduction to Electrical Propulsion Systems

"In spite of the simplicity and unquestioned utility of thermal space propulsion systems, it is electromagnetic propulsion that captivates the imagination. The generation of thrust by the thermal acceleration of matter is somehow crude compared with the action of electromagnetic fields on charged particles. It is esthetically more satisfying to think of the ordered progress of ions or plasma down an accelerating tube rather than the helter-skelter motion of gas atoms through the nozzle of a thermal engine. There are many ways in which electromagnetic fields may be used to eject matter at high velocities from space vehicles.

"Most of the thermal systems display only modest specific impulses; generally, they are less than 3,000 seconds. While such engines possess many uses and are now the dominant class of space engines, it is not difficult to foresee the need for propulsion systems with specific impulses an order of magnitude higher. In particular, the satellite and interplanetary classes of missions present excellent opportunities for the application of the high specific impulses and low thrust-to-weight ratios that are typical of the electrical propulsion systems. (Also long term attitude control represents a worthwhile application.)

"Perhaps the key performance characteristic of the electrical engine is its inability to launch itself from the Earth's surface. The requirement of electrical rather than thermal power for thrust production is the fundamental reason behind this disadvantage. First, thermal energy must be converted into electrical energy; second, electrical energy is changed into the kinetic energy of the exhaust particles in the jet stream. The two-energy conversion steps in series add an extra burden of weight to the engine. The thrust-to-weight ratios of electric engines are generally less than 10^{-3} for this reason. Similarly, the ultimate performance and usefulness of these engines depend not only upon the development of electrical power supplies of low specific mass but also upon the design of highly efficient propulsion units. Figure 9-47 illustrates the effects of specific impulse and specific mass upon the thrust-to-weight ratio.

"The application of electrical energy to propulsion permits the acceleration of matter to velocities much higher than those attainable in the temperature-limited thermal engines. While electrical energy may be used to heat the working fluid, as in the plasma jet, the true electrical engines call upon electrostatic and electromagnetic fields to accelerate electrically-charged particles. In a sense, the presence of electrical charge on a particle or ion is a handle by which the electromagnetic fields can grasp and accelerate it to velocities approaching that of light itself.

Ion Rocket

"The ion drive or rocket is the oldest and most respectable of the many electrical propulsion systems that have been proposed for space travel. The use of electrostatic fields for electron acceleration in radio tubes was early extrapolated into a thrust-producing ion gun... Although it is difficult to imagine ion beams that are powerful enough to propel spaceships, the present governmental support for hardware development substantiates the practicality and usefulness of the concept for low-thrust missions.....

"Every ion propulsion system has three basic elements. These are:

1. The emitter, consisting of some sort of device to ionize neutral matter and separate the positive ions from the electrons for subsequent separate acceleration.
2. The accelerator, which takes the ions produced by the emitter and electrostatically accelerates them to the velocity specified by the mission specific-impulse requirements.
3. The beam neutralizer is the final element. It recombines the positive ions and electrons to ensure that the net charge of the

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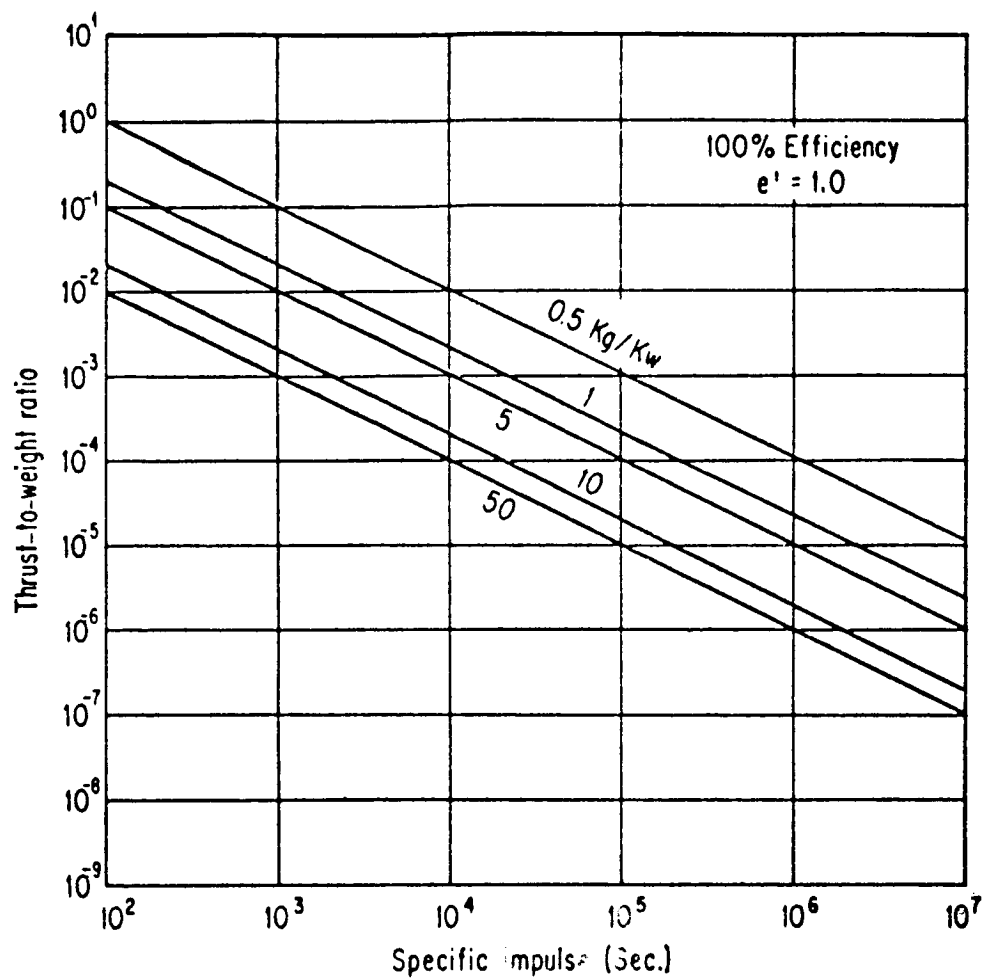


FIGURE 9-47. THRUST-TO-WEIGHT RATIO VERSUS SPECIFIC IMPULSE FOR ELECTRICAL PROPULSION SYSTEMS WITH POWER SUPPLIES OF DIFFERENT SPECIFIC MASS

(Source: Ref. 7)

propellant leaving the vehicle is neutral. It forestalls the build-up of space charge on the vehicle and consequent reduction of propulsion-system performance."

9.5.1 Research and Development Required

"The physical concept behind the ion drive is simplicity itself. Put a charged particle in an electrostatic field, and it will be accelerated... This direct procedure has esthetic appeal, and the equations descriptive of the phenomenon are readily understood. A closer look at any ion accelerator uncovers an impressive array of practical nuts-and-bolts problems. Ion-drive development is just entering the stage where the simplicity of the original concepts is being replaced by the complexities of the workable machine.....

"If the analytical determinations of radiation losses, beam focusing, and ion-source power consumption are confirmed in the laboratory, eventually ion drives will probably be over 75 percent efficient. This nearly perfect conversion of electrical energy into the directed kinetic energy of the propellant is essential if electrical propulsion is to play an important role in space propulsion. So far, the prospects are favorable. By the end of 1959 experimenters at General Electric had obtained total beam currents of 75 ma and propulsion-unit efficiencies of 15 percent. (Since then, model engines have been tested in space.-- Ed.)....

9.5.2 Operating Life (And Other Factors)

"Most of the ion propulsion systems that have been designed have specific impulses between 5,000 and 25,000 seconds. Thrust-to-weight ratios range between 10^{-5} and 10^{-4} . Table 9-13 lists the major parameters of interest for a representative cesium ion propulsion system (including design lifetime).

TABLE 9-13. DESIGN PARAMETERS
FOR A TYPICAL ION DRIVE *

Thrust.....	16 newtons
Efficiency.....	73%
Specific impulse.....	9,600 sec
Power consumed.....	1,000 kw
Propellant.....	Cesium
Propellant consumption.....	0.6 kg/hr
Design lifetime.....	10,000 hr
Thrust-to-weight ratio, including payload and propellant-structure allowance, 4,100 kg.....	1.7×10^{-4}
Mass of propulsion unit.....	500 kg
Mass of power supply.....	5,000 kg
Voltage.....	6,000 volts
Current in beam.....	121 amp
Beam power.....	730 kw

*Source: Ref. 7

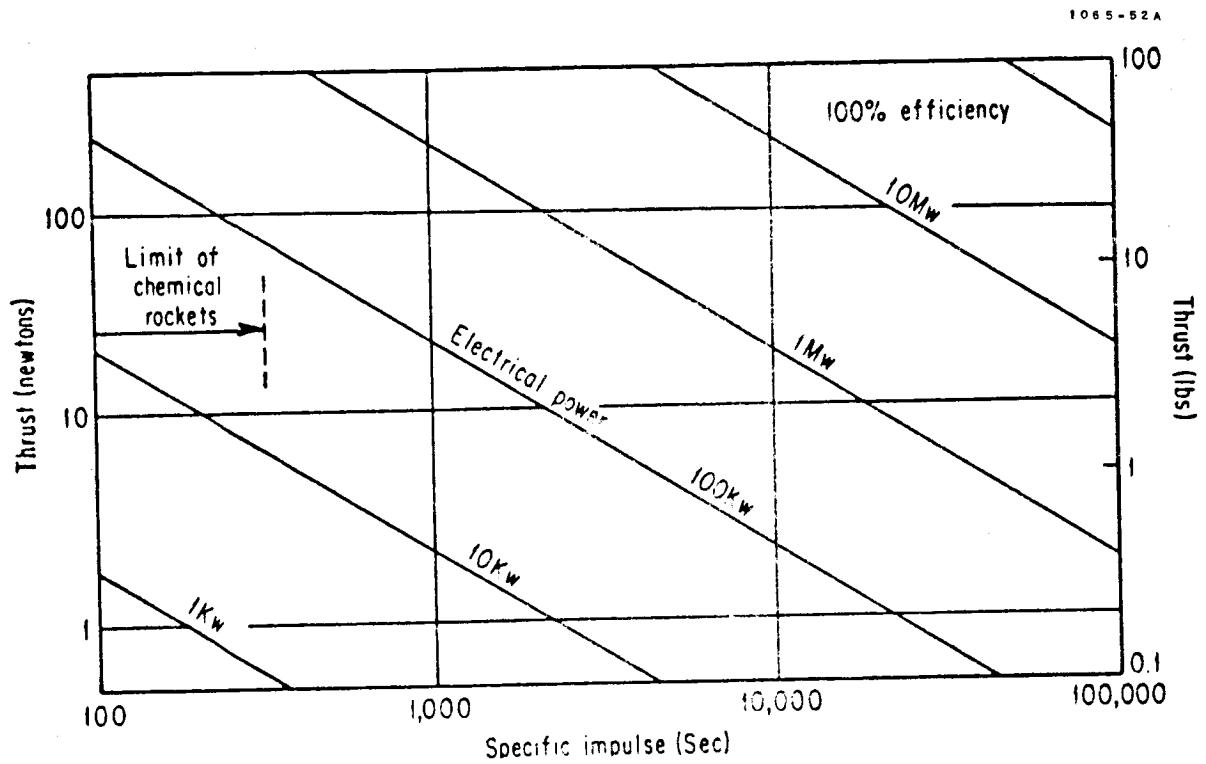


FIGURE 9-48. POWER REQUIREMENTS OF
ELECTRICAL SPACE PROPULSION SYSTEMS
(Source: Ref. 7)

Even small thrusts necessitate high power levels at high specific impulses.

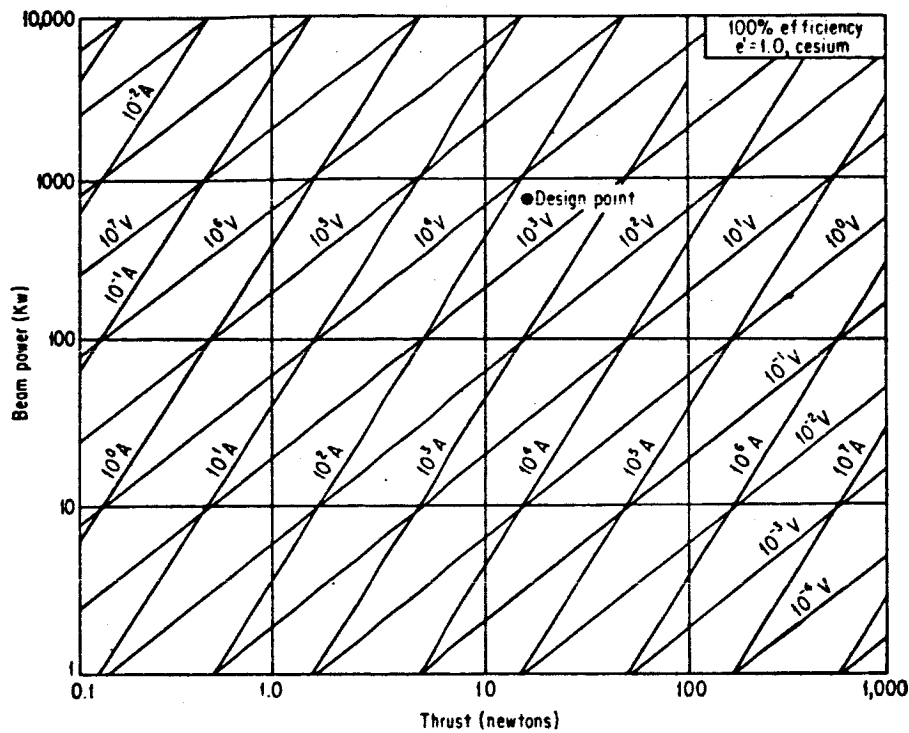


FIGURE 9-49. POWER-THRUST CURVES
FOR A CESIUM DRIVE
V = VOLTAGE, A = ATOMIC WEIGHT
(Source: Ref. 7)

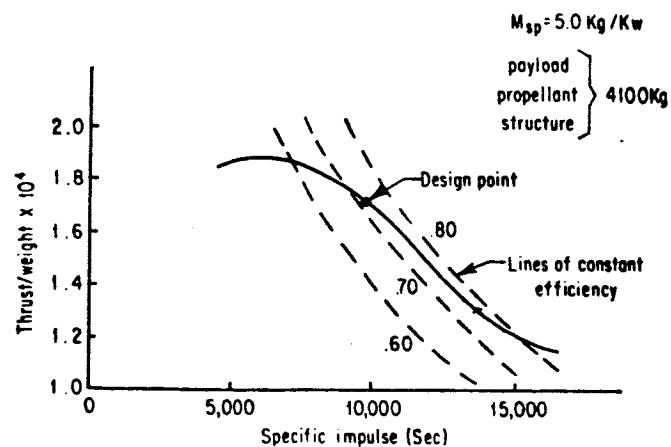


FIGURE 9-50. THRUST-TO-WEIGHT RATIO
(Source: Ref. 7)

Thrust-to-weight ratio as a function of specific impulse for a representative ion-driven spaceship. The thrust-to-weight ratio is reduced by decreasing efficiency at the low specific impulses and by increasing power-plant mass at the higher specific impulses.

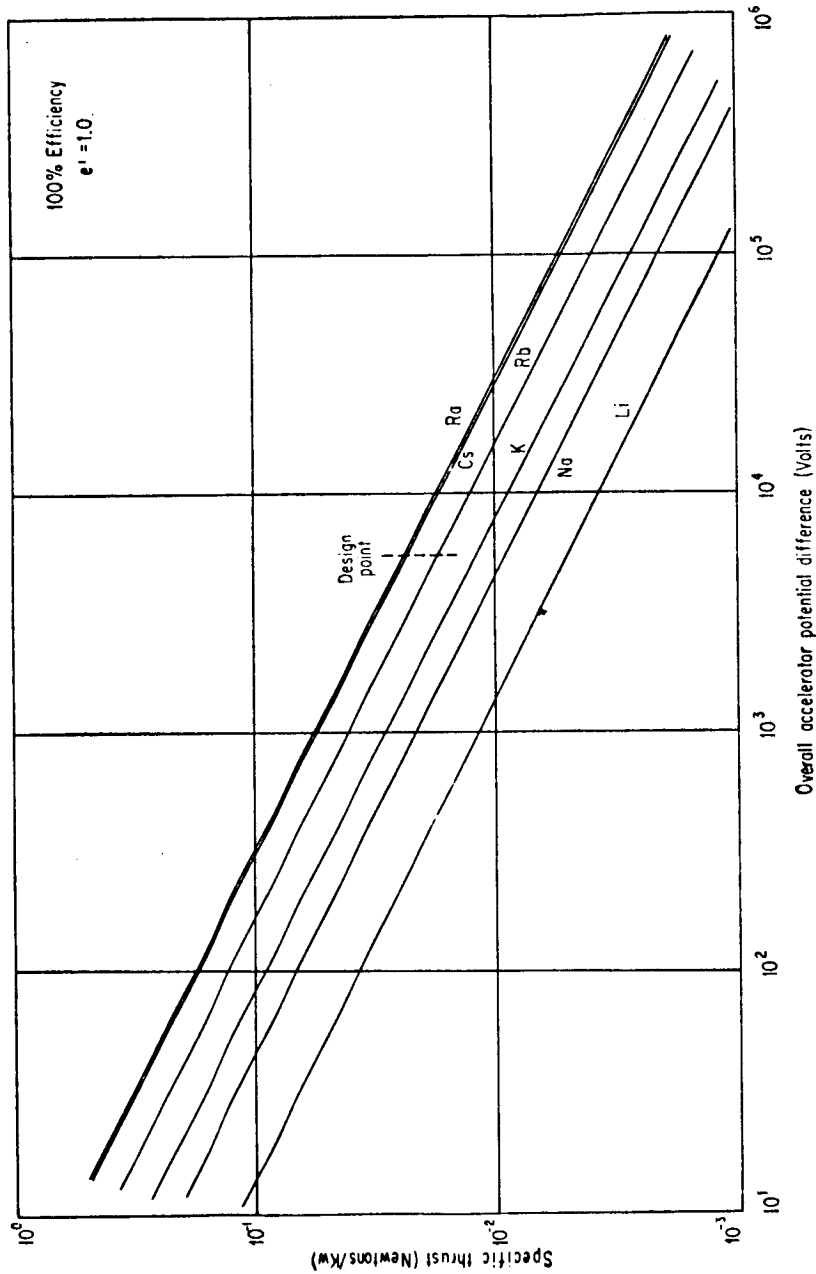


FIGURE 9-51. SPECIFIC THRUST FOR SEVERAL ION-DRIVE PROPELLANTS AS A FUNCTION OF ACCELERATOR VOLTAGE

(Source: Ref. 7)

Higher voltages produce higher specific impulses but require more power for a given thrust level. The specific thrust thus falls with higher accelerator voltages.

"The space vehicle that utilizes an ion propulsion system will not be the sleek, streamlined spaceship of the science-fiction magazines. Instead, extensive radiator surfaces will dominate the configuration, with the ion engines being insignificant protuberances or perhaps internally contained. While the unmanned, interplanetary, ion-driven probe will probably be a vehicle of only a few thousand kilograms, the manned, interplanetary spaceship will undoubtedly gross tens of thousands of kilograms.

"In addition to the electrical power supply, the ion drive will require propellant storage facilities, equipment for melting and pumping the propellant grid cooling systems with their own radiator sections, and separate power supplies for the emitters and beam neutralizers. It seems quite likely that all of these pieces of equipment can be incorporated into small, modular assemblies with a specific mass of about 0.5 kg/kw of power delivered to the propulsion unit." (End of quote from Ref. 7)

9.5.3 In-Flight Test Requirement

Some tests in orbit have been made, but several R&D flight tests are necessary before firm designs of vehicles for ionic-drive can be completed.

9.6 PISTON ENGINE WITH HYPERGOLIC FUEL

If electrical propulsion of roving surface vehicles is used, the range of action will be severely limited (the electric automobile problem).

A conceivable way out is to design an internal combustion engine for the main drive. Since atmospheric oxygen is not available on the Moon or Mars, oxidizer as well as fuel must be carried. Special, dual, metered-fuel-spurt injector systems would be required to operate from a cam shaft. Exhaust of combustion products would be to ambient outside pressure. In other respects the engine would resemble an automobile engine. Storable hypergolics probably would be the preferred reactants.

9.6.1 Research and Development Required

The hypergolic piston engine does not exist. Unforeseen problems will arise and working models will have to be tested in vacuum chambers. The possibility for extending rover radius of action to about 2,000 miles exists as a motive for further investigation.

9.7 TURBINE ENGINE WITH HYPERGOLIC FUEL

The problem is the same as that discussed in paragraph 9.6.

9.7.1 Research and Development Required

The choice between piston and gas turbine engines will have to be made on the basis of a specific set of mission requirements. In general the turbine offers higher efficiency and less weight per KW of output, but it requires speed reducing equipment for a suitable "mechanical impedance" match to the drive mechanism.

9.8 ELECTRIC MOTOR

(See paragraph 9.6) Electric motors driven from fuel cells have been proposed for lunar roving vehicles. These have an advantage in torque capability, but due to fuel cell weight may not be as effective as internal combustion drives.

9.8.1 Uncertainty of Lunar, Martian, Venusian Surfaces

Final vehicle design details, particularly those of traction and drive, require knowledge of environmental factors, which probably cannot be obtained without landed probes.

9.8.2 Weight

Electric motor weight is about 10 pounds per horsepower with controllers and gearing.

9.8.3 Mechanical Reliability

If ac motors are used, a variable frequency source of power, or a gear shift transmission would be required for speed control. DC motors have almost ideal speed control and low speed torque, but usually require commutators which are subject to wearout failure. So called brushless dc motors are really ac motors with silicon switches for dc to ac conversion.

In Earth environments, electric motor driven vehicles have operated for as long as 20 years with practically no maintenance (except brush replacement and bearing lubrication at infrequent intervals) and no failures. In unknown environments, such reliability is unlikely to be realized.

REFERENCE LIST FOR SECTION 9

- 9-1 (A) Aerospace Technical Forecast 1962-1972, Aerospace Industries Association.
- (B) 1960-1970 AIA Forecast.
- 9-2 V. Gradecak, Discussions on Planetary Transportation Systems at NASA MSFC FPO.
- 9-3 Paul Roman, "The Booster Crossroads," Space/Aeronautics, March 1965, page 58.
- 9-4 A. O. Tischler, and William Cohen, "Solid Propellant Motor Applications for Future Launch Vehicles," presented to 2nd annual meeting AIAA, July 1965.
- 9-5 L. F. Belew, W. N. Patterson and J. W. Thomas, Jr., "Apollo Vehicle Propulsion Systems," AIAA Paper No. 65-303, July 1965.
- 9-6 Edmund F. O'Conner, "Saturn Launch Vehicles," AIAA Paper No. 65-302, July 1965.
- 9-7 William R. Corliss, Propulsion Systems for Space Flight, (Textbook) McGraw-Hill, 1960.
- 9-8 V. Gradecak and J. C. Whiton, Discussions on Nuclear Pulse Space Vehicle Study at NASA MSFC FPO.
- 9-9 Robert J. Denington, William J. LeGray and Russel D. Shattuck, "Electric Propulsion for Manned Missions," NASA Lewis RC.
- 9-10 William J. D. Escher, "On the Utility of the Moon in Space Transportation - The Lunatron Concept," NASA MSFC Paper.
- 9-11 W. H. Wetmore, "High Energy Upper Stages - Potential Applications and Selection Criteria," AIAA Paper No. 65-345, July 1965.
- 9-12 Bernard Kovit, Associate Editor, "The Coming Kick Stage," Space/Aeronautics, August 1965, page 55.
- 9-13 H. O. Ruppe, Discussions on Study of Electrical Propulsion in Space at NASA MSFC FPO.

- 9-14 F. L. Williams, "Description and Status of the Saturn IB Program," Presented to AAS at Denver, Colorado, 8-10 February 1965.
- 9-15 David B. Miller, "Plasma Thrusters for Space Propulsion," IEEE Spectrum, June 1965.
- 9-16 William H. Avery and Gordon L. Dugger, "Hypersonic Airbreathing Propulsion," Astronautics and Aeronautics, June 1964.
- 9-17 H. H. Koelle, "Assessing Reuseable Space Vehicles," Astronautics and Aeronautics, June 1964.
- 9-18 "Saturn IB Payload Planners Guide," Missiles and Space Systems Division, Douglas Aircraft Co., Inc.
- 9-19 H. H. Koelle, Discussions of Trends in Earth-to-Orbit Transportation Systems at NASA MSFC FPO.
- 9-20 Walter H. Stafford and Robert M. Croft, "Space Flight Operations," NASA TN D-1119, July - December 1961.
- 9-21 Walter H. Stafford and Robert M. Croft, "Space Flight Operations," NASA TN D-613, January - June 1961.
- 9-22 Walter H. Stafford and Robert M. Croft, "Artificial Earth Satellites and Successful Solar Probes 1957-1960," NASA TN D-601, March 1961.
- 9-23 J. N. Smith, "Manned Mars Missions in the Unfavorable (1975-1985) Time Period, Executive Summary Report," NASA TM X-53140, 24 September 1964.
- 9-24 Lunar and Interplanetary Flight Section, Propulsion and Vehicle Engineering Laboratory, Marshall Space Flight Center, "Interplanetary Trajectory Analysis for Missions from Earth to Mercury, Jupiter, and Saturn," NASA TM X-53046, 11 May 1964.
- 9-25 Walter H. Stafford, et al, "Orbital Parameters for Artificial Earth Satellites up to 40,000 KM Altitude," NASA MTP-P&VE-F-63-19, 29 August 1963.
- 9-26 Walter H. Stafford, et al, "Parametric Performance Analysis for Interplanetary Missions Utilizing First-Generation Nuclear Stages," NASA TN D-2160, October 1964.

- 9-27 Walter H. Stafford, et al, "Performance Analysis of High-Energy Chemical Stages for Interplanetary Missions," NASA TN D-1829, January 1964.
- 9-28 Walter H. Stafford and Carmen R. Catalfamo, "Performance Analysis of Chemical Stages in the 300 to 400 second Specific Impulse Range for Interplanetary Missions," NASA TM X-53002, 5 February 1964.

Section 10

COMMUNICATIONS

10.0 GENERAL COMMENTS

Communications includes all equipment functions needed to perform point to point or broadcast transmission and reception of data pertinent to accomplishing a mission.

10.0.1 Bits, Power, Noise and Bandwidth

Shannon's equation:

$$\text{Information transmitting capacity} \approx \text{bandwidth} \times \log_2 \left[1 + \frac{\text{average signal power}}{(\text{bandwidth}) (\text{noise power density})} \right]$$

(in bits per second) (cycles per sec)

expresses the ideal maximum bit rate possible with a communications channel in which each pulse (symbol) may have a maximum number of distinguishable levels per symbol of $\left[1 + \frac{\text{average signal power}}{\text{average noise power}} \right]$.

$$\text{Therefore, average number of bits per symbol} \approx \log_2 \left[1 + \frac{\text{average signal power}}{\text{average noise power}} \right].$$

Also, the number of independent symbols per second which can be transmitted is roughly equal to the channel bandwidth W in cycles per second. The rate at which information can be transmitted is the product of the number of symbols per second and the number of bits per symbol.

Since in simple binary coding the "symbol" can have two states, low or absent (zero) and high or present (one), the hasty conclusion may be drawn that it is always possible to send two bits of information for each cycle per second of bandwidth, with a signal to noise ratio of only one. It is helpful to remember that if a one cps sine wave is altered in any way from pure sinusoidal, it must contain harmonics. Also, whether that one cycle is present or absent, it is really only one bit per unit cycle of time. That is, simple binary encoding always requires at least one cps of bandwidth per bit per second. To take full advantage of the theoretical maximum

number of distinguishable amplitude levels, one would have to devise an optimum code (for the available S/N ratio). Since this is impractical, the "Shannon limit" is almost never reached, but is approached by analog systems.

As a rule of thumb, the bit rate usually is slightly less than bandwidth, in real life digital systems, and with certain modulation techniques may be as little as 1/6 of the bandwidth.

Quoting Reference 7:

"Transmission of Messages in English (Non-vocal). It has been estimated that the average word of English text contains about 12 bits of information, if there is no restriction as to the subject matter of the message. Thus, transmission of English at 50 words per minute would require ideally a channel capacity of about 10 bits per second. This requirement could be reduced if the possible subject matter of the messages were restricted to some limited category, or if the transmission were at a slower rate. (Reliable reproduction of the human voice speaking English would require a considerably greater channel capacity.)

"Telemetry of Scientific Data. The required channel capacity in this case depends upon the number of different kinds of data to be telemetered, the desired accuracy of the readings, and the desired frequency of readings. Thus, the number of bits per second could conceivably range from almost zero to almost infinity. The following example illustrates a possible situation: Suppose it were desired to measure a quantity to an accuracy of one part in a thousand, and to take a reading every second. Also assume that the measured quantity actually varies significantly in one second. The required channel capacity would be $\log_2 1000 \approx 10$ bits per second.

"Television. It is estimated that the average commercial television picture requires very roughly 10^7 bits per second in channel capacity. However, a planetary surface, even with a human base on it, would be likely to move around much more slowly than Jackie Gleason, for example. Also, the resolution and definition required might not be as high. These factors could reduce the required channel capacity. " (For example, a one frame per second TV would require about $10^7/30$ or about 3×10^5 bits per second. Refer to Section 2, "Data Processing", for tabulation of TV bit rates and bandwidths - Ed.) (End of quote from Ref. 7)

10.0.2 The Spectrum

The use of modulated energy for telecommunications is implicitly a matter of producing vibration (of a medium) in the space between the communicating entities. Figures 10-1 and 10-2 show the spectrum of vibrations currently known, by which transfer of information is possible. In general, the communications of the next 20 years are likely to be concerned chiefly with electromagnetic vibrations in free space or ether, and are likely to be limited to two broad bands:

Radio energy between 10^4 and 10^{11} Hertz (cps)

Optical energy between 10^{11} and 10^{16} Hertz

Quoting Reference 5:

"It is expected that broad new areas of the vibratory spectrum will be further developed for communications. Command communications with the ability to survive attack, and be secure against message alteration by jamming or by natural phenomena are essential. It appears easiest to supply these capabilities with systems operating under 300 kilohertz or above 300 megahertz. There is a steady increase in the amount of information, civil and military, which must be transmitted. Accompanying the need for increased total information is a requirement for greatly increased data rates. The greater spectral bandwidth, ease of beam transmission and current crowding of other spectral regions indicate that the over 300 mhz region must be used for the high volume systems."

Figure 10-3 forecasts the expected relative emphasis on various portions of the spectrum.

10.0.3 The One-Way Range Equation

The power (S_R) at a receiver may be expressed as

$$S_R = \frac{(\text{Avg. transmitter power in watts}) (\text{Transmitter gain factor}) (\text{Effective area of receiving antenna in square meters})}{4\pi (\text{Range in meters})^2 (\text{System loss factor})}$$

"Free Space Loss. For line-of-sight transmission in free space, the relation of received power P_R to transmitted power P_t , between two antennas with a gain of unity, if treated by the direct ray theory, may be represented by:

$$\frac{P_R}{P_t} = \left[\frac{(\text{Velocity of light in meters per second})}{4\pi (\text{distance in meters}) (\text{frequency in cycles per second})} \right]^2$$

In decibels this ratio is

$$\text{Free space loss} = L_{FS} = -36.6 + 20 \log_{10} d_{\text{miles}} + 20 \log_{10} f_{\text{mhz}}$$

Suppose that a space vehicle carries a 100-watt transmitter with a unity gain antenna and transmits, with a free space loss of 10 db, to a receiver with an effective area of 10^4 square meters (118 m diameter) on the Earth. The received signal power for some representative ranges then would be:

<u>Range</u>	<u>Power (watts)</u>
5×10^4 kilometers ('distant' Earth satellite)	$1/4 \times 10^{-10}$
5×10^5 kilometers (lunar distance)	$1/4 \times 10^{-12}$
10^7 kilometers (closest asteroids)	10^{-16}
10^8 kilometers (Mars at close approach)	10^{-18}

A typical radar receiver might be sensitive to 10^{-14} watts as a point of comparison.

Note that a 10-watt transmitter using a directional antenna with a gain of ten''(or a one watt transmitter using an antenna with a directionality gain of 100)''could produce the same received signals. This points up the desirability of using Earth-tracking directional antennas on space vehicles."

Table 10-1 gives some ball-park estimates of transmitter power for some rule-of-thumb distances and bit rates (noise environment, etc. was considered).

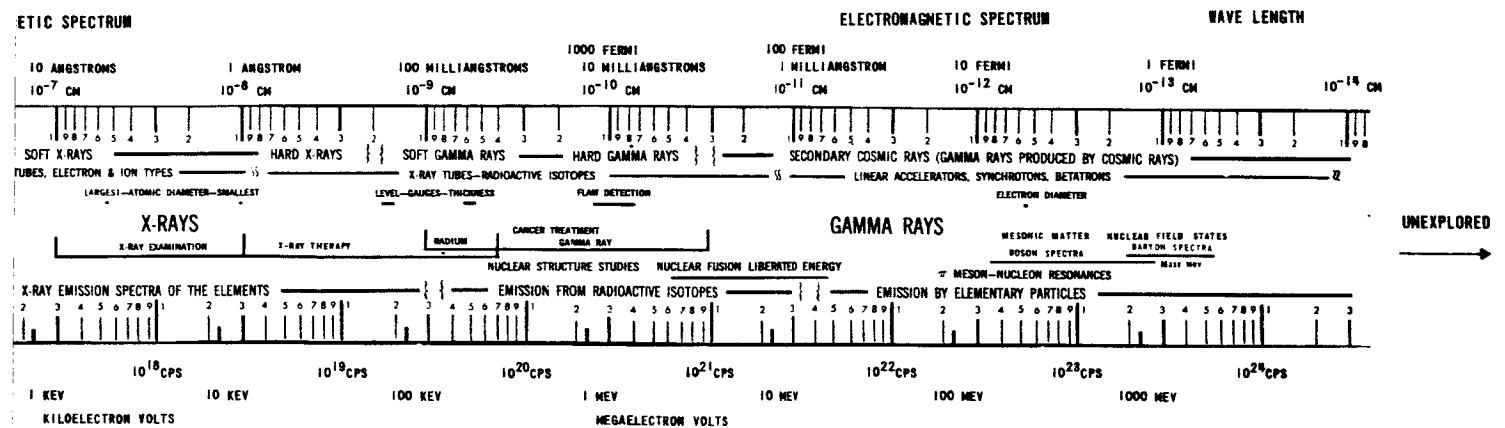
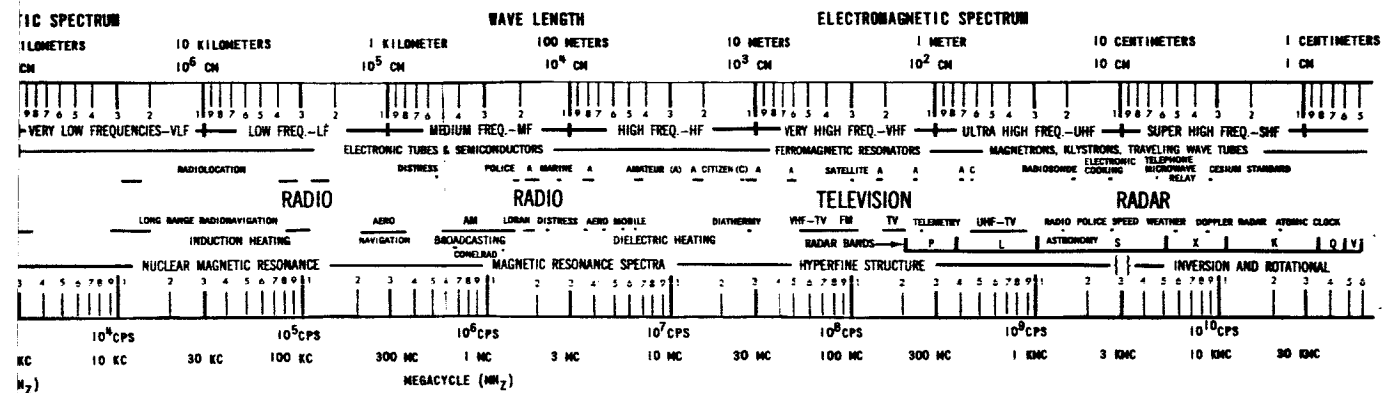


FIGURE 10-1. SPECTRUM OF VIBRATIONS USEFUL FOR TRANSFER OF INFORMATION

10-5

10-6

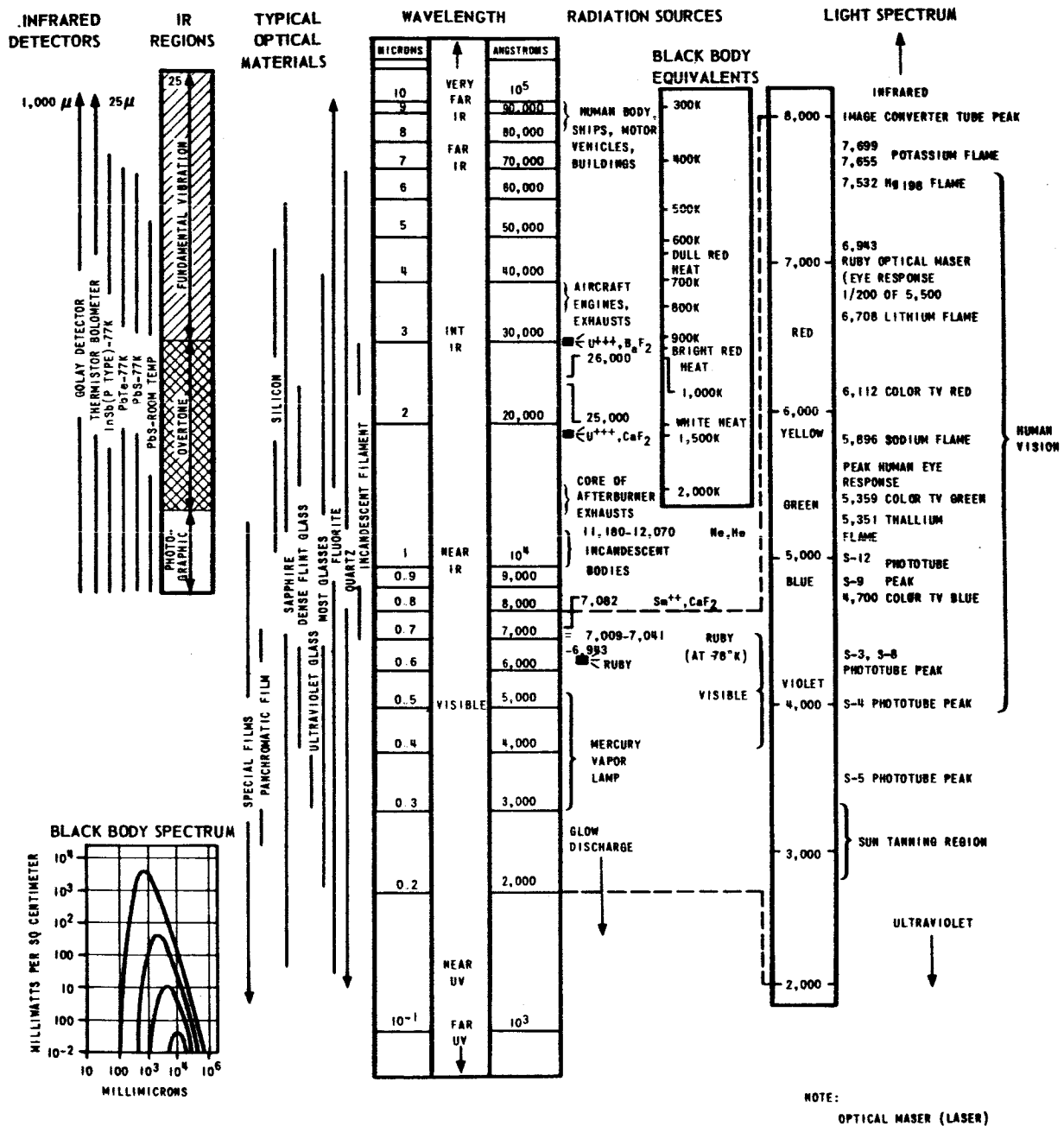


FIGURE 10-2. THE USEFUL OPTICAL SPECTRUM
(Unidentified open literature source)

0985-403A

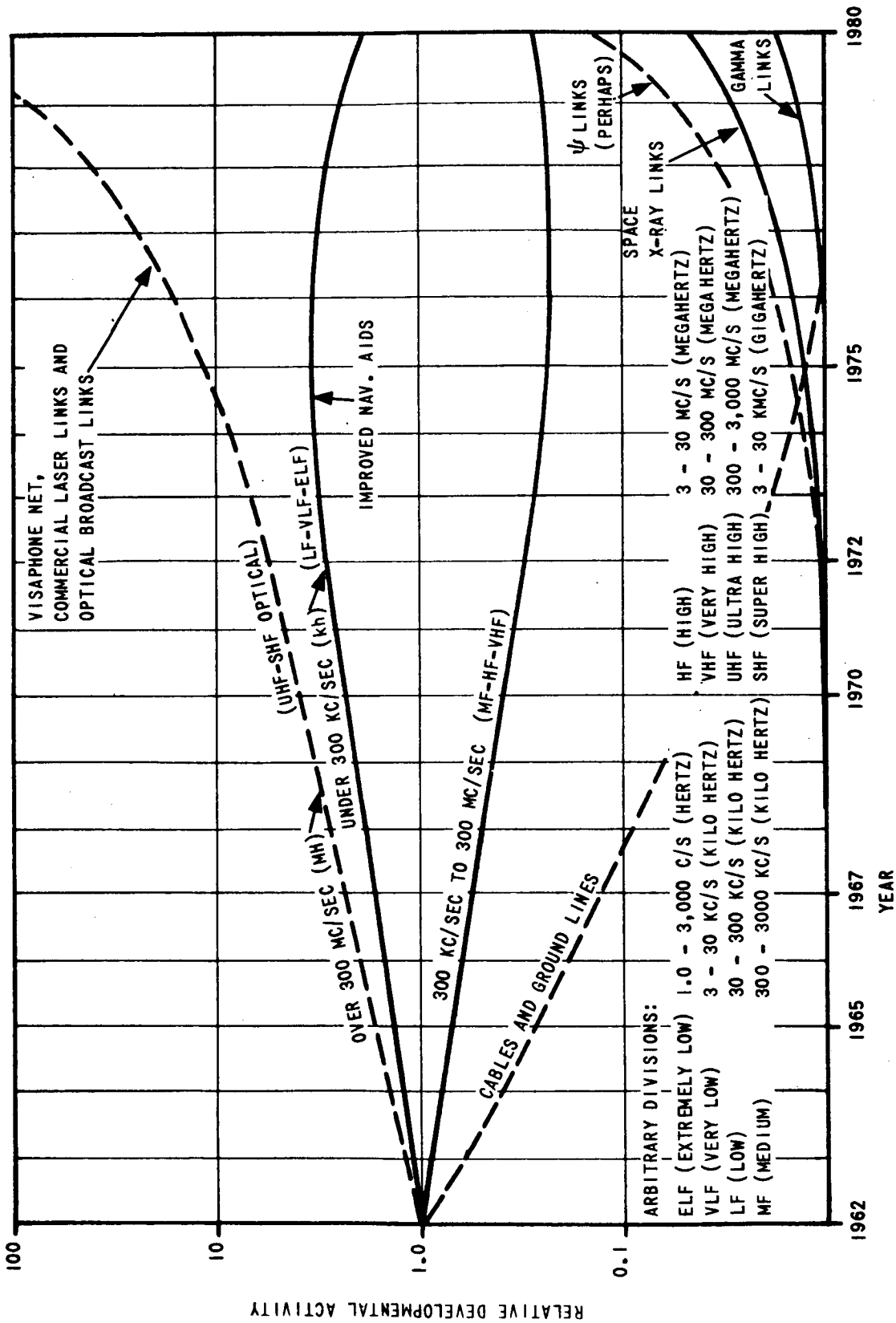


FIGURE 10-3. RELATIVE DEVELOPMENT WORK EXPECTED IN COMMUNICATIONS SPECTRUM (REF. 1962 LEVEL)
(Sources: Ref. 5 and Benfield)

TABLE 10-1. MINIMUM THEORETICAL VALUES OF AVERAGE
TRANSMITTED POWER FOR GIVEN RANGE
AND REQUIRED CHANNEL CAPACITY

$A_r = 10^4$ meters², $G_t = 1.00$, $N_o = 2 \times 10^{-14}$ watts/megahertz, $L = 10$ db. **

Bit Rate
(Minimum required values of average transmitted power*)

Range	10 bits/sec	10^3 bits/sec	10^6 bits /sec
5×10^4 kilometers (distant Earth satellite)	4×10^{-6} watt	4×10^{-4} watt	0.4 watt
5×10^5 kilometers (lunar distance)	4×10^{-4} watt	4×10^{-2} watt	40 watts
10^7 kilometers (near asteroid)	0.2 watt	20 watts	2×10^4 watt
10^8 kilometers (Mars at close approach)	20 watts	2×10^3 watt	2×10^6 watt

*Note: Power input for a practical system is 2 to 10 times these
ideal values.

** A_r Receiver antenna area
 G_t Transmitter gain factor
 N_o Noise factor
 L Loss factor

Figure 10-4 shows the probable range of a 1,000-watt transmitter (with a 26 db directional antenna) as a function of a bit rate.

Figure 10-5 forecasts expected improvement factor of communication range.

Figure 10-6 forecasts expected requirements and capabilities of communication systems for the near planets.

Figure 10-7 and Table 10-2 merely recount the possibly required links and some of the problem areas.

10.1 LF/MF TRANSCEIVER

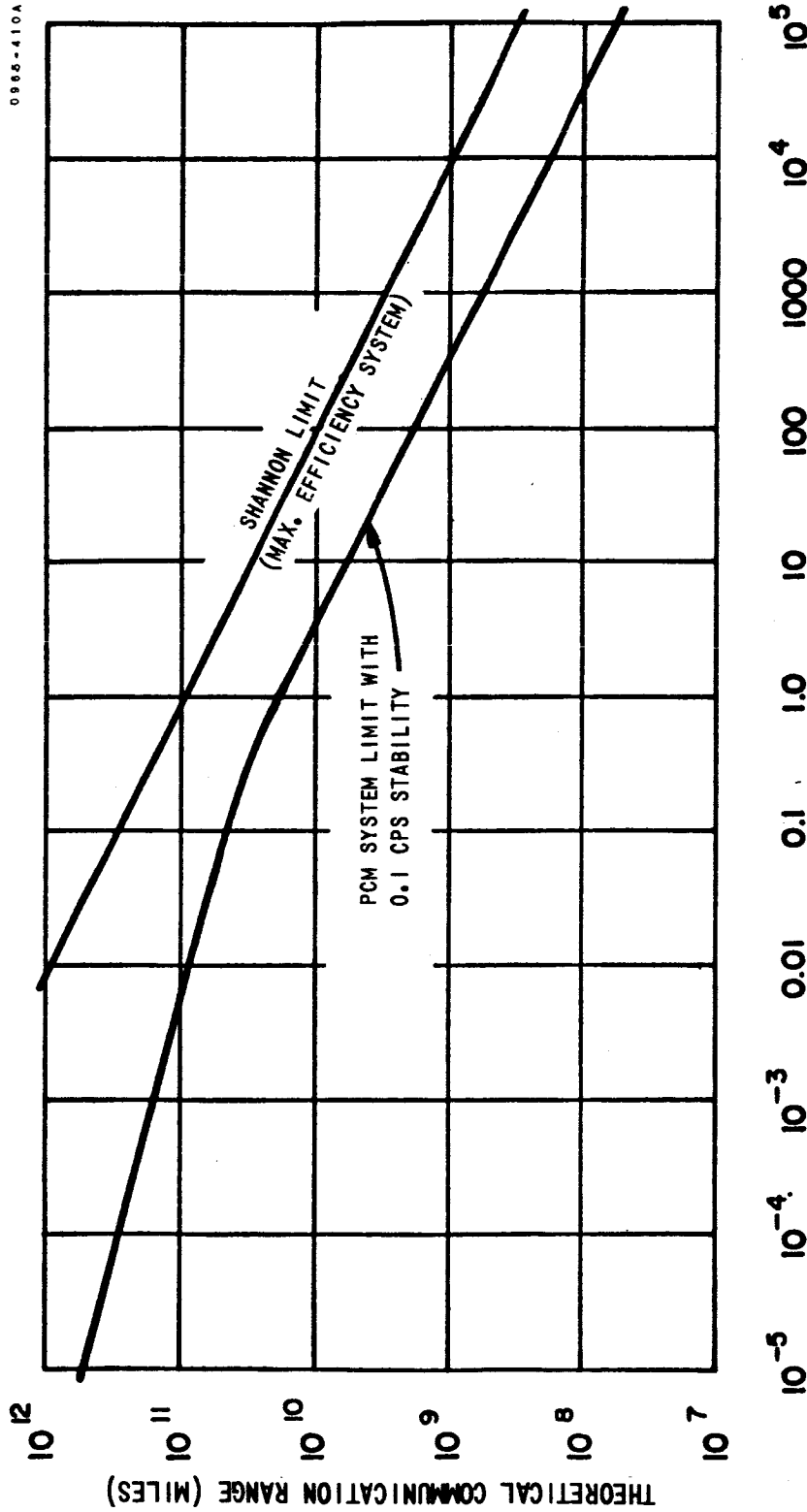
(Low frequency/medium frequency transceiver - 30 kh_z to 31 mh_z)

(Quoting from Reference 30) Lunar surface communication beyond line-of-sight is complicated by the moon's shorter radius compared to Earth's, the lack of an appreciable ionosphere, and weight limitations on equipment. With repeater relays, microwave or other line-of-sight frequencies could be used; however, this type of system rapidly becomes complex, and the technique has basic limitations (Figure 10-8(A)).

Alternatively, Earth stations could be used as active repeaters for point-to-point communications on the Moon's surface. This method doesn't lend itself to a simple, economical system and isn't much simpler than the line-of-sight repeater approach. A third method would use low and medium frequencies for surface-wave propagation, which is well within the state of the art.

10.1.1 Over The Horizon Propagation

It is desirable to operate at lower frequencies because, for a given radiated power, the communication range increases with a decrease in frequency. However, this must be balanced against allowable antenna height as restricted by payload weight and mobility on the lunar surface. The range obtained at the optimum frequency as a function of antenna length is shown in Figure 10-8(E). Because of the characteristic shown in Figure 10-8(E), range can always be extended for a fixed antenna length by operating at a frequency below the resonant frequency of the antenna. (End of quote from Ref. 30).



MAXIMUM INFORMATION RATE (BITS PER SEC.)

XMTR POWER = 1000 WATTS
 RCVR NOISE TEMP = 100° K
 XMTR ANTENNA GAIN = 26 db
 RCVR ANTENNA APERTURE = 300 M²

FIGURE 10-4. ILLUSTRATIVE, THEORETICAL RANGE VERSUS
 BIT-RATE FOR A MODEL SYSTEM
 (Source: Ref. 1 & 4)

0965-405A

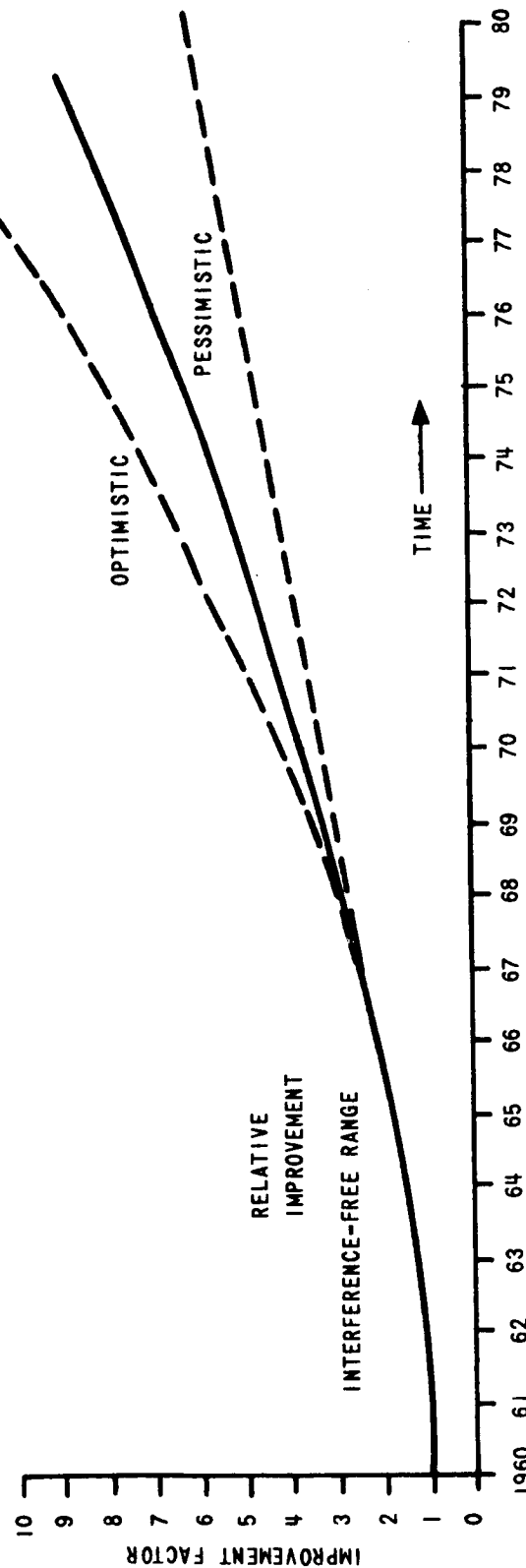


FIGURE 10-5. ESTIMATED RELATIVE IMPROVEMENT IN
EFFECTIVE COMMUNICATIONS RANGE

(Source: Ref. 5, Extrapolated)

Internally generated noise causes a serious limitation in distance of radio communication. Our entry into space has focused attention on this problem since we now have requirements for communicating over millions of miles. Solid state masers, non-linear reactance amplifiers and tunnel diodes are used in an effort to obtain the required gain and bandwidth with little internally generated noise in the amplifier output. In contrast to an equivalent noise figure of about 1,000°K for recently produced receivers, 290°K is obtainable without cooling, in reactance amplifiers at 5,000 megahertz. Considerable improvement in semiconductor diodes and quantum mechanic tunneling is required to provide low-noise amplifying devices without the complication and weight of masers and non-linear reactance amplifiers. Receiver noise figures in the order of 20°K should be obtainable (Ref. 5).

0965-419A

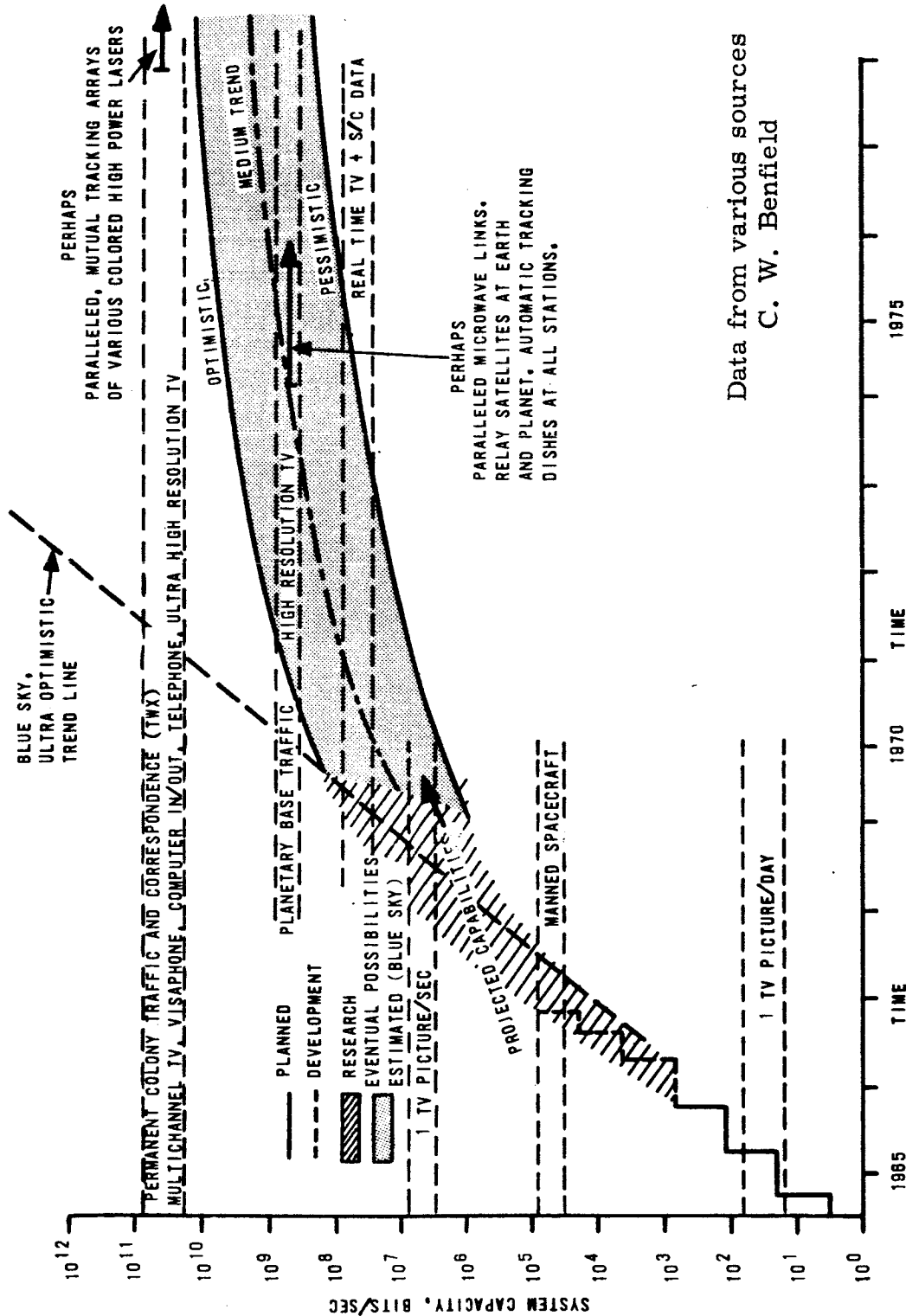


FIGURE 10-6. PROJECTED COMMUNICATION REQUIREMENTS FOR NEAR PLANETS

(Unlike signal-propagation time) message time, or data rate, can be improved by technological advances. Data rate appears as bits/second for a typical planetary distance of 240-million miles, with mission capabilities represented by various bit rates.

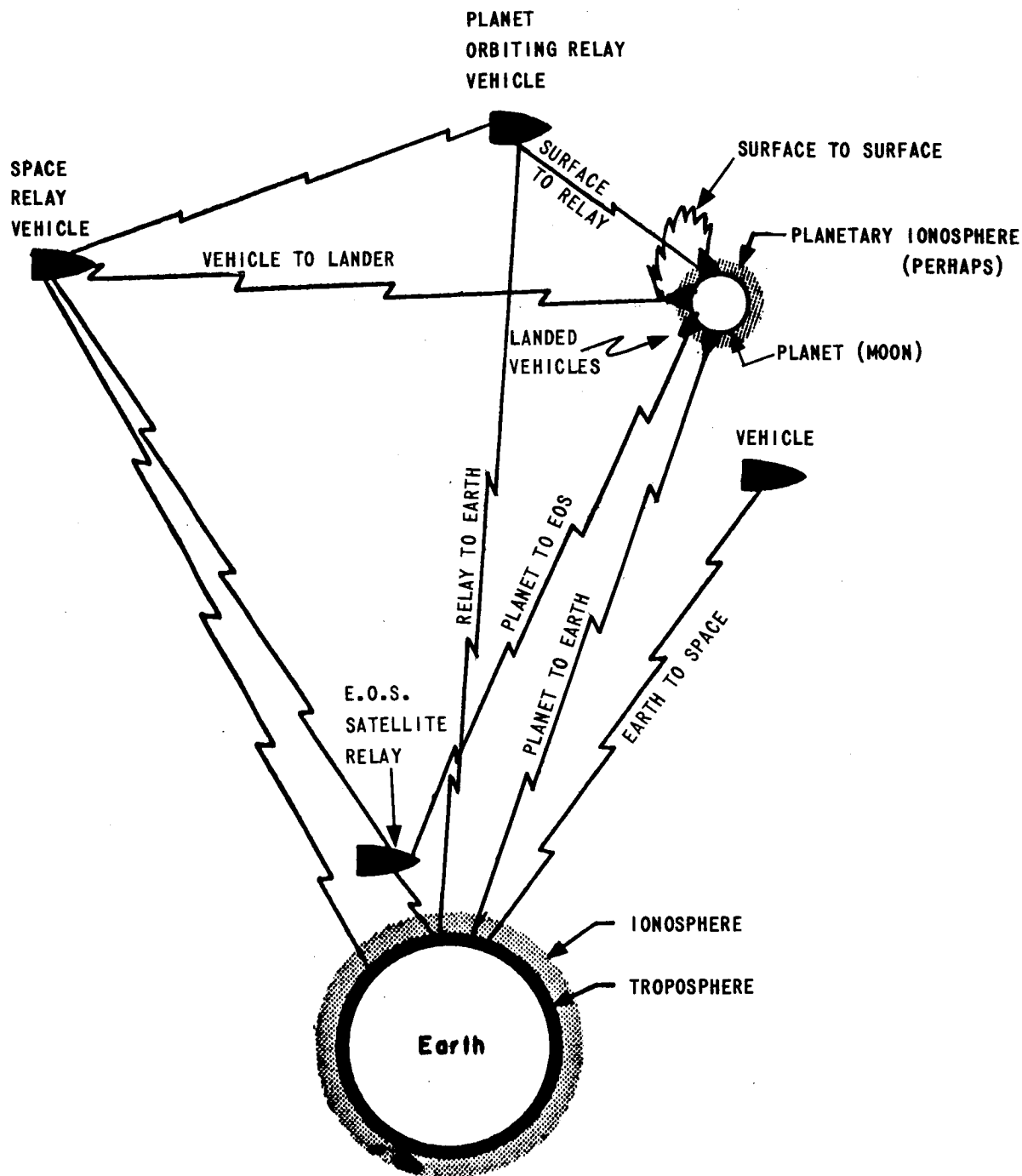


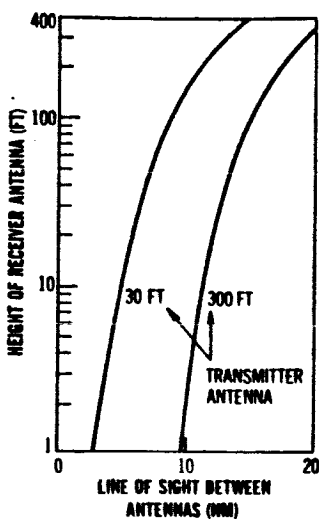
FIGURE 10-7. POSSIBLE SPACE COMMUNICATION LINKS
(Modified from Ref. 1 and 4)

TABLE 10-2. COMMUNICATIONS SUBSYSTEM SUMMARY TABLE

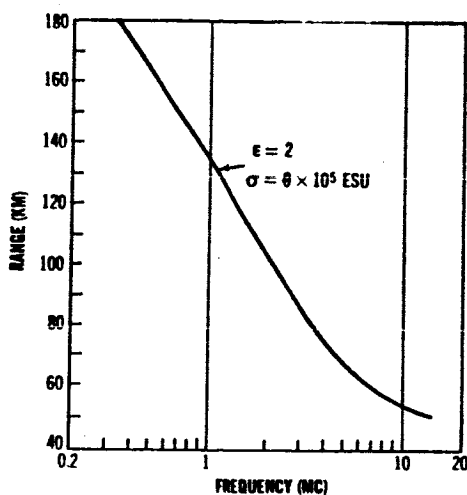
First Terminal	Second Terminal	Problems and System Limitations
(1) Earth	Earth	VLF & LF links require high power, depend primarily upon ground wave, large antennas, small bandwidths. MF & VHF links require judicious choice of times when ionospheric reflections permit sky wave paths. UHF, SHF & EHF or optical are limited to line of sight unless scatter can be made "visible" to the receiver. Medium to high bandwidths, small antennas, some atmospheric absorption. Tower to tower links are excellent. Cable links are expensive and require several years lead time although many feasible links exist. Bandwidth usually limited to coaxial cable's high frequency cut-off. Relay satellites are permitting long-distance, wide-band links and such systems are expected to improve rapidly.
(2) Earth	Orbiting Satellite of the Earth	VLF & LF generally unusable due to ionospheric shielding. MF & VHF occasionally usable when tropospheric ducting is applicable. UHF, SHF & EHF are usable, but due to line-of-sight limitations, satellite must be above horizon. Tracking directional antennas are used to improve receiver and transmitter "gain". Optical links are limited by cloud cover, atmospheric refraction, scattering, and reflection, but with sufficient peak power (> 10 megawatts) these limitations tend to become minor. Doppler shifts must be compensated.
(3) Orbiting Satellite of the Earth	Earth Receiver	VHF & UHF usually used. Satellite usually has omnidirectional antenna, but may be oriented to local vertical (by attitude control) and use a hemispherical pattern antenna or restricted, conical major lobe. Satellite transmitter is usually power limited, which requires high gain antennas at Earth terminal. (Az-El tracking antennas give best results; Satellite must be above horizon.)
(4) Earth	Space Probe	Conditions similar to (3) but power and directivity at Earth terminal must be adequate for greatest range to be used. This usually requires extremely large antennas with precision tracking in right ascension and declination at Earth locations. Probe must be above horizon of tracking station, which is a function of elevation angle and attitude of the probe (See Figure 3.6.1-3). Three DSIF stations can handle probes in the ecliptic plane - one at a time - but many probes aloft at once may require either undesirable time sharing or construction of new DSIF facilities.
(5) Earth	Vehicle on Moon or Moon Base (on near side)	Conditions similar to (4) above. Possibility exists for deploying large antennas on lunar surface or vacuum plating crater walls to provide semi-directional "dishes" facing toward Earth. Eventual installation of extensive antenna "farms" on the moon may permit use of lower frequencies for direct broadcast to Earth receivers (such as "TV from the moon").
(6) Moon Base #1 (near side) or Lunar Roving Vehicle	Moon Base #2 (near side)	Probable lack of lunar ionosphere requires either high power VLF ground wave system, UHF line-of-sight (tower-to-tower for long distance), or moon-Earth-to-moon relay or use of a lunar satellite relay (VHF & UHF or optical). Cables may be strung (eventually) between fixed bases. Electron guns may be used to "orbit" a curved beam of modulated current at low altitude and directed along a great circle between bases to achieve circumlunar communications, such as between observatory on far side and relay station on near side (feasibility not yet proved). A high "launch" tower is required for the "gun".
(7) Moon Base	Orbiting Satellite (of Earth's Moon)	Satellite must be over horizon. Directional tracking antennas preferred unless nuclear power plant is available for ground transmitter. Doppler shift requires compensation. Optical links probably will be highly effective when precise tracking can be accomplished.

TABLE 10-2. COMMUNICATIONS SUBSYSTEM SUMMARY TABLE
(Continued)

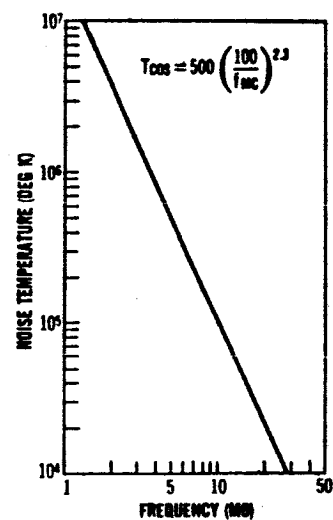
First Terminal	Second Terminal	Problems and System Limitations
(8) Moon Base on Opposite Side from Earth	Earth	Relay Stations are essential. These may be (1) a circumlunar cable to near side base, (2) a recording lunar satellite for readout on near side, (3) a tower-to-tower circumlunar link to near side base, (4) a network of lunar satellite relays phased 120 degrees in longitude (and at highest stable altitude) illuminated by a directive tracking antenna at far-side base and readout by tracking antennas on Earth, or (5) perhaps the orbiting electron beam link "launched" from a high tower to a high tower electrode at the near-side base (feasibility of this idea is still debatable).
(9) Earth	Lunar Satellite	Conditions similar to (4) above. Satellite is obscured by Moon during nearly half of each orbit except when its orbit plane is parallel to local horizon(s) of Earth station(s). If continuous communication is required, a network of phased satellites may be used as mutual relays in such a way that at least one is always visible to each other one and to the Earth tracking antennas.
(10) Earth	Planetary Satellite	While planet is not near opposition or conjunction, no unusual solar noise problem arises due to antenna illumination by the direct Sun. Problem is similar to (9) and (4) above except for distance which implies higher power and directivity and a time delay due to two-way transit at velocity of light. Planet can be eclipsed by Sun or Moon and satellite is eclipsed by planet during part of its orbit except under limited, special phasing conditions. Planetary satellite is assumed to be power limited, so directive antenna is desirable; this implies requirement for a steerable antenna, a spatial attitude reference (star trackers), an Earth tracker and sufficient precision of tracking to bring the Earth into the antenna's major lobe. Mariner, Ranger and Voyager studies apply. Bandwidth is severely limited unless power is in multimegawatt region.
(11) Lunar Roving Vehicle	Earth	Earth tracking directive antenna can be under Earth-terminal's remote control, if an omnidirectional acquisition antenna is provided on the LRV; if the LRV is manned, initial acquisition can be "eye balled" in. If Earth is below LRV's horizon, some relay link is necessary (if contact is to continue) and the problem is similar to (8) above.
(12) Planetary Roving Vehicle or Colony	Earth	Conditions are similar to (11) above but suffer from much greater range and time lag. Rotation of planet regularly obscures a single planetary base even with total Earth longitude coverage (three or more DSIF stations). Remedy is to provide one for a set of three or more longitudinally phased relay satellite(s) around planet to act as relays. Multiple-doppler, refraction, and atmospheric absorption problems exist. Bandwidth is severely limited until nuclear power and large directive transmitters can be installed on planetary surface. Intermittent schedules required when minimal systems are used.
(13) Spacecraft or Space Station	Spacecraft	Communications limited to line-of-sight, unless relay vehicles are used. High powered, high bandwidth laser links will permit extreme ranges (> 10 ⁹ kilometers) if attitude control and directivity permit. Where much more than adequate laser beam power is available, diffusers can serve to relieve pointing requirements. UHF, SHF and occasionally VHF will be used in the near future (for MOL, etc.).
(14) Outer Planets	Earth	Extremes of power and directivity are required at both terminals. Time lags between call and reply extend to many hours. If SHF or optical links are to be used, a relay station in Earth orbit or (preferably) on the Moon is desirable to permit conversion to UHF before entering Earth's atmosphere. Bandwidth severely limited (unless power is extremely high) for radio. High power and directivity with beam spreading diffusers may be required to permit use of wide band laser links. Eventual X or gamma laser links may permit solving bandwidth/power problems. Early probes may record all data during planetary flyby, then readout during closer approach to Earth; however, this does not solve the command and control problem and would imply delegating all control to an onboard programmer.
(15) Re-entering Spacecraft	Earth	Spacecraft is "blacked out" by a shield of plasma surrounding it. Recent Gemini system tests indicate that considerable progress has been made in overcoming the problem. (Details are not available.)



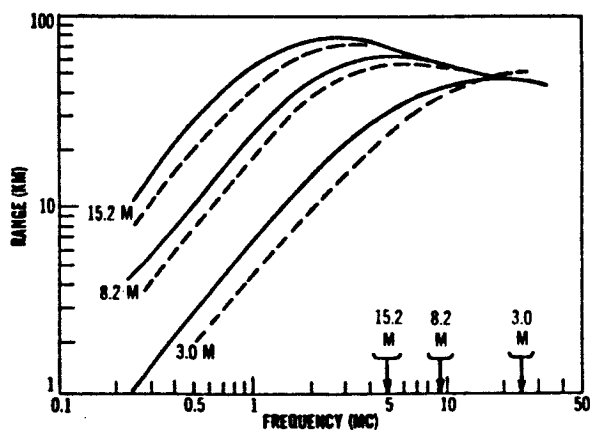
(A) Range limitations of line-of-sight communications systems as related to the heights of the transmitting and receiving antennas.



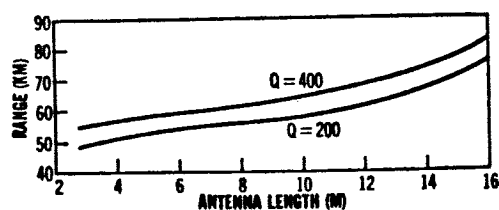
(B) Range of lunar surface communications vs frequency for ground-wave communications, considering field strength, lunar-surface conductivity (σ) and dielectric constant (ϵ), and cosmic and receiver noise factors.



(C) COSMIC noise vs frequency, based on calculations from theory.



(D) Lunar range vs frequency for ground-wave communication with short vertical whips and one-kilowatt input. When used as quarter-wave antennas, these whips have resonant frequencies as indicated by arrows at bottom of graph. Tuning-network Q is 200 for the dashed curves and 400 for the solid, the dielectric constant of the moon surface is 2, surface conductivity is 9×10^5 ESU, and antenna radius is 0.25 in.



(E) Lunar range for an optimum frequency as a function of antenna length.

FIGURE 10-8. LUNAR SURFACE COMMUNICATION
(Source: Ref. 30)

However, it appears that no reliable system exists for circumlunar communication. A possible concept involves orbiting a modulated beam of slow electrons around a great circle path between towers. This system would have to be developed experimentally in the actual lunar environment. There is a good possibility of ground wave transmission beyond line of sight on the Moon using the LF/MF frequency regime.

A VHF/UHF transceiver can be used for line of sight (surface) and for communication with lunar satellites and Earth terminals.

10.1.2 Reliability, LF

Since communication is range dependent, the reliability of contact at extremes of the allowable range is marginal. Multipath phasing may produce blind spots as the roving vehicle moves. In any case, ranges much beyond 100 kilometers present real feasibility problems.

10.2 VHF/UHF TRANSCEIVER (30 MEGAHERTZ TO 3 GIGAHERTZ)

Various data rates of 10^3 to 10^8 bits/second will require high bandwidth capabilities in some concepts. This makes VHF/UHF, or even SHF (above 3 gigahertz), operation necessary. Multichannel voice and digital data links are required at ranges from a few kilometers to a few hundred million kilometers.

10.2.1 Low Cost

In concept (EIO1) the desired meteorological rockets will be needed in great numbers and are totally expendable. This requires that all on-board equipment be made as cheaply as possible to provide reliable communications only during the few minutes of flight and to survive several months of preflight storage. For this application the transceiver production cost should be less than \$10 each.

10.2.2 Standby Reliability

In concept EIO1 this really means storage reliability for several months before flight.

In other concepts, this term refers to partially powered operation, whereby the transceiver is capable of acting upon command. This really requires long periods of continuous receiver operation and represents a receiver reliability problem.

10.2.3 Antenna Directivity

In general, large antenna dishes are required for narrow beam (high gain) antennas. Some small vehicles require high gain antennas of minimum practical weight and size.

Multiple antenna arrays can be made to behave somewhat like a very large single antenna. The cost of such arrays exceeds that of a single dish up to about 66 meters diameter, after which the multiple arrays are cheaper (see Figure 10-9). In any event, the array approach appears less desirable for space vehicles, except where non-mechanical antenna steering is desired.

Certain low-noise areas of the Earth should be selected for such multiple antenna farms and feasibility design studies probably should begin on the assumption that a planetary base communications link will be needed during the middle 1980s.

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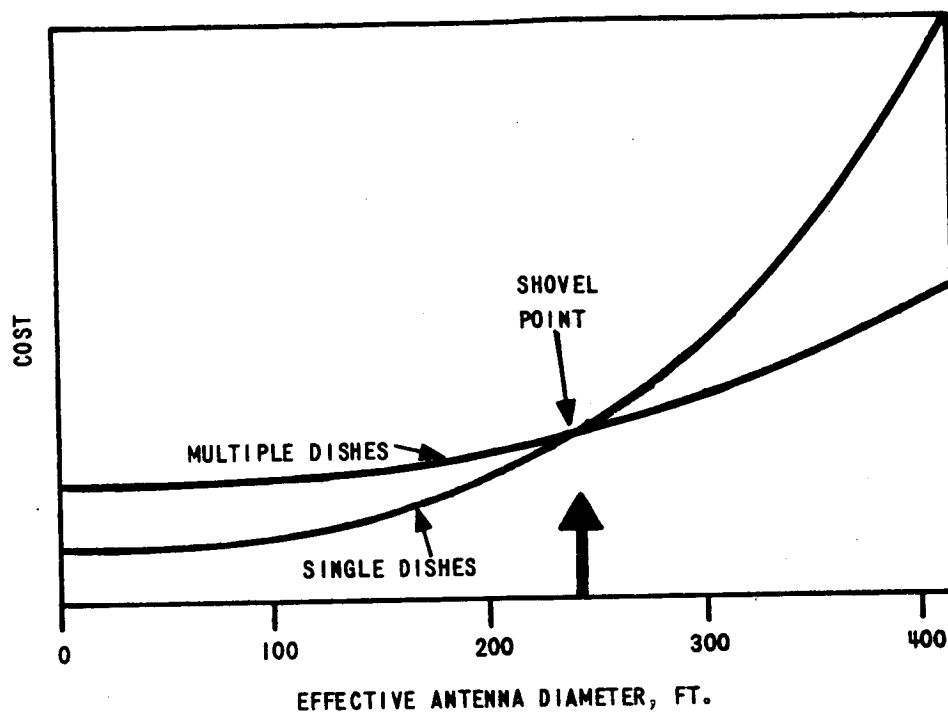


FIGURE 10-9. ONE VERSUS MULTIPLE ANTENNA COSTS

When deployed petal arrays are used, accuracy of the paraboloidal surface is difficult to maintain, hence directivity suffers by comparison to a rigid dish. Deployed petals have been considered for Mars landers, however.

When satellites are attitude stabilized by gravity gradient torques, the disturbances caused by electromechanical antenna steering or lobing would be intolerable. In such cases, phased arrays may be used and electronic steering of the main lobe accomplished. The definitions of steering and directivity tend to become confused in regard to such arrays.

10.2.4 Power

In general, space communication systems will be designed to use the lowest feasible transmitter power at the vehicle to reduce overall weight. At planetary ranges, power and bit rate are intimately related (see Table 10-1). Usually, it is bit rate that suffers as a result of necessarily reduced power. Whatever power is available, the bit rate would be increased to the point where the system is again a loaded channel, or just feasible. For example, if the Mars lander transmitter power could be increased to 2 megawatts isotropic, a use would be found for the 10^6 bits per second allowable data rate, and antenna directivity would again be invoked to increase effective radiated power (ERP) and bandwidth.

10.2.5 Coding (Communications Efficiency)

As explained in paragraph 10.1.1, the bit rate is limited to the bandwidth if a single-symbol, two-state code is employed. By designing the system for pulse amplitude coding, the digital bit rate could be increased toward the noise defined Shannon limit. But, amplitude discrimination would be required and any coding which tried to force the data rate beyond the noise limit would fail. Such optimum coding methods can easily be jammed by a slight noise increase.

Research into coding methods is needed.

10.2.6 Reliability Antenna Steering

Antenna steering can be accomplished by the direct method of pointing and training the dish and vernier control by off focus positioning of the radiating element (called lobing).

Alternatively, steering within limits can be accomplished by varying the phase of feeds to various radiators in a phased array to alter the radiation pattern of the array.

Direct, mechanical pointing involves electromechanical servomechanisms with consequent low reliability and heavy power drains.

Phased array pointing or steering will require R&D for each application to permit design of an adequate array and the required phase shifter drives.

Synchronous communication satellites stabilized by gravity gradients particularly require phased array steering in order to avoid the disturbing torques required in mechanical steering (these would completely swamp the gradient stabilization system).

10.2.7 Transmitter Efficiency

Considerably more power must be supplied to a transmitter than is radiated from the antenna. The difference must be radiated away as heat energy and presents a "waste heat rejection" or "cooling" problem.

Figure 10-10 forecasts the expected improvement in transmitter efficiency.

The effective radiated power (ERP) of a transmitter often is many times the input power. This merely means that the ERP within the narrow, focused beam or main lobe is equivalent to that from a non-directional antenna radiating at the stated ERP level. In paragraph 10.0.3 this gain factor appears in the range equation in such a way that an antenna gain through directivity is effectively the same as increasing radiated power by the gain factor (but only in the direction of the main lobe).

10.2.8 Multiple Access

In such services as telephone relay satellites and navigation advisory satellites, it is desirable that the station treat each new customer or user as though his traffic was the only one being handled, whenever he calls.

The telephone dialing system with multiple line selection is analogous to this. A finite number of simultaneous channels would be set up, and each calling user would be connected to some open channel, until all channels were in use, whereupon a busy signal would be returned to the next caller, and he would try again later.

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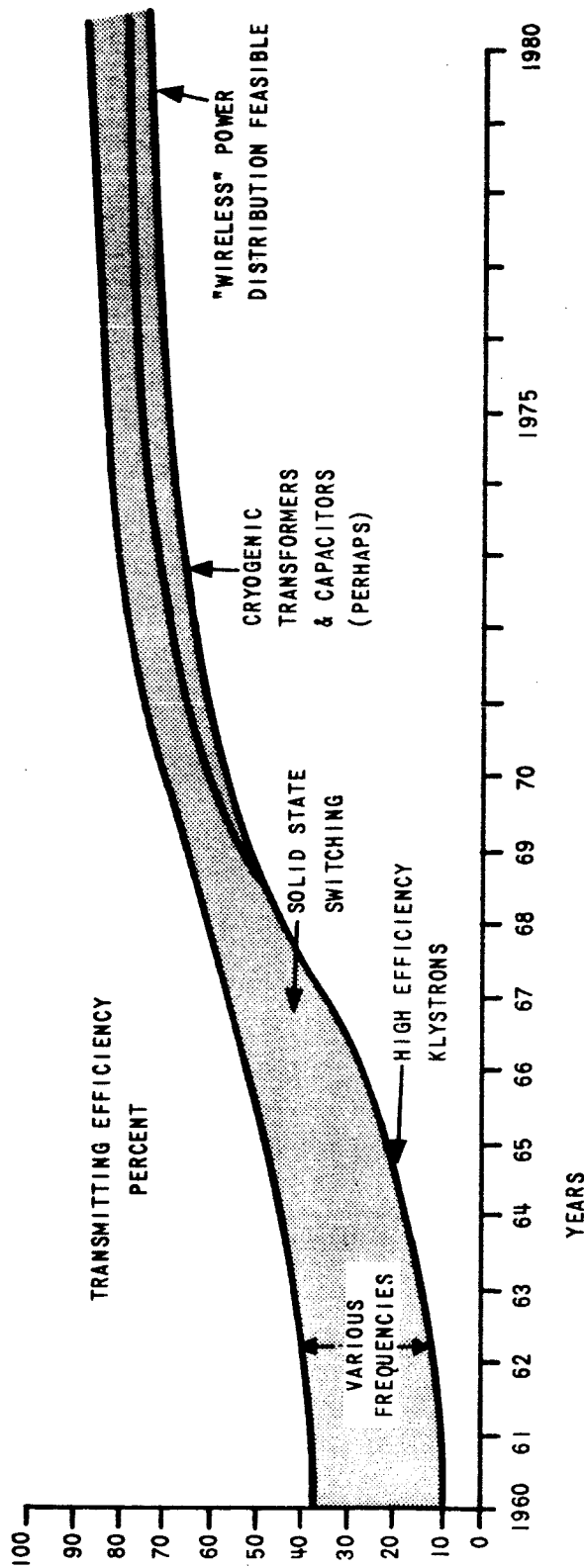


FIGURE 10-10. TRANSMITTER EFFICIENCY FORECAST

Significant improvement in the power efficiency of radio transmitters is required over the next decade. Although our rocket development will soon permit much higher payloads, the demands for electrical energy in spacecraft and manned aircraft will continue to increase at least as rapidly. Improvements in solid state devices, such as tunnel diodes, and application of them to radio transmitters, will be required to raise efficiencies to the 60 - 70 percent range by 1970. (Source: Ref. 5)

Since the users are all likely to call on the same frequency, an interference problem is likely.

Also, it is desirable for the satellite's antenna to be directed toward the user. This will require the user to monitor several channels on which his reply might come. Several different beams at different frequencies are possible with the phased array, but use of different beams at the same frequency probably presents an impractical requirement.

R&D is required for multiple access and multiple usage of the phased array antenna.

10.2.9 Reliability

Whenever unattended operation is critical to mission success, the designer faces a hardware reliability problem. Redundancy of functional elements permits considerable improvement, but also involves a payload penalty and adds the failure rate of the alternative selection process to that of the basic equipment.

The general trend is toward ruggedized microelectronics wherever possible. Jet Propulsion Laboratories has apparently adopted a policy of using the simplest conceivable specialized hardware to just do the job. In some instances the mission plan is altered to permit use of simpler equipment. Simplicity and reliability are almost synonymous.

10.3 UHF TRANSCEIVER (300 MHz TO 3 GHz)

10.3.1 Transmitter Efficiency

See paragraph 10.2.7.

10.3.2 Antenna Directivity (Also see paragraph 10.2.3)

Where both high directivity and low power, size and weight are involved, as in planetary probes, the additional requirement for precision pointing usually is found. This requires, in turn, a pointing reference and navigation system (to permit open loop pointing) or an active tracking antenna with a cooperative transponder at the other terminal.

10.3.3 Auto Erection and Alignment (Antenna Deployment)

High gain antennas usually are physically large. When these must be transported within a small vehicle, an electromechanical antenna deployment mechanism is required. Failure of this mechanism could cause catastrophic loss of data due to reduced ERP (or no power at all).

Refer to paragraph 10.2.3. Precision alignment of the flexible, folding arrays presents a problem.

*Includes squib actuated pyrotechnics, etc.

10.4 UHF/SHF TRANSCEIVER (300 MHz TO BEYOND 30 GHz OPERATION)

Wherever extreme data rates and consequently high bandwidths are required, it is absolutely necessary to increase carrier frequency to the lowest feasible channel capable of modulation at the stated bandwidth. The required antenna size for a given directivity decreases, or conversely antenna gain for a given size increases as frequency increases. Figure 10-11 shows variation of parameters with frequency normalized to decibels versus that parameter at one gigacycle.

10.4.1 Signal/Noise Ratio

$$\text{Signal-to-noise ratio} = \frac{\text{average signal power}}{\text{average noise power}} = \frac{S_r}{N_o W}$$

(Quoting from Reference 1)

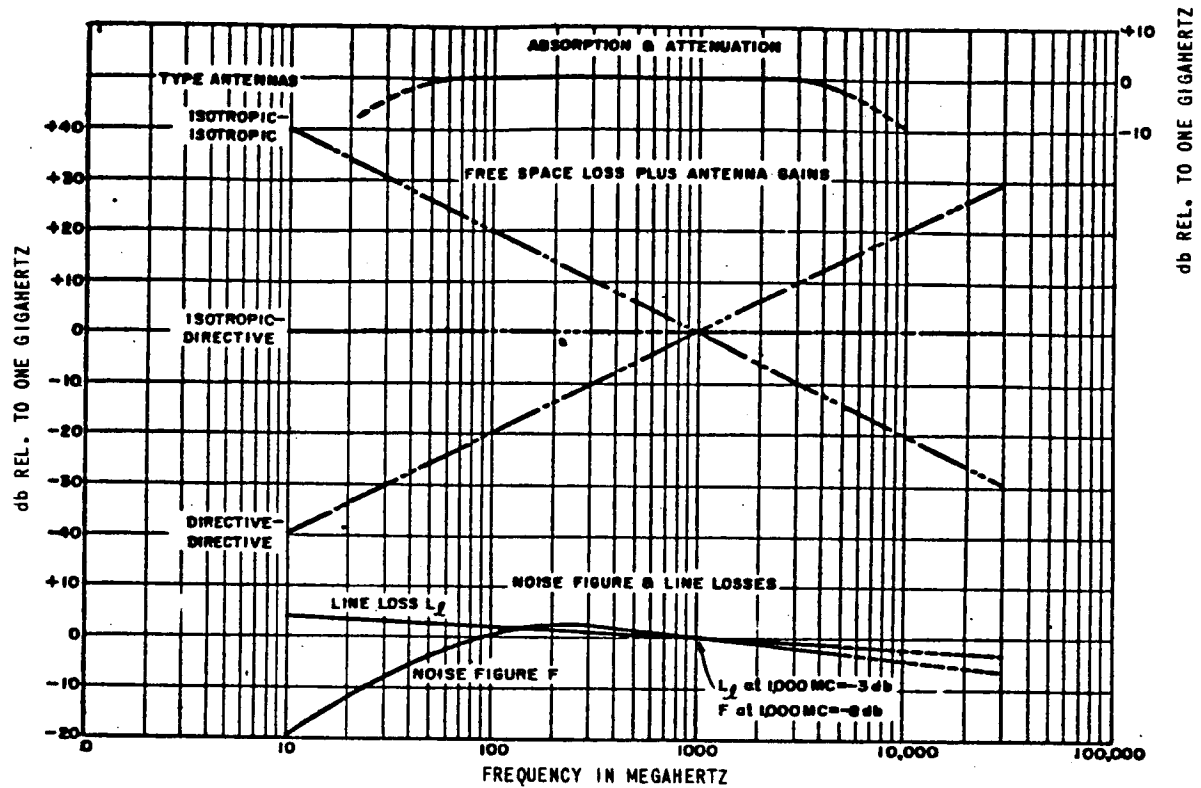
"Let N_o represent the noise power density (average noise power per unit bandwidth) in the neighborhood of the center frequency of the band. For many types of system noise, if the channel bandwidth W is small compared with the carrier frequency, one can regard the total average noise power as being proportional to W :

$$\text{Total average noise power} \approx N_o W$$

Types of system noise are:

- Internal receiver noise (thermal noise).
- Solar and cosmic noise. (The Sun and other extraterrestrial sources are emitters of noise-like radiation in the frequency bands of interest to communication.)
- Some types of man-made interference such as noise barrage jamming.

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NOTE: PLOT OF PARAMETER VARIATIONS FACILITATES DETERMINATION OF OPTIMUM FREQUENCY TO BE USED. VALUES ARE REFERRED TO ONE GIGAHERTZ; USE THE FOLLOWING FACTORS FOR CORRECTION FOR OTHER FREQUENCIES:

GROUP	ANTENNA SYSTEM	DISTANCE ONLY	POWER ONLY
I	ISOTROPIC-ISOTROPIC	MULTIPLIED BY $1000/f_{MH}$	MULTIPLIED BY $(f_{MH})^2/1000$
II	DIRECTIVE-ISOTROPIC	NO	NO
III	DIRECTIVE-DIRECTIVE	MULTIPLIED BY $f_{MH}/1000$	MULTIPLIED BY $1000^2/f_{MH}$

FIGURE 10-11. COMMUNICATIONS PARAMETER VARIATIONS
(Source: Ref. 1)

Typical microwave receivers (operating at room temperature)

$$N_o = 10^{-14} \text{ to } 10^{-13} \text{ watt/megacycle (internal noise)}$$

Maser amplifier

$$N_o = 10^{-16} \text{ to } 10^{-15} \text{ watt/megacycle (internal noise)}$$

Estimates of cosmic and solar noise radiation must at present be based on extrapolations from values measured at the Earth's surface, although this situation may soon be rectified. The following example is based upon current estimates for an omnidirectional receiver in the 250 megacycle region:

Solar noise (at Earth's orbit)

$$N_o = 4 \times 10^{-16} \text{ watt/megacycle}$$

Cosmic Noise

$$N_o = 2 \times 10^{-15} \text{ watt/megacycle}$$

We see that for present-day receivers internal noise is more significant than solar or cosmic noise; however, for masers or other advanced detectors, solar and cosmic noise may be more significant than internal noise.

It is seen that bandwidth W enters the expression $W \log_2 (1 + \frac{S_r}{N_o W})$ in two places, with opposite influence on ultimate attainable channel capacity.

The net effect is that $W \log_2 (1 + \frac{S_r}{N_o W})$ increases as W increases. This

increase is fairly rapid if $\frac{S_r}{N_o W} \gg 1$, but it becomes insignificant for

$\frac{S_r}{N_o W} \ll 1$. This illustrates that one cannot gain channel capacity indefinitely

by increasing bandwidth if the total average signal power is held fixed. "

(End quote from Ref. 1)

The minimum usable carrier-to-total noise power ratio (sometimes called the threshold) will determine whether or not the input signal can be detected in the presence of noise associated with the receiver input, including the antenna system.

In the frequency ranges considered feasible for space communications, the thermal noise of the receiver predominates at isolated locations where the terminals are likely to be.

Figure 10-12 shows variation of $\frac{S_r}{N_o W}$ as a function of carrier frequency for various antenna combinations.

10.4.2 Doppler Compensation

Due to the relative velocities of vehicle and some other terminal the received signal will be shifted in frequency:

$$\text{Received frequency} \cong (\text{transmitted frequency}) - \frac{(\text{transmitted frequency})(\text{relative velocity})}{(\text{velocity of light})}$$

The doppler shift,

$$\Delta f \cong - \frac{f_o V}{C} \quad (\text{First order approximation})$$

Since velocity will be changing, Δf will vary. Fixed tuned AM receivers would experience fading and FM receivers would experience distortion. Nominal, zero-modulation carrier frequency must be adjusted at the receiver on the basis of the known relative velocity to avoid distortion. Automatic frequency control systems and phase lock loops for this work are usually state of the art, but add to cost and complexity of receiving equipment.

10.4.3 Antenna Directivity (See paragraph 10.2.3)

This refers to electronic steering of phased arrays in some concepts and the general problem of attaining high gain.

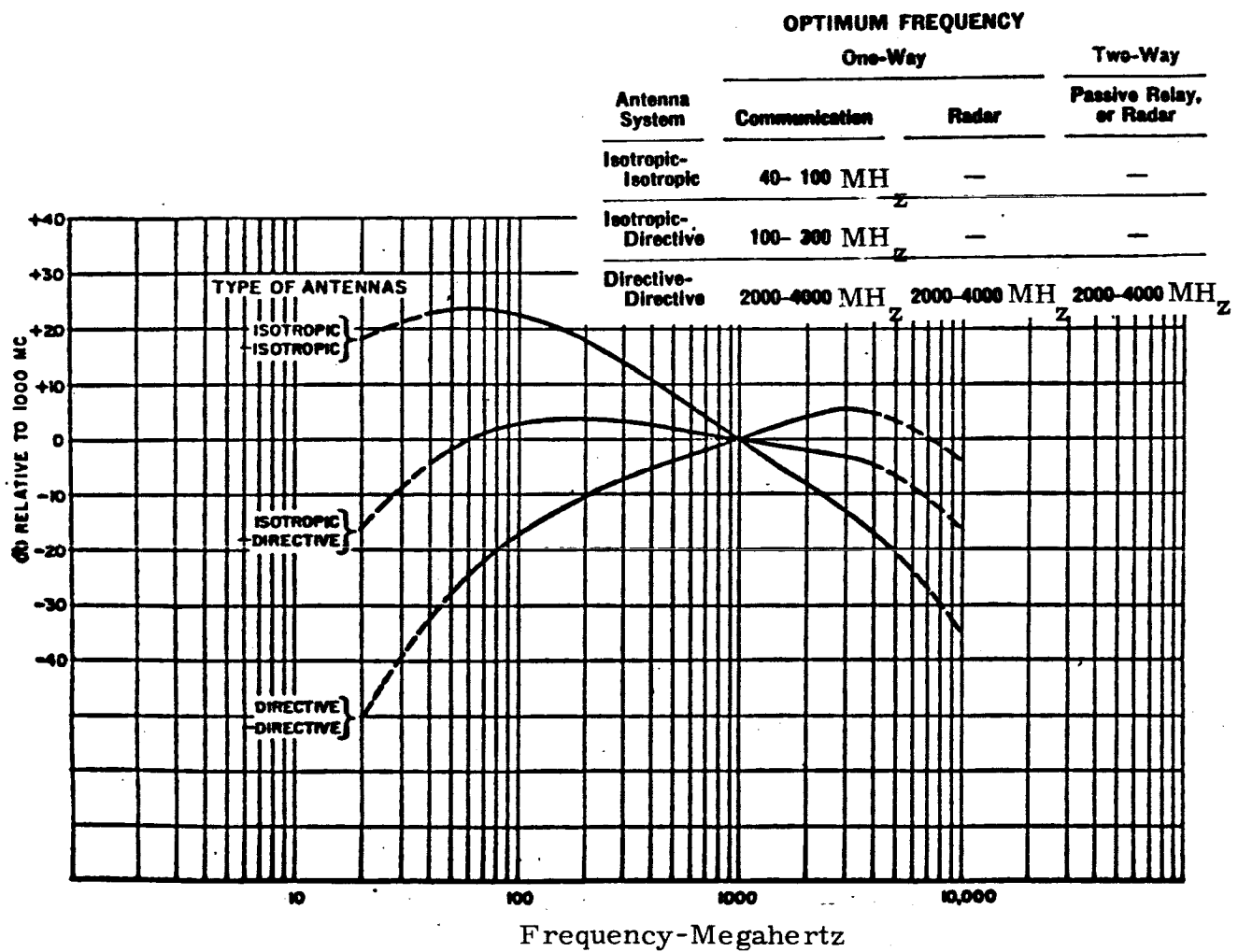


FIGURE 10-12. PLOT OF CARRIER-TO-NOISE RATIO
VERSUS FREQUENCY
(Source: Ref. 7)

10.4.4 Transmitter Efficiency (See paragraph 10.2.7)

At the SHF end of the spectrum, solid state amplifiers are in the early development stage. Present transmitters usually rely upon Klystrons and travelling wave tubes. These devices require cathode heater power which does not contribute to antenna power.

It used to be the custom to ignore heater power in calculating RF amplifier efficiency. For this reason, efficiencies of vacuum tube amplifiers often sounded better than they really were. When comparing vacuum tube and solid state amplifiers, heater power and driving power must be considered for a realistic comparison of system capability.

10.4.5 Antenna Gain

Essentially the same as antenna directivity. See paragraph 10.2.3 and paragraph 10.2.7 (Transmitter Efficiency).

10.4.6 Multiple Access

See paragraph 10.2.8.

10.4.7 Reliability

See paragraph 10.2.9.

10.4.8 Antenna Pointing

Except for phased array electronic beam steering, antenna pointing is essentially the same term as antenna steering (see paragraph 10.2.6).

10.4.9 Transmitting Power

See paragraph 10.2.4.

10.4.10 Antenna Tracking

In some cases refers to antenna steering (paragraph 10.2.6).

In certain navigation satellites an interferometer antenna has been proposed as a means for tracking ground stations. The concept is feasible in principle, but may not be sufficiently precise for the navigation functions. Phased array electronic steering is the inverse of this type of tracking. Hypothetically a phased array at each terminal, constantly tracking the other terminal

and steering its transmitter beam toward that terminal could provide a maximum gain system. This might be called mutual, interferometer tracking.

10.4.11 Solar and Cosmic Noise (See paragraph 10.4.1)

A highly directive antenna would have difficulty tracking a Venus probe as it crossed the apparent solar disk. The receiver would be overwhelmed by solar noise. Similarly, the command receiver of a Mars probe would have difficulty in receiving the DSIF signals during inferior conjunction. Mission planning must take such factors into account. Figures 10-13, 10-14 and 10-15 illustrate this consideration.

Since the RF noise sources in the sky have not been adequately plotted, it is to be expected that an occasional communications interruption will occur due to having one of these aligned with the transmitter as viewed from the receiver. Loss of mutual tracking may occur in such an event, and signals to the low gain antennas will be used to re-acquire the receiver's attention.

10.4.12 Power Requirement

See also paragraph 10.2.4 and Table 10-1.

At planetary ranges the bandwidth is severely restricted by lack of transmitter power. In some specialized communication satellites the power for communications becomes the major load on the auxiliary power subsystem, particularly when wide beam antennas are used for wide area coverage.

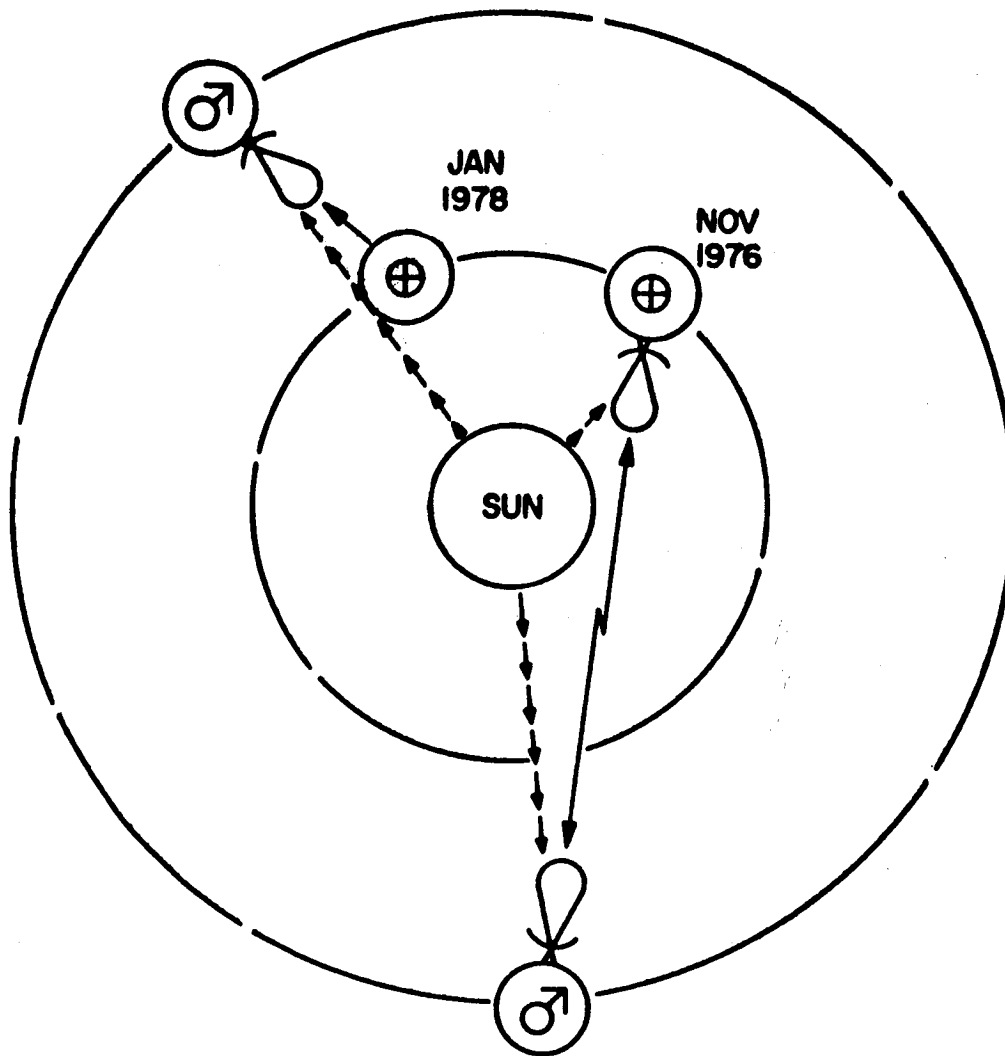


FIGURE 10-13. SUN-EARTH-MARS GEOMETRY TO
ASCERTAIN RF INTERFERENCE BY THE SUN
(Source: Ref. 26)

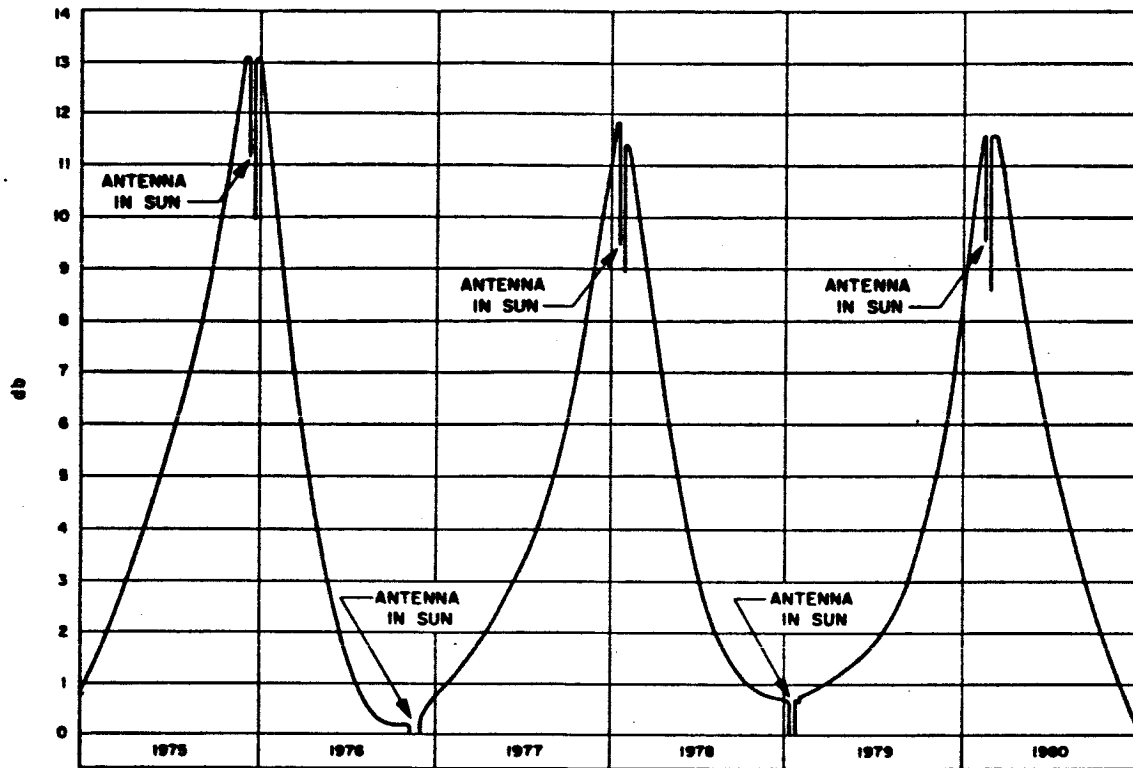


FIGURE 10-14. RELATIVE COMMUNICATION CAPABILITY FROM EARTH TO MARS (REF. 26)

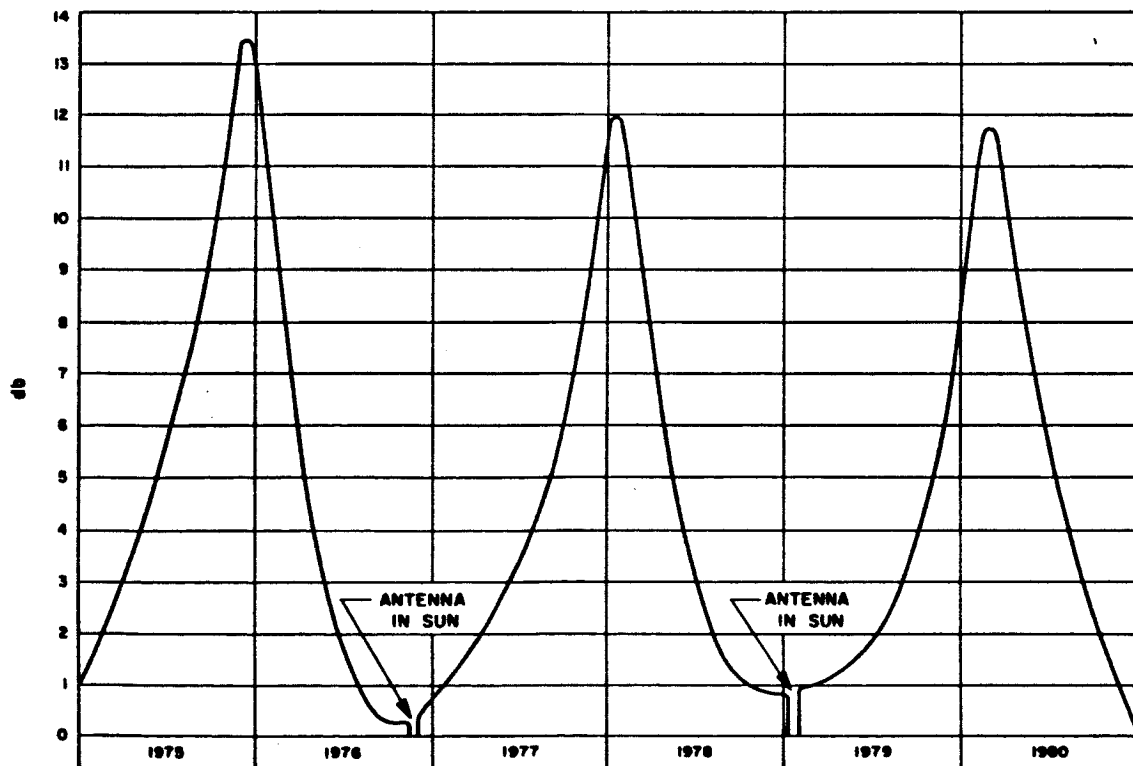


FIGURE 10-15. RELATIVE COMMUNICATION CAPABILITY FROM MARS TO EARTH (REF. 26)

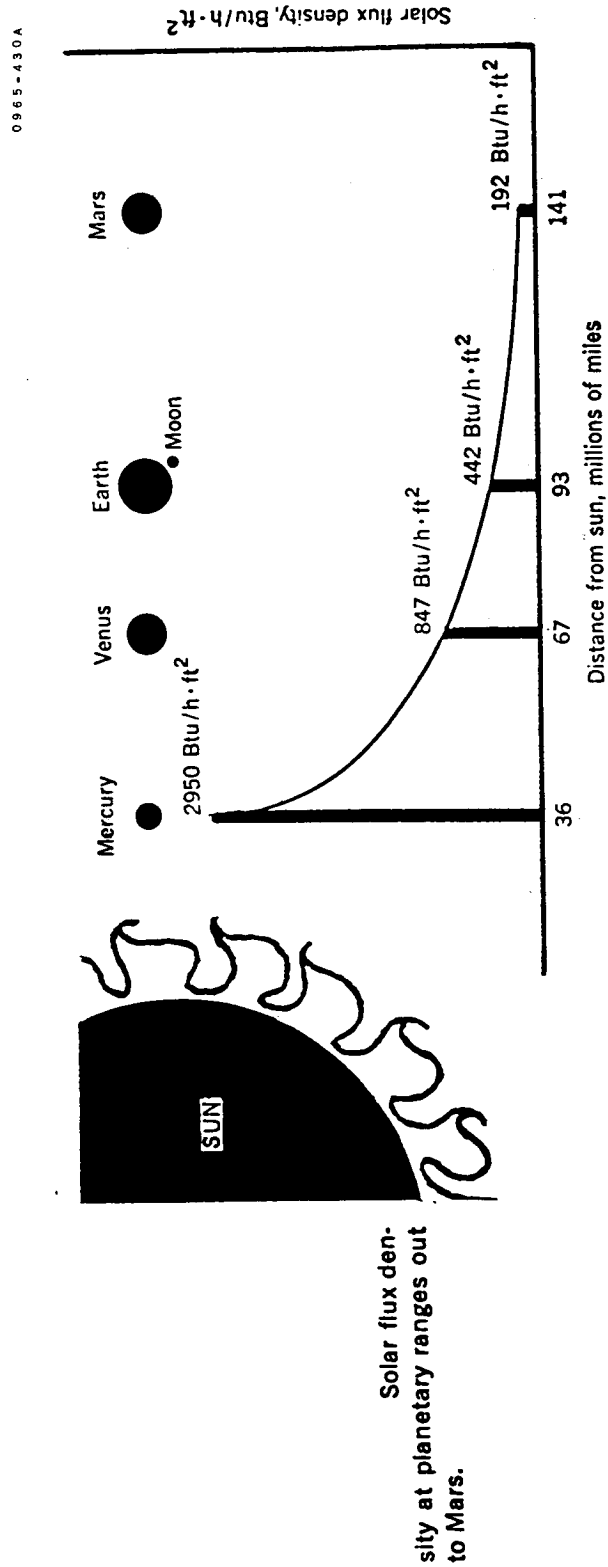


FIGURE 10-16. SOLAR FLUX DENSITY AT PLANETARY RANGES OUT TO MARS (REF. 28)

10.5 SHF TRANSCEIVER

(3 to 30 GHz_Z)

10.5.1 Reliability

(See Paragraph 10.2.9.)

10.5.2 Antenna Directivity

(See Paragraph 10.2.3.)

10.5.3 Antenna Tracking (See Paragraphs 10.4.10, 10.2.6)

The tracking problem will become particularly critical in situations where an orbiter or fly-by is required to track an atmospheric probe or lander during entry.

10.5.4 Solar and Cosmic Noise (See Paragraph 10.4.11)

10.5.5 Interference with Observations

In radioastronomy satellites, the inevitable rear lobes from the transmitting antenna will couple RF energy into the radioastronomy sensors. In some cases reflection and back scatter from the atmosphere will cause similar interference with the RF observations. In severe cases a time sharing system might be required.

The size of antennas may on occasions present an obstacle to experiment package optics.

10.5.6 Antenna Pointing

See Antenna steering, reliability (Paragraph 10.2.6).

10.5.7 Thermal and Radiation Protection

With the sun illuminating part of a paraboloid antenna, thermal distortion can occur which affects beam directivity and alignment. Alignment is especially critical at SHF.

Raising the temperature of receiving elements will increase the receiver's thermal noise with consequent deterioration in performance (by raising the threshold). Receiver inputs should be kept as cool as possible, particularly in the SHF bands. Masers often are used as receiver input amplifiers, and sometimes cooling to cryogenic temperatures is desirable for lowest possible threshold.

Radiation damage to semiconductors can manifest as noise and can even result in failure of the device.

10.6 SHF TRANSMITTER

(Beyond 3 gigacycles.)

Photographic satellites, such as meteorological satellites, astronomical telescopes and resources survey satellites will require extremely high bit rates, since their primary function is gathering of pictorial type data, preferably with high resolution scans and color. Usually, this information must be stored until the vehicle is within sight of a ground station. It then will have a few minutes to dump the entire contents of its tape recorder to the ground station. This accounts for the high data rates and for choice of SHF.

10.6.1 Wide Band Modulation

Bandwidths of 20 percent of carrier frequency would be quite feasible, if adequate modulation methods were available. With 20 percent modulation of a 3 gigahertz (3×10^9 cps) carrier, the bandwidth would be 6×10^8 cycles per second, with a 30 gigahertz carrier, this could be 6×10^9 cps. Wide band, low distortion modulation methods are needed.

10.6.2 Antenna Steering

(See Paragraph 10.2.6.)

10.6.3 Transmitter Efficiency

(See Paragraph 10.2.7.)

10.6.4 Comm Efficiency (Coding)

(See "Coding" paragraph 10.2.5.)

The extremely high data rates make it desirable to apply optimum coding as well as maximum bandwidth modulation and high power. With a high signal to noise ratio and the maximum number of discernible amplitude levels per symbol, properly coded, the Shannon limit can be approached (but not reached). This could permit great reduction in the stated bandwidths, which were estimated on the basis of simple binary coding (< 1 bit/sec per cps). It may be necessary to use analog TV to permit taking best advantage of available bandwidth. This in turn requires a much more sophisticated on-board tape recorder than that for digital TV, but would permit eliminating the digitizer section of the video encoder.

10.6.5 Transmitter Reliability

In some applications the transmitter is required to operate continuously for one to five years. Few ground-based transmitters presently can operate unattended for more than a few months at high power levels. Transmitter design offers a reliability challenge in such applications as the satellite telescopes and meteorological satellites.

In manned vehicles, the transmitter can receive the occasional minor adjustments required and should approach the performance of modern broadcast and radar equipment, which seldom fails catastrophically, but rather requires minor attention at frequent intervals.

10.7 LASER TRANSCEIVER

The extremely high carrier frequency of coherent light offers the possibility for modulation bandwidths in the gigacycles even with a low percentage of modulation. Visible light lies in a narrow band between 10^{14} and 10^{15} cycles per second (see Figure 10-1).

Furthermore, the effective peak power within the laser's color or frequency can exceed the average power of the sun within that band, by many orders of magnitude. In addition to this, the wavelengths are so short that reflectors of small dimensions can permit beam widths in the arc-second range with consequently tremendous effective antenna gain.

10.7.1 Research and Development Required

Laser technology is yet in its infancy. The maximum average power output achieved from a CW laser to date is 40 watts. The maximum bandwidth of modulation was around 250 MHz. Transmitter efficiencies are less than 1 percent which largely offsets the directivity gains. However, it begins to appear that almost any material combination can be made to "lase." Efficiencies of gallium arsenide junction lasers have been estimated at around 16 percent and direct modulation is possible. With successful R&D, the next ten years could perhaps see operation of pulsed, solid state CW lasers having pulse repetition rates in the hundreds of megahertz, peak powers in the hundreds of megawatts, average powers in the thousands of watts and efficiencies of 20 percent or more. Amplitude modulation of such lasers might yield bandwidths of several gigahertz. Present lasers can carry all the world's radio and TV broadcasts simultaneously as a single modulation envelope. However efficiency is very low and powers are quite modest (1 or 2 watts).

Atmospheric absorption will limit laser communications on the earth to about 40 miles per relay link, but they are expected to begin augmenting existing microwave tower channels within a few years.

In space, the laser presents the attractive possibility for real time, multi-channel, high resolution, color, stereo TV between planets, if all of the technical problems can be overcome. The NASA hopes to test laser communications using the advanced technology satellite series as test beds.

Needless to say the extreme directivity of the laser requires previously unheard of pointing precision, attitude reference accuracy and vehicle stability.

10.7.2 Laser Tracking

Figure 10-17 shows the estimated pointing requirements as a function of future time. Laser systems require pointing accuracies better than 10^{-5} degrees.

Means will be needed for rotary scan or raster scan modulation to permit laser systems to acquire and automatically track a remote corner reflector or transponder. The problem is in the early study phases.

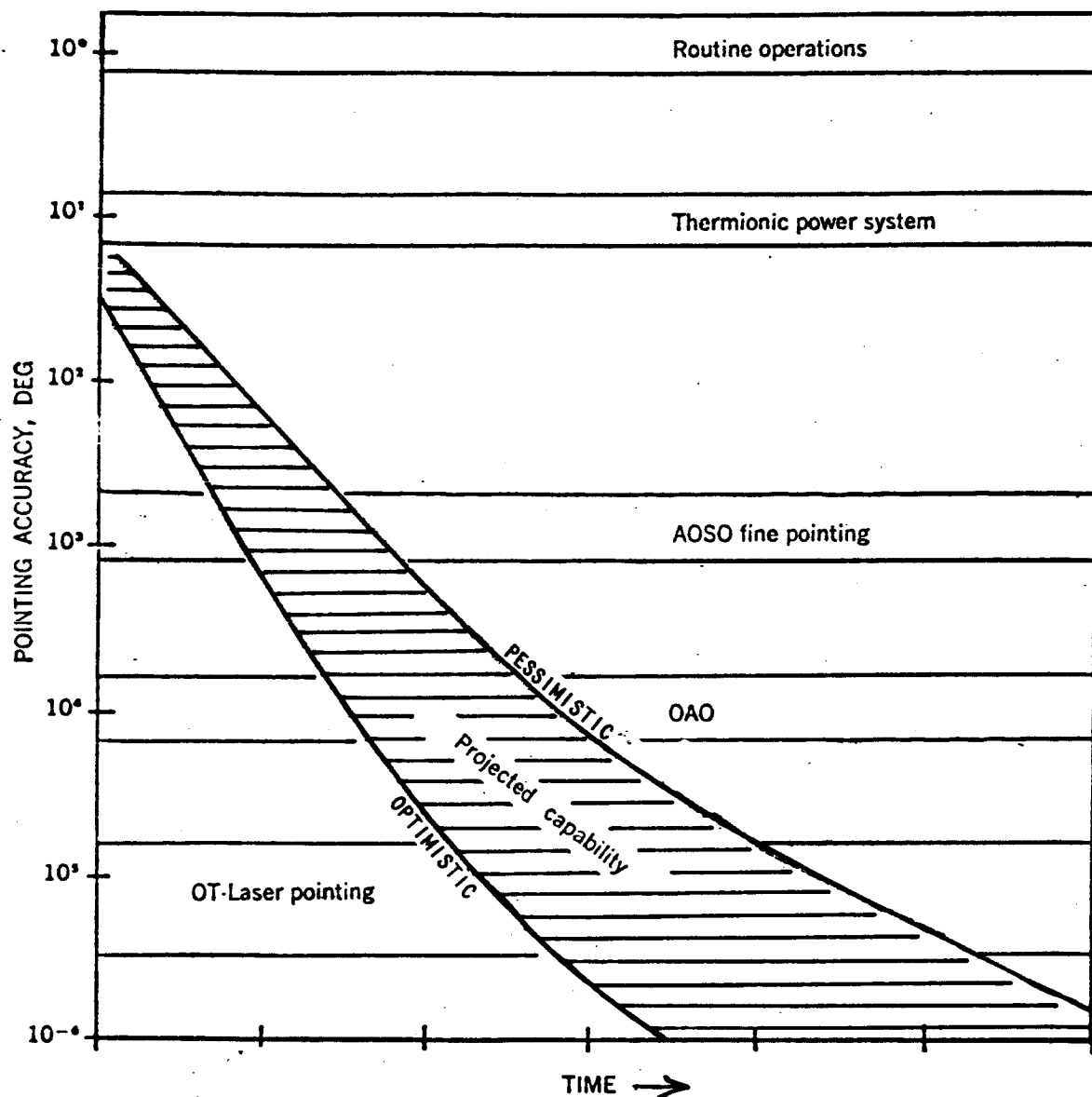


FIGURE 10-17. SENSOR POINTING REQUIREMENTS
(Published in several NASA Papers, e.g., Ref. 14)

10.7.3 Laser Acquisition

A laser near Venus, during a typical hypothetical mission, points back at the Earth. Its beam covers a circle about 400 miles in diameter on the Earth's surface. A receiver outside that circle would not be able to see the laser at all. Means must be developed for close, mutual tracking of laser terminal transceivers. Initial acquisition will require a scan procedure.

10.8 AIRBORNE OSCILLATOR/OSCILLATOR STABILITY

Present spaceborne oscillators can be maintained stable to better than 5 parts in 10^8 . Atomic frequency standards are better (for example, the cesium beam tube is stable to better than 2 parts in 10^{11}). But space versions of these atomic clocks are yet to be designed. The quartz clock used with the cesium beam tubes has a stability of 1 part in 10^8 . Unfortunately the cesium beam tube can operate at only one frequency (about 9.193 gigacycles). To gain an output at some other frequency with this stability requires complicated, digital countdown and multiply circuits capable of operating at gigacycle speeds.

10.9 INTERFEROMETER RECEIVER

In principle, an RF interferometer antenna array yields adequate phase information to permit determination of two angles designating the direction of a signal source from the plane and pole of the array. It has been suggested that this method be used to determine the positions of ships and aircraft from a satellite.

10.9.1 Research and Development Required, Interferometer Tech.

Present RF interferometer technology is marginal for navigation direction finding purposes. When the required array size for a suitable base line is established, physical instability of the array is very likely to be troublesome. The ultra precision automatic phase measuring circuits, and digital differential analyzers for coordinate transformations will require R&D.

10.10 VHF/FM TRANSCEIVER

FM program material could be broadcast simultaneously to 1/3 of the Earth from a synchronous satellite by providing a transceiver in the satellite to rebroadcast signals transmitted from Earth.

10. 10. 1 Antenna Deployment

(See also Paragraph 10. 3. 3.)

The large antenna required to provide 17 degree beam width may have to be folded to permit lofting it to synchronous orbit. Since it is to broadcast to rabbit ears on the ground, it must radiate high power (otherwise special, highly-directional ground antennas will be required to receive the broadcasts. This would reduce the propaganda effectiveness of the system).

Proper antenna erection will be required for mission success.

10.10. 2 Transmitter Life

This is the transmitter, unattended reliability problem (see Paragraph 10. 6. 5).

10. 10. 3 Transmitter Efficiency

(See Paragraphs 10. 4. 4, 10.2. 7.)

10.11 VHF TV TRANSCEIVER

The problem is similar to 10. 10 (FM transceiver), except that TV as well as FM sound is being handled. This requires much greater bandwidth and consequently much greater power.

10. 11. 1 Transmitter Life

Transmitter failure will be expensive, since it requires replacement of the entire satellite in synchronous orbit. The value of this tool for moulding world opinion will make it a mandatory expenditure, so replacement will be considered as part of operating cost. Every extra hour of service can represent a saving of perhaps several hundred thousand dollars. The transmitter should operate for over one year (preferably five years) with no maintenance.

10. 11. 2 Transmitter Efficiency

(See Paragraph 10. 4. 4, 10.2. 7.)

10.12 ANTENNA, PARABOLIC

A 30-foot dish is to be deployed in space at synchronous altitude to permit receiving low power ground or mobile stations.

10.12.1 Parabolic Antenna Size, Deployment

(See Paragraph 10.10.1.)

10.13 ANTENNA, PHASED ARRAY

To avoid physically pointing a dish antenna, (which would disturb the gravity gradient stabilization system) a phased array will have its main lobe directed by proper phase shifting among the array of individual radiators.

10.13.1 Antenna Gain, Phased Array

Can the array be made to yield a gain comparable to a steered dish?
Can the main lobe beam width be made very small, without severe I^2R losses occurring within the phase cancelled elements

10.13.2 Research and Development Required

The theory is not new but the technology is. Phased arrays of many elements are presently large and heavy, and development is required to attain configurations suitable for space applications.

10.13.3 Antenna Pointing

(See Paragraph 10.2.6.)

REFERENCE LIST FOR SECTION 10

10.1 P. Swerling, "Some Information-Theory Considerations in Space Communications", Rand Corp. Lecture No. 14 of "An Introduction to Astronautics", 1958.

10.2 Reference Data for Radio Engineers, Fourth Edition, 1956.

10.3 Kraus, "Proc. of IRE", Vol. 46, No. 1, 1958, p. 92.

- 10.4 C. M. Crain and R. T. Cabler, "Communication in Space Operations," Rand Corp. Lesson 15, No. P-1394 of An Introduction to Astronautics, 1958.
- 10.5 "Aerospace Technical Forecast, 1962-1972 and 1960-1970," Aerospace Industries Assn. of America, Inc., 1725 DeSales Street N. W., Washington 6, D. C.
- 10.6 John G. Lawton, "Adaptive Detection and Waveform Estimation," Cornell Aeronautical Lab Tech Quarterly (Research Trends) Spring/Summer, 1965.
- 10.7 Leang P. Yeh, "Communicating in Space," Part I, Electronic Industries, February 1959, page 54.
- 10.8 Frank Leary, "Communications of the Future," Electronics, July 1959, page 37.
- 10.9 R. W. Sanders, "The Limitations on Space Vehicles Imposed by Communications Requirements," ARS Paper 1667-61, March 1961.
- 10.10 George E. Mueller, "A Pragmatic Approach to Space Communications," Proceedings of the IRE, Vol. 48, No. 4, April 1960.
- 10.11 R. W. Sanders, "Communication Efficiency Comparison of Several Communication Systems," Proceedings of the IRE, Vol. 48, No. 4, April 1960.
- 10.12 R. W. Sanders, "Digilock Communication System," IRE Wescon Convention Record, Part 5, August 1960.
- 10.13 Gerald M. Truszynski, "Space Communications," NASA Bulletin recounting talk at 1963 annual meeting of the Air Traffic Control Association.
- 10.14 Albert J. Kelley, "NASA's New Electronics Research Center," Astronautics and Aeronautics, May 1965, page 58.
- 10.15 Richard P. Mathison, "Mariner Mars 1964 Telemetry and Command System," IEEE Spectrum, July 1965, page 76.
- 10.16 "System Capabilities and Development Schedule of the Deep Space Instrumentation Facility," Tech. Memo 33-83, Jet Propulsion Laboratory, Pasadena, California, 24 April 1964.

- 10.17 JPL Staff, "Mariner: Mission to Venus," New York: McGraw-Hill Book Co., Inc., 1963.
- 10.18 J. N. Bryden, "Mariner (Venus 162) Flight Telecommunication System," Tech. Report 32-377, JPL, 15 January 1963.
- 10.19 Space Programs Summary No. 37-23, Vol. II, JPL, 30 September 1963.
- 10.20 Space Programs Summary No. 37-24, Vol. VI, JPL, 31 December 1963, page 71.
- 10.21 B. D. Martin, "The Mariner Planetary Communication System Design," Tech. Report 32-85 (Rev. 1), JPL, 15 May 1961.
- 10.22 J. C. Springett, "Command Techniques for the Remote Control of Interplanetary Spacecraft," Tech. Report 32-314, JPL, 1 August 1962.
- 10.23 J. C. Springett, "Psuedo Random Coding for Bit and Work Synchronization of PSK Data Transmission Systems," presented at International Telemetry Conference, London, England, 1963.
- 10.24 R. P. Mathison, "Tracking Techniques for Interplanetary Spacecraft," Tech. Report 32-284, JPL, 1 August 1962.
- 10.25 R. P. Mathison, "Spacecraft Telecommunications System Design," 1963 Year Book, WGLR Conference, Munich, Germany.
- 10.26 W. K. Victor, et al, "Telecommunication Aspects of a Manned Mars Mission," JPL Tech. Report No. 32-501, 20 August 1963.
- 10.27 Charles C. Kirsten, "Data Acquisition at Planetary Ranges," NASA SP-11, 1-3 November 1962, page 319.
- 10.28 Frank Leary, "Television from Space," Space/Aeronautics, March 1964, page 70.
- 10.29 John D. Alber, and H. Ingram Richard, "Environmental Problems in External Space Equipment Design," IEEE Spectrum, July 1965, page 106.
- 10.30 Marc Chomet, et al, "Antennas for Lunar Surface Communication," Space/Aeronautics, 1964, page 101.

Section 11

VEHICLE STABILIZATION AND CONTROL

11.0 GENERAL COMMENT

The technological area of "Vehicle Stabilization and Control" for the purpose of this study includes all stability augmentation and control functions such as sensing, computation and control actuation to provide the desired response or attitude of the vehicle.

The following is quoted directly from Reference 1.

The two types of control exercises are: (1) turning the vehicle in a prescribed manner to a prescribed degree; and (2) stabilizing the heading or attitude in a desired direction despite the disturbing action of unwanted torques. Table 11-1 shows some of the functions required of vehicles and gives an indication of the accuracy with which they are to be performed. Outside of the operations a vehicle performs in becoming initially aligned, the maneuvering accuracy required of a vehicle can be as high as a minute of arc for certain astronomical applications. The requirement reduces to the order of degrees when solar arrays are to be turned toward the Sun. There is a greater accuracy spread in the stabilization required of vehicles: in measuring spectral characteristics of stars a vehicle may be called upon to hold its position to within 0.1 second of arc for over 2 hours; on the other hand, the vertical orientation or the heading of some vehicles or some of their parts may not have to be controlled closer than to within a few degrees. (1964 SOA - Ed.)

An abbreviated list of the missions calling for these operations is given in Table 11-2.

11.0.1 General Description of Attitude Control Systems

Figure 11-1 summarizes the structure of a vehicle attitude control system. The system is seen to consist of a vehicle, of an evaluation of its performance relative to a reference behavior, and of control. The evaluation portion of the system embraces sensors of vehicle behavior, a comparison with a reference behavior, and an interpretation of discrepancies in terms

TABLE 11-1. EXAMPLE ATTITUDE CONTROL FUNCTIONS AND
ACCURACY REQUIREMENTS (Reference 1)

<u>Initial Orientation</u>		<u>Attitude Change</u>		<u>Attitude Stabilization</u>	
<u>Operation</u>	<u>Accuracy</u>	<u>Purpose</u>	<u>Accuracy</u>	<u>Attitude</u>	<u>Accuracy</u>
Stop tumbling	0.005 deg/sec	Measurement	1 minute	Toward Sun	1 sec-several degree
Extend arrays		Solar power	Several deg	Toward star	0.1 sec
Align toward Sun	1/4-1 degree	Thrust vector alignment	0.3-1 deg	Toward vertical	0.1-several degree
Roll about sun-line	0.2 degree			Zero yaw angle	1-several degree
Acquire star	0.5 degree				
Acquire Earth	Several deg				
Align experiment package	Several deg				

TABLE 11-2. MISSIONS AND ORBITS OF SELECTED SATELLITES
(Reference 1)

Vehicle	Mission	Launch	Life (Months)	Attitude	Altitude (Nautical Miles)	Inclination (Degrees)	Weight (Pounds)
Syncom	Communications	1963	--	Inertial	19,000	33	73
Tiros	Meteorology	Apr 1960	3	Inertial	400	48	280
Nimbus	Meteorology	1963	6	Vertical	600	80.5	650
OSO I ¹	Scientific	Mar 1962	6	Solar	300	33	450
POGO ²	Scientific	1963	12	Vertical	200-600	33	950-1,500
EGO ³	Scientific	1963	12	Vertical	150-60,000	33	950-1,500
OA0 ⁴	Scientific	1964	12	Celestial	400	33	3,300

1 Orbiting Solar Observatory. 3 Eccentric Orbiting Geophysical Observatory.

2 Polar Orbiting Geophysical Observatory. 4 Orbiting Astronomical Observatory.

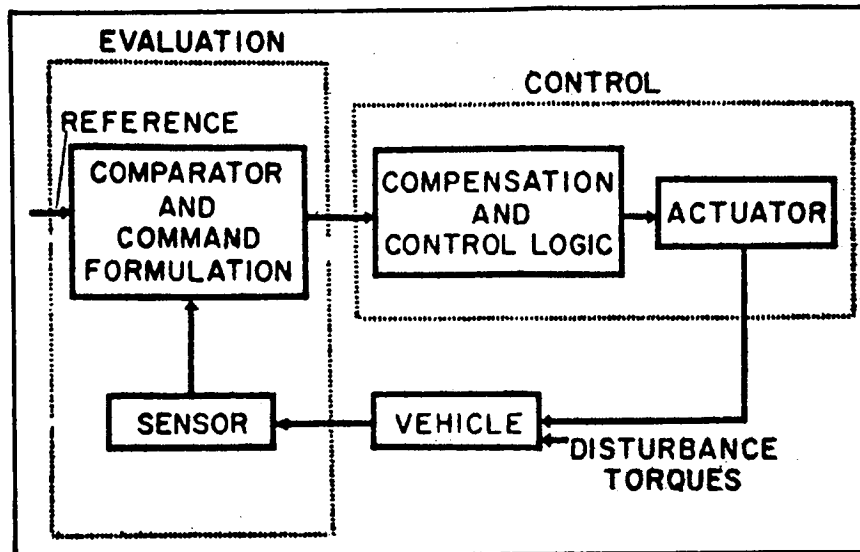


FIGURE 11-1. ATTITUDE CONTROL SYSTEM ELEMENTS
(Source: Ref. 1)

that are usable by the vehicle. The control portion of the system operates on this information formulating some law by which to govern the next behavior of the vehicle. (End of quote from Reference 1.)

Table 11-3 gives some rule-of-thumb values for the magnitudes of disturbing torques which vehicles may have to be stabilized against.

Figure 11-2 shows the wide variety of possible configuration choices for the elements of a stabilization and control subsystem. It was obviously impractical in this study to make a detailed configuration choice for each concept application.

Figure 11-3 shows three (typical) configurations proposed by Lockheed as alternatives to the hypergolic reaction jet systems. The characteristics of these are compared in Table 11-4.

Figure 11-4 shows an approximate subsystem choice area diagram for reaction control motor systems.

The behavior of reaction control subsystems usually is defined in terms of the "limit cycle". Figure 11-5 which is a plot of the attitude rate dead zone versus attitude dead zone. The vehicle drifts in attitude at not more than the minimum uncontrolled rate, until the attitude limit is reached whereupon a rate is commanded to reverse the drift.

11.0.2 In-Orbit Gyrocompassing

See Figures 11-6 and 11-7.

Many concepts require the vehicle to be stabilized to local vertical. This requires a horizon sensor to sense pitch and roll errors. A rate gyro senses roll rate and initiates a command to produce yaw rotation such that roll rate (due to the once per orbit rotation of the vehicle) is reduced to zero. This causes the pitch axis to point perpendicular to the orbit plane ($\pm$ about one degree). This method serves as a rough azimuth reference in orbit and is termed "gyrocompassing".

For space or "inertial" stabilization a pair of servo star trackers can be used to sense attitude errors within a few arc-seconds. Image dissector star trackers can sense errors as small as 0.01 arc-second.

TABLE 11-3. TYPICAL ATTITUDE DISTURBANCE TORQUES
(Reference 1)

	<u>Magnitude</u>
Vertically Oriented Vehicles	Torque in roll due to yaw angular velocity 50 dyne cm/deg/hr Torque in roll due to roll angle 200 dyne cm/deg Torque in pitch due to orbit eccentricity 200 dyne cm Torque in pitch or roll due to gravity gradient 500 dyne cm/deg
Inertially Oriented Vehicles	Torque due to gravity gradient 2,500 dyne cm Torque due to magnetic field 2,500 dyne cm

FUNCTIONAL
ELEMENTS
(LEVEL 8)

ALTERNATE
CONFIGURATIONS
(LEVEL 9)

ROTATION SENSORS

RATE GYROS ① ② ③
ATTITUDE GYROS (OR RATE INTEGRATING
GYROS ④ ⑤ ⑥ ⑩ ⑪ ⑫)
ESG's (GIMBALED) ④ THRU ⑫
ESG's (STRAPDOWN) ⑩ ⑪ ⑫
INERTIAL PLATFORM MEASURING UNITS (IPMU,
IMPU OR IMU) ④ ⑤ ⑥
STRAPDOWN INERTIAL MEASURING UNIT
(SIMU) ⑩ ⑪ ⑫
LASER ROTATION SENSORS ① ② ③
HORIZON SENSORS ④ ⑤
HORIZON SCANNERS ④ ⑤
STAR TRACKERS ⑦ ⑧
SUN TRACKERS ⑦ ⑧
VORTEX (FLUID) GYROS ① ② ③
ANGULAR ACCELEROMETERS ⑬ ⑭ ⑮
STAR FIELD RATE METERS ① ② ③
INTERFEROMETER ANTENNAS ④ ⑤
RADIO DIRECTION FINDERS ⑥
VISUAL HORIZON ④ ⑤
VISUAL DRIFT METER ⑥
GRAVITY GRADIENT SENSORS ④ ⑤
MAGNETIC COMPASS ⑥
FLUX GATE ⑥
GYROCOMPASS ⑥
VERTICAL GYRO ④ ⑤
CRYOGENIC GYRO ④ THRU ⑫
RADAR ④ ⑤ ⑥ (VS LANDMARK)
HOMING TRACKERS ④ ⑤
IR
RF
MAP
LIGHT
RADAR

MEASURED QUANTITIES

BODY RATES

PITCH RATE ① $\dot{\phi}$
ROLL RATE ② $\dot{\psi}$
YAW RATE ③ $\dot{\theta}$

BODY ATTITUDES

LOCAL VERTICAL

PITCH ④ ϕ
ROLL ⑤ ψ
YAW ⑥ θ

CELESTIAL (NOSF)

RIGHT ASCENSION ⑦
DECLINATION ⑧
ROLL ⑨

CARTESIAN

COS α ⑩
COS β ⑪
COS γ ⑫

BODY ACCELERATIONS

$\ddot{U}$ ⑬ $\ddot{X}$ ⑯
 $\ddot{V}$ ⑭ $\ddot{Y}$ ⑰
 $\ddot{W}$ ⑮ $\ddot{Z}$ ⑱

REFERENCES (AND CONTROLS)

VOLTAGE REGULATORS ① ③
ZENER DIODE
GLOW TUBE
STANDARD CELL BALANCE

VOLTAGE DIVIDERS ①
R/C/L NETWORKS
POTENTIOMETERS
STRAIN GAGE BRIDGES
THERMISTOR/VARISTOR NETS
STAIRCASE D/A CONVERTERS
VACUUM TUBE CIRCUITS
TRANSISTOR CIRCUITS
VARIABLE TRANSFORMERS
TAP SWITCHES

SYNCHROS ⑤
SYNCHRO GENERATORS
CONTROL TRANSFORMERS
SYNCHRO DIFFERENTIALS
MAGNESYN

SWITCHING CIRCUITS ② ③ ④ ⑤
SELECTOR SWITCHES
MANUAL KEYBOARDS
PUSH BUTTONS AND TOGGLES
PATCHCORD PANELS
A/D CONVERTERS
BUFFER REGISTERS
PUNCHED TAPE/CARDS
CODE DISKS

PRESSURE REGULATORS ①
AIR OPERATED RELAY
FREQUENCY REGULATORS
CRYSTAL OSCILLATORS
ATOMIC CLOCKS

SHAFT POSITION

- ① REFERENCE SUMMED WITH SENSED OUTPUT PRODUCES ERROR SIGNAL TO DRIVE SERVO ACTUATORS
- ② OCCURENCE OF A PROGRAMMED EVENT
- ③ MOTION OR VARIATION HAS REACHED A SET LIMIT
- ④ OCCURENCE OF AN EVENT ORDERED BY A DISCRETE SIGNAL
- ⑤ REFERENCE SHAFT ANGLE TO BE SUMMED ELECTRICALLY WITH OUTPUT ANGLE TO PRODUCE AN ERROR SIGNED

COMPARATORS

DIFFERENTIAL AMPLIFIERS
SUMMING AMPLIFIERS
SUMMING JUNCTIONS
ADDERS (SUBTRACTORS)
CONTROL TRANSFORMERS
SYNCHRO DIFFERENTIALS
MECHANICAL DIFFERENTIALS
BRIDGE CIRCUITS
VALVE SPOOLS
PILOT

SUM OUTPUT AND REFERENCE
PROVIDE ERROR SIGNAL

COMPENSATION SENSORS

- BAROMETRIC ALTIMETERS
- THERMOSTATS
- THERMISTORS
- RADAR ALTIMETERS
- AIRSPEED INDICATORS
- PROGRAMMER TIMERS
- ACCELEROMETERS
 - LINEAR
 - ANGULAR
 - SWITCHING
 - PENDULUMS
 - MERCURY SWITCH
 - ELECTROLYTIC POTENTIOMETER
- STATIC PRESSURE SENSORS
- RADIO COMMAND COMPENSATION
- DOPPLER RADAR (GROUND SPEED)
- INERTIAL MEAS. UNIT (ALT. AND SPEED)
- HAND INPUT COMPENSATION
- PARTICLE COUNTERS
 - PN JUNCTIONS
 - GEIGER TUBES
 - ELECTROMETER TUBES
- EVENT SENSOR SWITCHES
- FEEDBACK ("FOLLOW-UP") SENSORS
 - POTENTIOMETERS
 - LIMIT SWITCHES
 - SYNCHROS
 - CODE DISKS
 - LINEAR DIFFERENTIAL TRANS.
 - PROXIMITY SENSORS
 - PRESSURE SWITCHES
 - FLOW METERS

COMPENSATION AND CONTROL LOGIC (COMPUTER)

- ELECTROMECHANICAL ANALOG COMPUTER
- ELECTRONIC ANALOG COMPUTER
 - VACUUM TUBE
 - TRANSISTOR
 - THIN FILM
 - INTEGRATED CIRCUITRY
- MECHANICAL (MANUAL) ANALOG COMPUTER
- ELECTRICAL DIGITAL COMPUTER
- ELECTRONIC DIGITAL COMPUTER
 - VACUUM TUBE
 - TRANSISTOR
 - THIN FILM
 - INTEGRATED CIRCUITRY
 - CRYOGENIC
- HYDRAULIC ANALOG COMPUTER
- PNEUMATIC ANALOG COMPUTER
- PURE FLUID ANALOG COMPUTER
- PURE FLUID ANALOG COMPUTER

- ① PROVIDE INFORMATION FOR GAIN ADJUSTMENTS TO COMPENSATE FOR CHANGES IN FLIGHT ENVIRONMENT

- ① PERFORM ALL COMPUTATIONS RELATED TO LIMIT CYCLE, GAINS, PROGRAMS, GENERATION OF DISCRETES AND COORDINATE TRANSFORMATIONS

SUBSYSTEM (LEVEL 7)

CONTROL AMPLIFIERS

- BALL/DISK INTEGRATOR
- ELECTRONIC FEEDBACK AMPLIFIER
 - VACUUM TUBE
 - TRANSISTOR
 - INTEGRATED CIRCUITRY
- RELAYS
- AMPLIDYNE GENERATOR
- D/A SWITCH BANK
- HYDRAULIC AMPLIFIER VALVES
- ELECTRO-HYDRAULIC VALVES
- MOTOR DRIVEN POTENTIOMETER
- PNEUMATIC AMPLIFIER
- ELECTRO PNEUMATIC VALVE
- PURE FLUID AMPLIFIER
- EXPLOSIVES AND SQUIBS
- PILOT

AMPLIFY ERROR SIGNAL TO
DRIVE OUTPUT CONTROLLER

INFORMATION DISPLAYS
(AND DISPLAY ACTUATORS)

- ATTITUDE BALL
- COSINE ATTITUDE BALL
- SYNCHRO REPEATERS
- METERS
 - GALVANOMETERS
 - SERVOMETERS
 - INSTRUMENT SERVOS
 - BOURDON GAGES
 - DYNAMOMETERS
- CATHODE RAY TUBES
- EYE TUBES
- NUMERICAL DISPLAYS
 - NIXIE/PIXIE GLOW TUBES
 - FLASH CARD RELAYS
 - BEAM SWITCHING TUBES
 - EDGE LIGHT PLEXIGLASS
 - NEON GLOW MATERIAL
 - ELECTROLUMINESCENT MATRIX
 - RELAY STEP COUNTERS
 - CHARACTERON TUBES
 - PRINTERS
- INDICATORS
 - PILOT LIGHTS
 - ANNUNCIATOR FLAGS
 - LUMINESCENT DIODES
 - "FLOW METER" GAGES
- WINDOWS
- RADIO TELEMETRY INPUTS

PROVIDE INFORMATION TO
INTERFACE WITH PILOT.

INPUT CONTROLS
(E REFS.)

- JOYSTICKS
 - PNEUMATIC VALVE
 - STRAIN GAGE
 - POTENTIOMETER
 - HYDRAULIC VALVE
 - LINEAR DIFF. TRANS.
 - SYNCHRO CONTROL TRANS.
 - SWITCHING
 - DIGITAL CODE DISK
- 3-AXIS CONTROL BALL
- CROSS CONTROLS, THUMBWHEELS)
- TOGGLES
- LEVERS
- CRANKS
- PULL CORDS
- PUSH BUTTONS
- KEYBOARD
- HAND VALVES
- PEDALS
- CHOMP SWITCH
- RADIO COMMAND INPUTS

PERMIT PILOT ACTION TO
INTERFACE WITH CONTROL
LOOPS

3

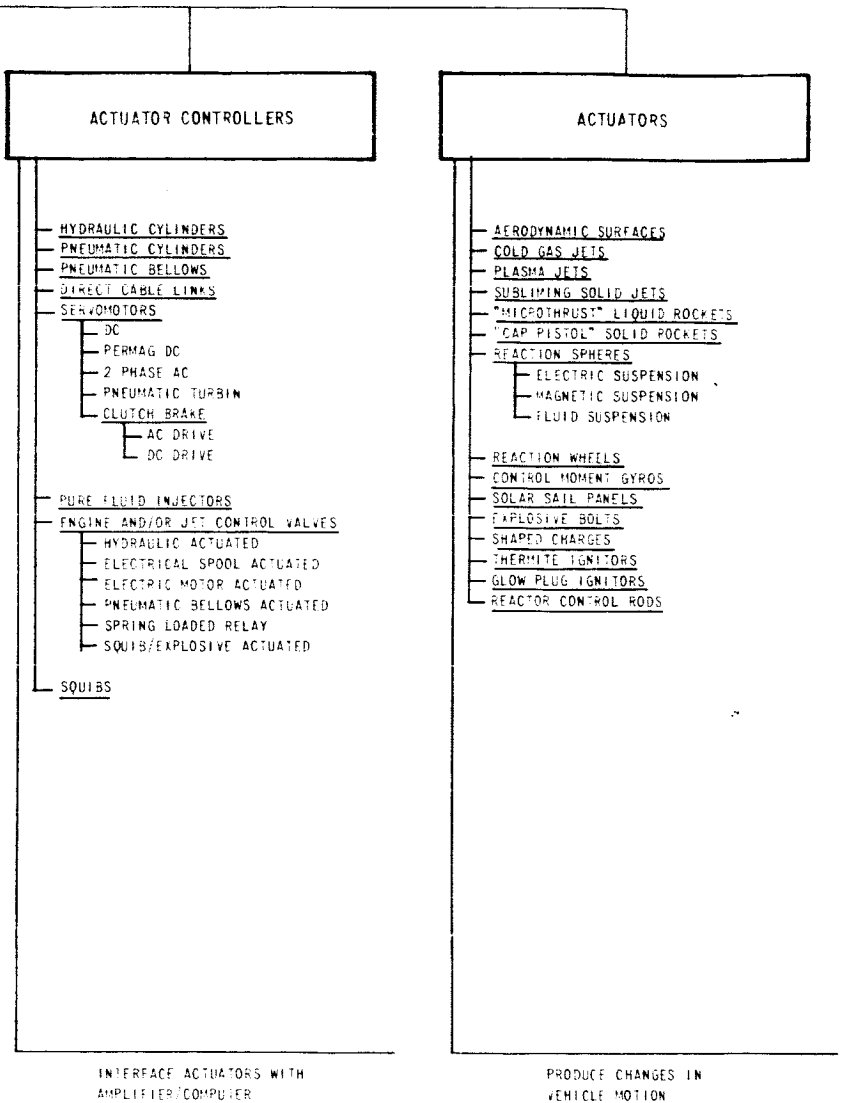
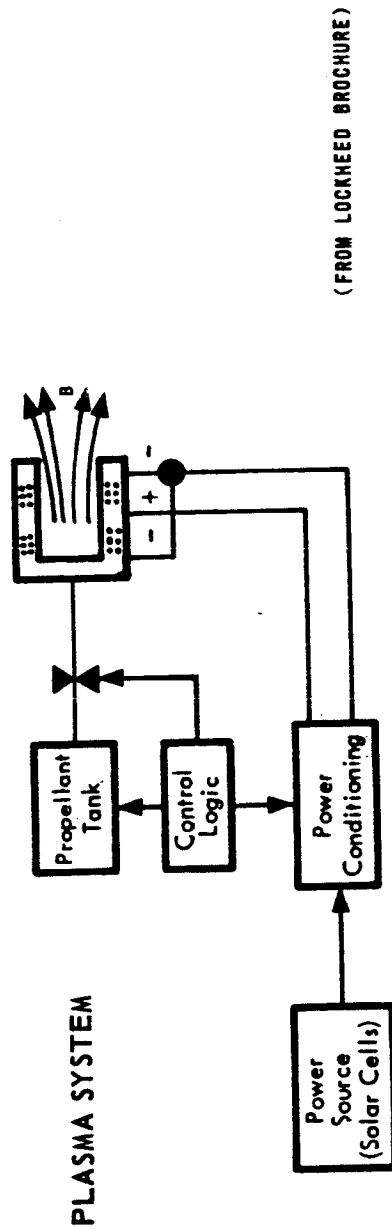
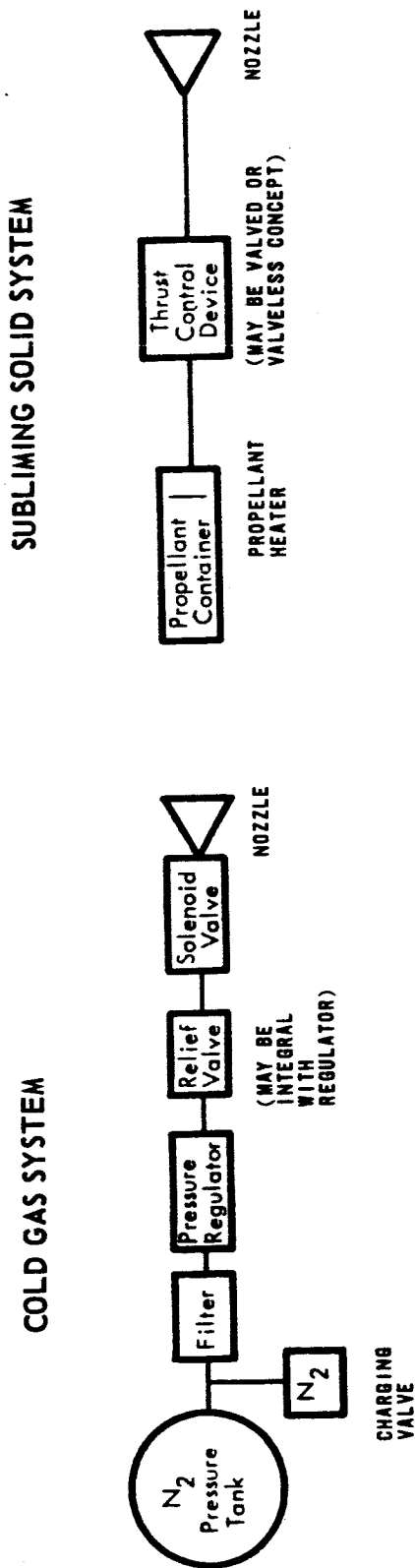


FIGURE 11-2. STABILIZATION AND CONTROL SUBSYSTEM CONFIGURATION CHOICES

11-7

11-8



- Notes:
1. Exit tube and nozzle heater may be required for certain applications.
 2. Pressure or temperature controller may be required for precise thrust control.
 3. Subliming Solid Systems can cut in half the weight of Cold Gas Systems, and utilize any container shape with minimum volume.
 4. Plasma Systems, developed by Lockheed, can decrease by 50 percent the weight of Subliming Solid Systems. For large total impulse requirements, further appreciable weight economies will result because the propellant weight will be cut by factors of 10 to 20.

FIGURE 11-3. LOCKHEED ATTITUDE REACTION CONTROL SYSTEMS

TABLE 11-4. COMPARISON OF REACTION CONTROL SYSTEMS

	<u>Advantages</u>	<u>Disadvantages</u>
Liquid Monopropellant or Bipropellant		Not suitable for low thrust, because re- quired metering orifices are too small
Cold Gas	Covers all thrust ranges Large usage has established reliability Low-pressure, low-temperature operation	High pressure reservoir Potential leakage is consistent problem Large volume requirement High weight factor
Subliming Solids	High response thrust control by use of solenoid valve Simplified valveless design for low- response thrust control Low-pressure, low-temperature operation Packageable to any shape requirement Very long life High reliability potential Special techniques may provide high thrust (10-2 to 10-1 lb) systems without excessive power requirements for heater	Heater may be required to provide heat of sublimation if not available from solar radiation Heater may be required to prevent con- densation in exit line and nozzle
Plasma	New techniques provide simplified design Very high performance systems Weight one-fourth that of Cold Gas system	High power requirements for high thrust

Source: Lockheed Missiles and Space Co.

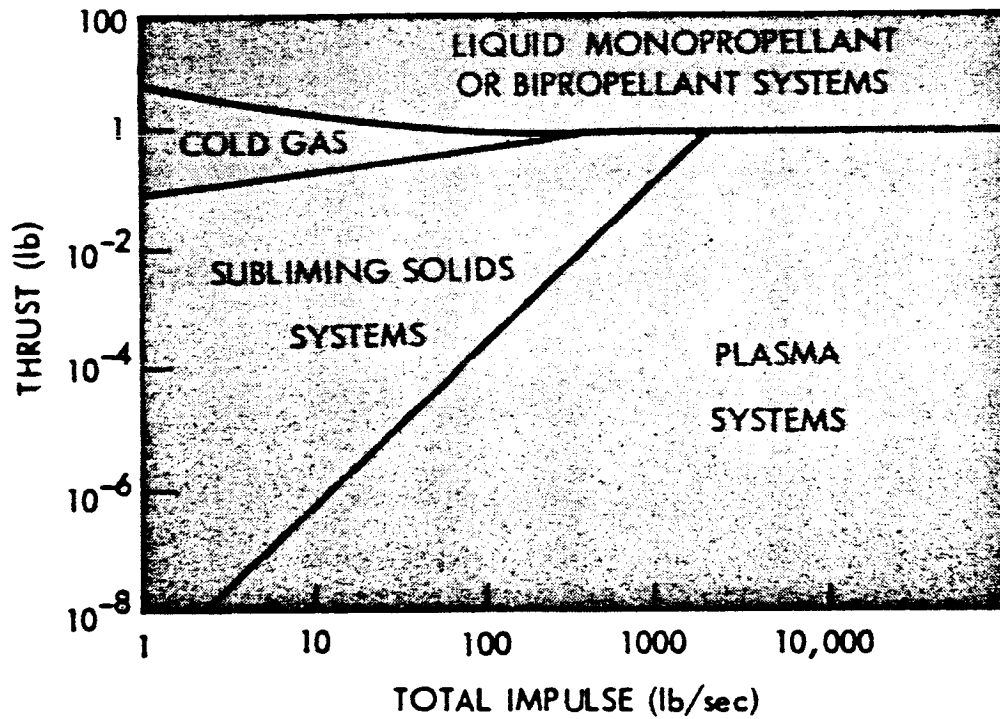


FIGURE 11-4. REACTION CONTROL SYSTEM
OPERATIONAL REGIMES
(Source: LMSC)

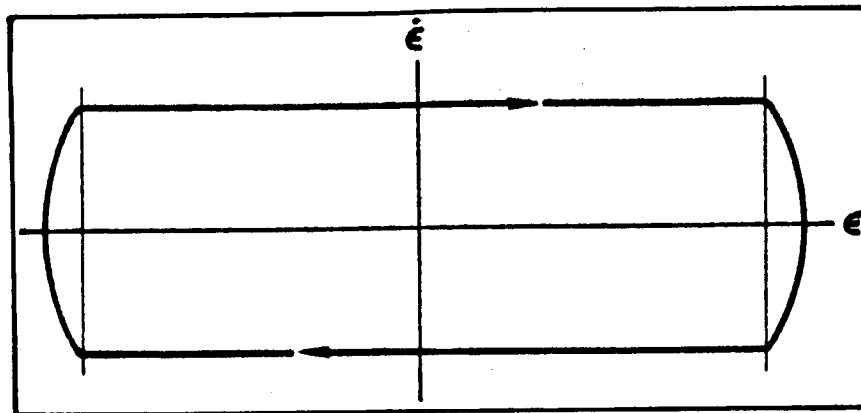


FIGURE 11 - 5. IDEALIZED VEHICLE LIMIT
CYCLE BEHAVIOR

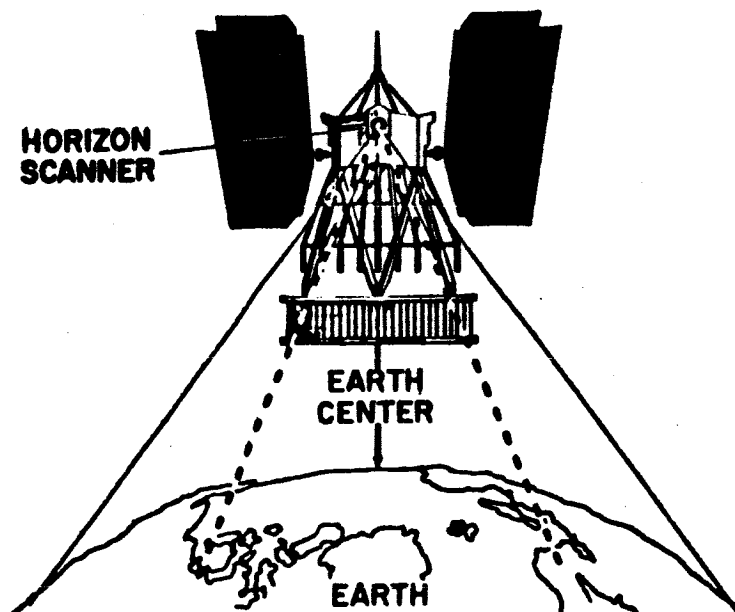


FIGURE 11-6. HORIZON SCANNER CONTROL
(Source: Ref. 3)

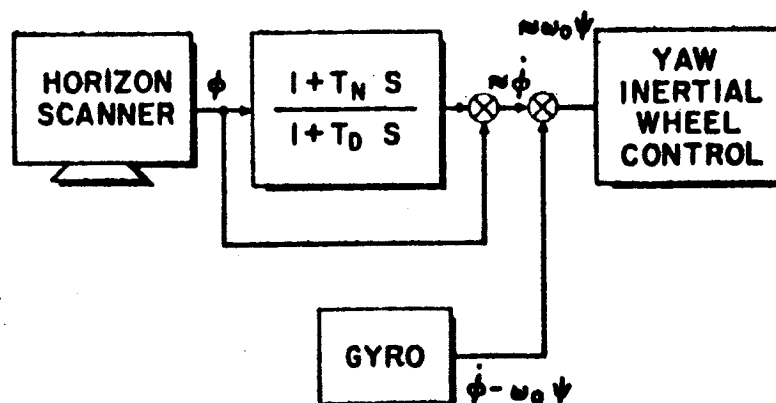


FIGURE 11-7. BLOCK DIAGRAM OF YAW
CONTROL SCHEME
(Source: Ref. 3)

(Quoting from Reference 3)

11.0.3 Control Moment Gyros

"Figure 11-8 itemizes certain control system equipment studies at Ames. An example of these studies is given by a 1962 investigation of a twin-gyro attitude-control system. This control system utilizes two gyros mounted on the same frame, but their rotors spin in opposite directions. For torquing a space vehicle, the gyro gimbals are commanded to turn in opposite directions. The resulting precessional torques from the two gyros combine to react on the vehicle about the desired axis, but cancel each other's cross-coupling torque about the other axis.

In the experimental equipment associated with this study, the vehicle inertia and the torque-free environment of space are simulated by a large structure mounted on a low-friction air bearing. The pilot's seat is at one end of the structure. Three sets of special twin-gyro controllers are mounted on the frame in such a manner as to torque the frame about each of three orthogonal axes. Error signals for control are provided by a set of star trackers. The resulting stabilization system has demonstrated high dynamic performance as well as stabilizing the structure to better than one-fourth second of arc.

11.0.4 Other SCS Methods

Inertia wheels, magnetic torquers, gas and vapor jets have been studied in addition to the gyro control system described. Each has advantages and disadvantages which affect their compatibility with any particular mission.

11.0.5 Pilot Control

The pilot control studies indicated on Figure 11-8 are concerned with improving the display of guidance and control information to the astronaut and optimizing his method of controlling the vehicle. A particularly interesting device under investigation is a solid-state electroluminescent panel suitable for displaying the digital computer output to the astronaut."
(End quote from Reference 3)

11.0.6 Electronic Circuitry

As with many other space vehicle subsystems, the SCS requires considerable electronic circuitry. In this area significant progress is being made toward smaller, lower power, higher reliability electronics. Figure 11-9 forecasts the expected progress in the packaging density of electronic

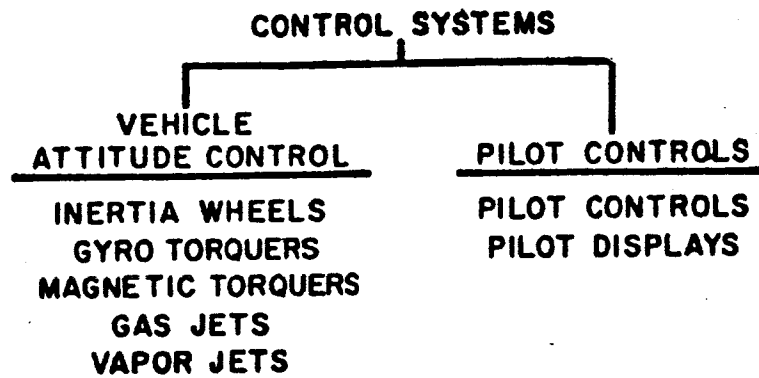
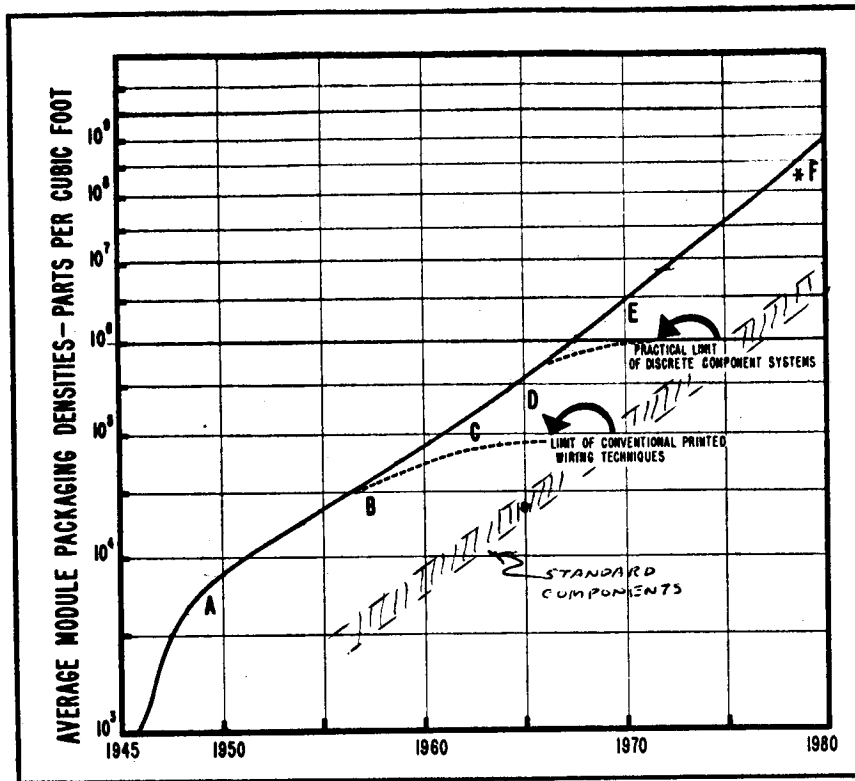


FIGURE 11-8. CONTROL EQUIPMENT
RESEARCH AT AMES
(Source: Ref. 3)



Some concept of the packaging densities that may be expected in the future through microelectronic thin-film circuitry is shown in this chart.
(Source: Ref. 4)

FIGURE 11-9. FORECAST OF PACKAGING DENSITY
OF ELECTRONICS PARTS
(DIVIDE BY 22 FOR STANDARD COMPONENT)

circuitry. Similar forecasts made in the early 1950's were almost unbelievable, but have proved to be fairly close to actual subsequent progress.

11.0.7 Reaction Jets

Wherever high torques and fast reactions are required in SCS, reaction jets are used. Figures 11-10 through 11-13 forecast some of the significant parameters related to the different types of reaction jets.

11.0.8 Control Rockets

One advantage of reaction jets is that they also can be used as control rockets for translational maneuvering.

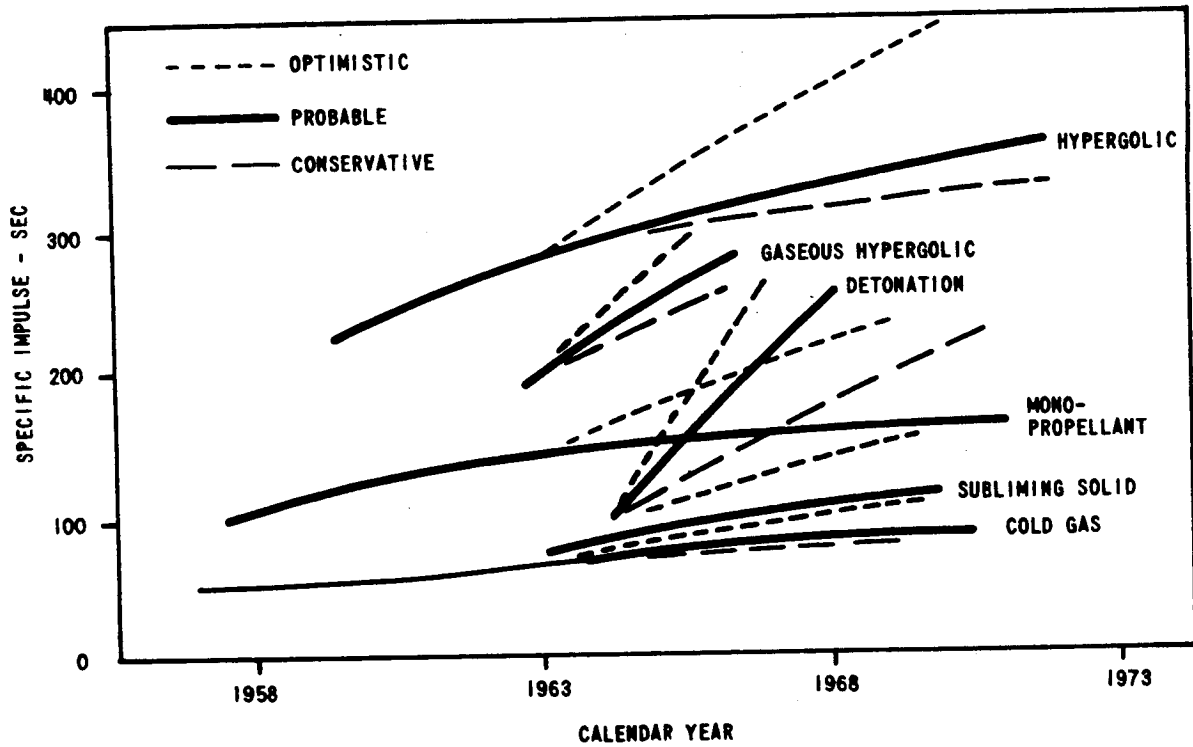


FIGURE 11-10. FORECAST OF SPECIFIC IMPULSE - REACTION CONTROL ENGINES
(PULSED OPERATION AT PRACTICAL MINIMUM PULSE WIDTH)

(Internal Honeywell source unidentified)

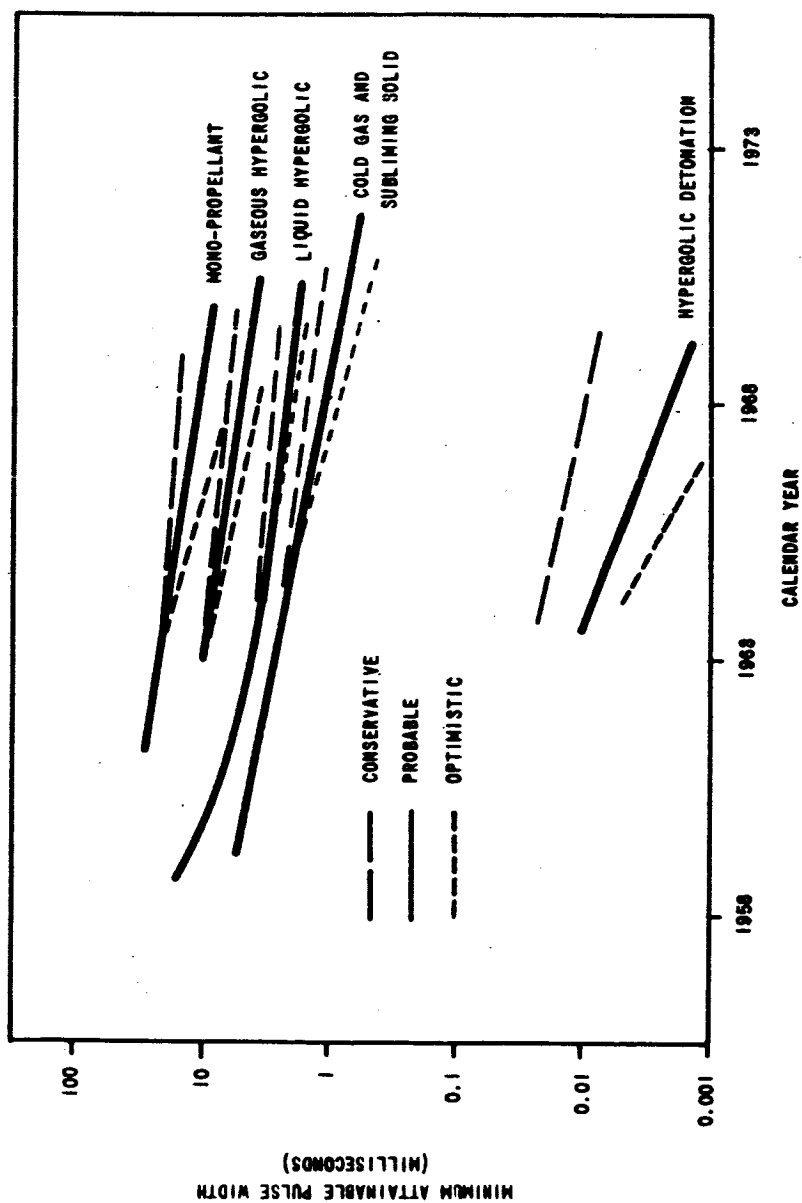


FIGURE 11-11. FORECAST OF MINIMUM ATTAINABLE
PULSE WIDTH - REACTION CONTROL ENGINES
(PRACTICAL MINIMUM CONSISTENT WITH PROPELLANT
THRUST RESPONSE CHARACTERISTICS)
(Internal Honeywell source unidentified)

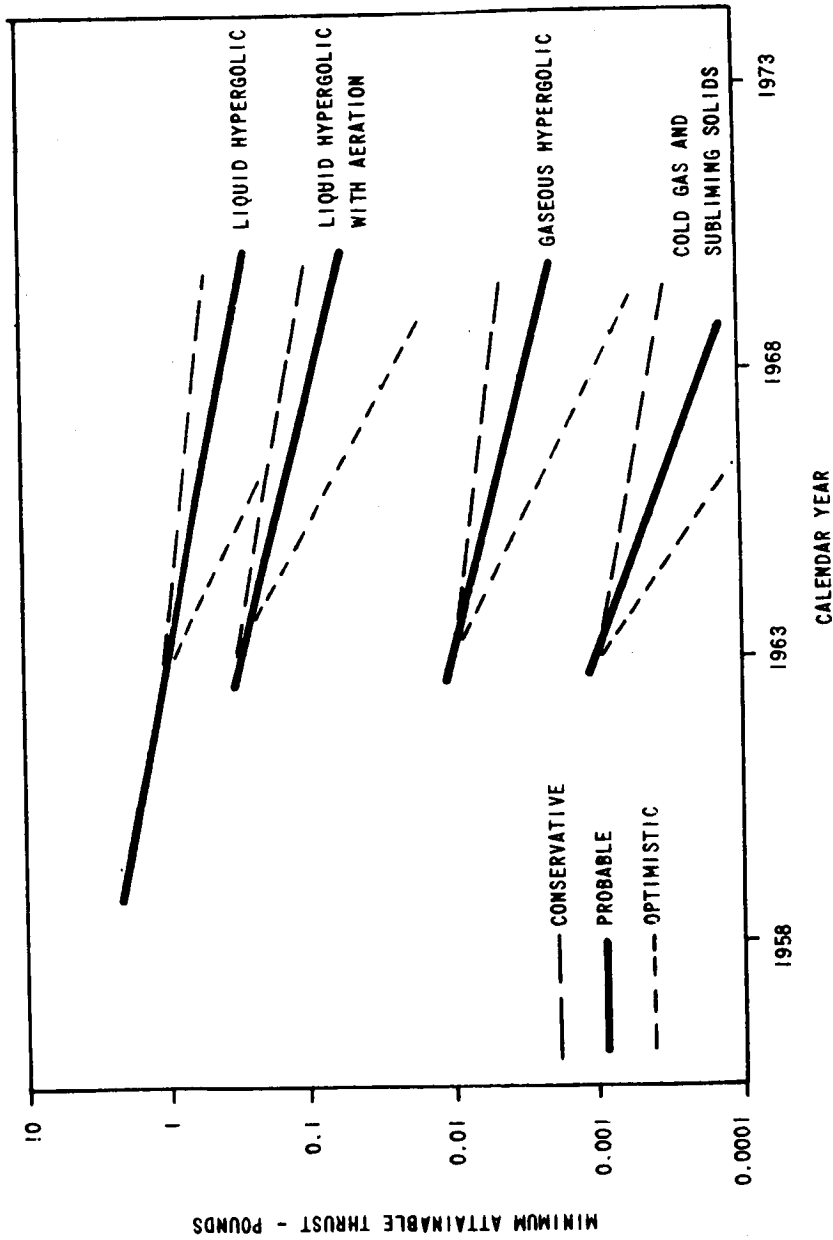


FIGURE 11-12. FORECAST OF MINIMUM ATTAINABLE
THRUST - REACTION CONTROL ENGINES
(PRACTICAL MINIMUM THRUST CONSISTENT WITH
REASONABLE EFFICIENCY)
(Internal Honeywell source unidentified)

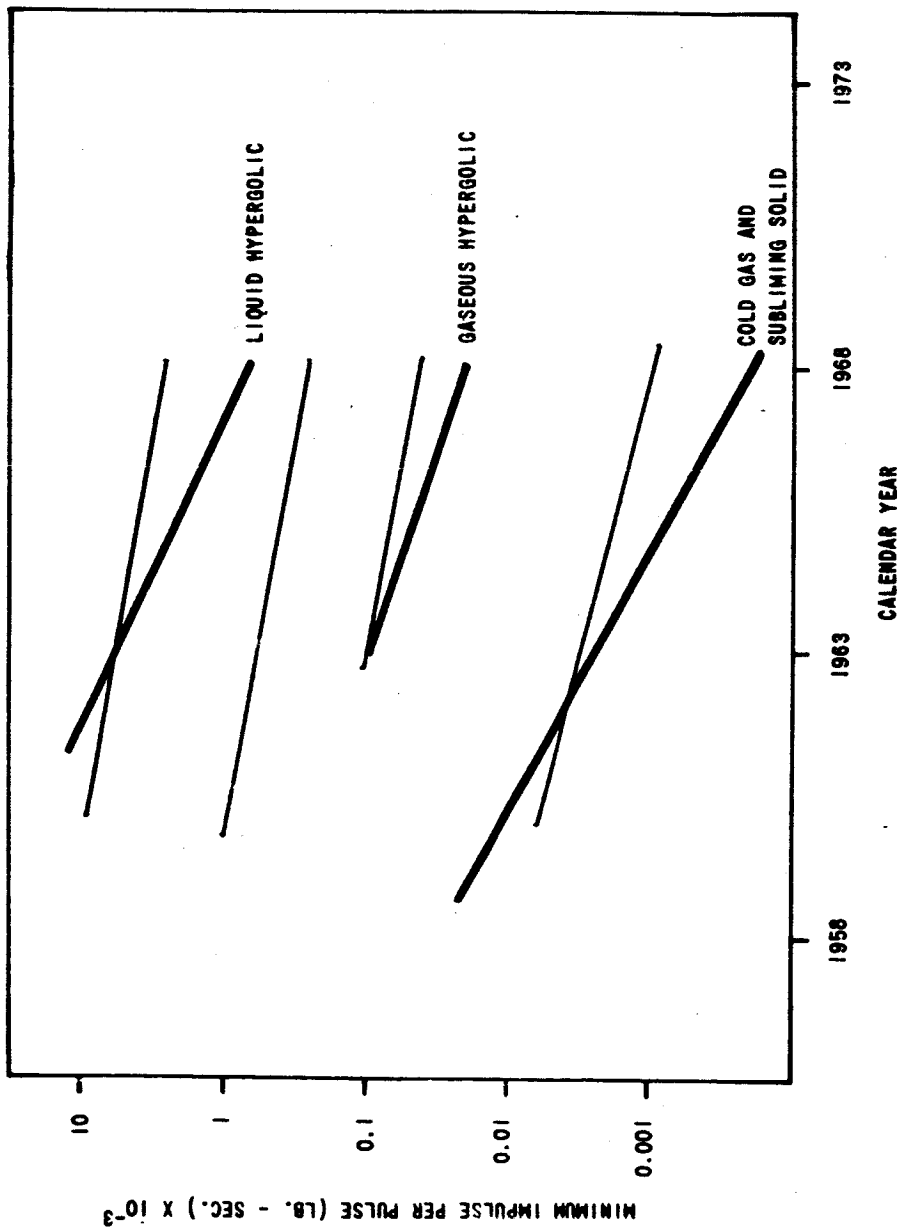


FIGURE 11-13. FORECAST OF MINIMUM IMPULSE PER
PULSE - REACTION CONTROL ENGINES
(PRACTICAL MINIMUM IMPULSE BIT CONSISTENT
WITH REASONABLE EFFICIENCY)
(Internal Honeywell source unidentified)

(Quoting from Reference 5)

The control rocket concept calls for relatively small rocket engines to provide positional control of vehicles operating in low density atmosphere or in space. They have been employed to supplement aerodynamic control on vehicles such as the X-15 and to maintain attitude control on the Mercury spacecraft. The same basic system will be used on future space-airplane, space-glider, and ballistic reentry vehicles. In addition to providing roll, pitch and yaw control the small control rocket engines are used for obtaining precise velocity corrections, propellant ullage forces and any other required type of relatively small motion. It is expected that, in the future, several functions will be performed by a single rocket.

Propellants. Several different propellants and propellant combinations have been used in the past and will continue to be employed in the future. Typical propellants are: cold gas, monopropellants such as hydrogen peroxide and hydrazine, and hypergolic bipropellant combinations, such as the present day storables. Future use of monopropellants, bipropellants of increased energies, and cryogenic propellants is expected. The high specific impulse available from cryogenics such as LH_2 -and-fluorine and LOX-and-hydrogen make them particularly attractive. Ultra-high energy propellants of the boron family are expected to provide a longer mission capability. Even though the weight of propellant used for the attitude and reaction system is small when compared to the total propellant load carried by a vehicle, weight will become more important as mission duration increases. Therefore, the performance improvement offered by higher energy propellants cannot be ignored. The storability of propellants will become more important as mission times rise.

State of the Art. Figures 11-14 through 11-17 forecast the state of the control rocket art.

Valves. Propellant control valves on control rockets operate either by a continuously controlled function, or by a pulse type signal. In both cases the amount of power required to excite the valves is an important drain on the vehicle power supply. Figure 11-15 presents an estimated reduction trend for injector valve power requirements.

Engine Life. An extension of engine life is required and it is expected that a major effort will be placed on this development over the next several years.

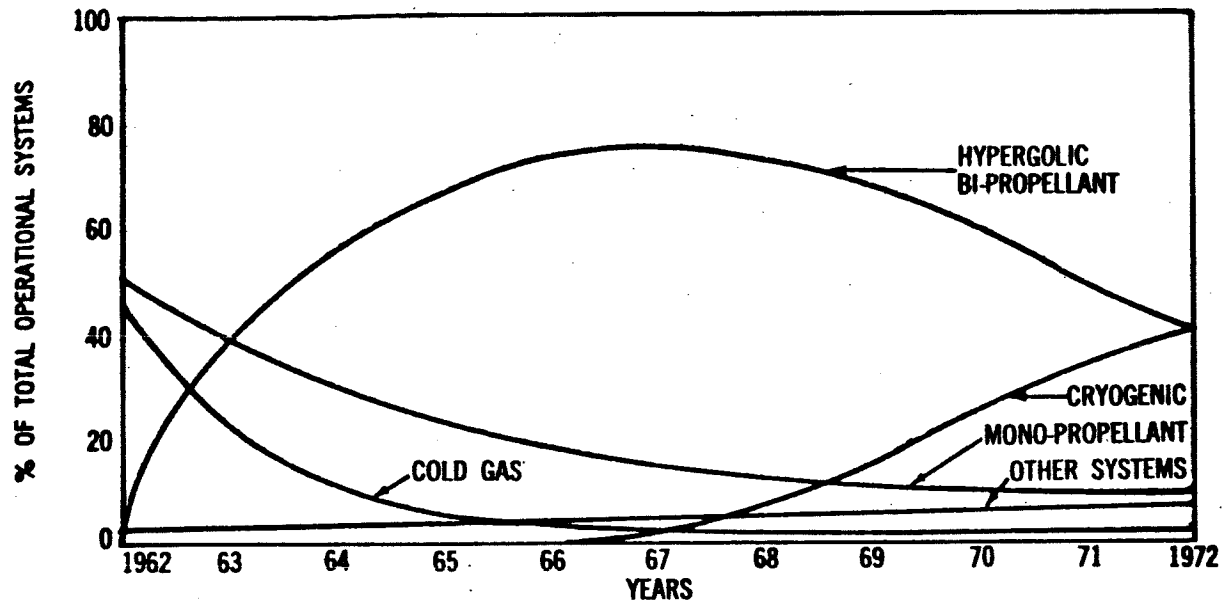


FIGURE 11-14. TYPES OF CONTROL ROCKET SYSTEMS IN OPERATION
(Source: Ref. 5)

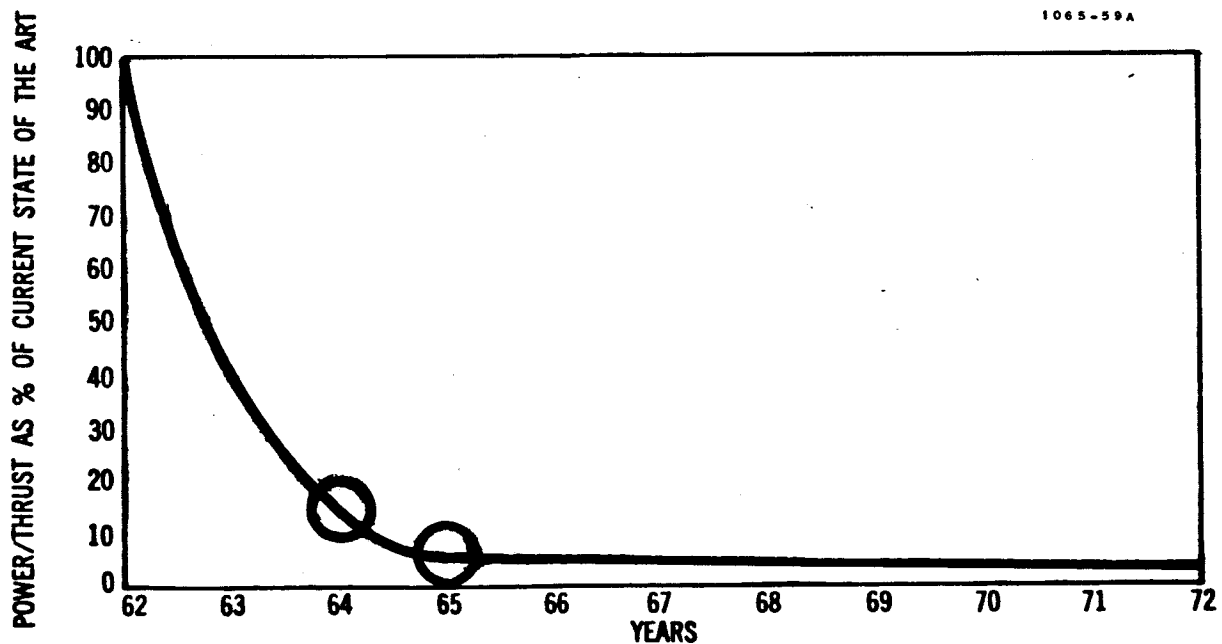


FIGURE 11-15. ELECTRICAL POWER REQUIREMENTS
FOR CONTROL ROCKET SYSTEM
(Source: Ref. 5)

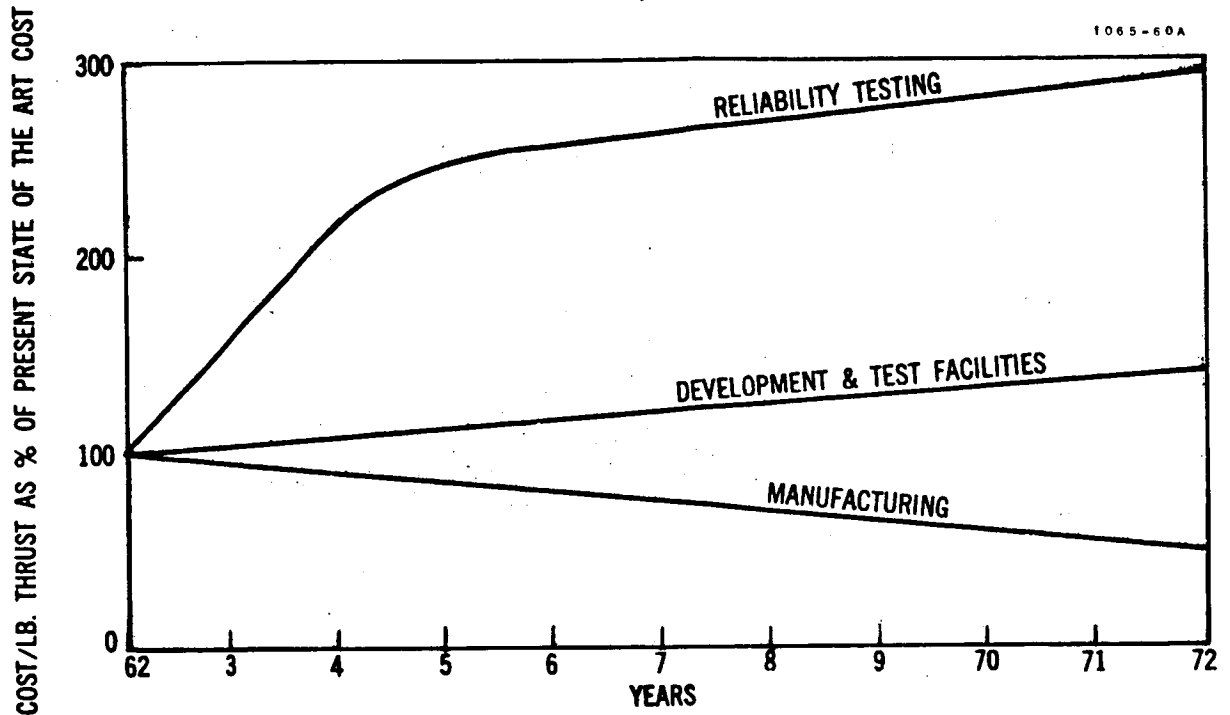


FIGURE 11-16. SYSTEM COSTS
(Source: Ref. 5)

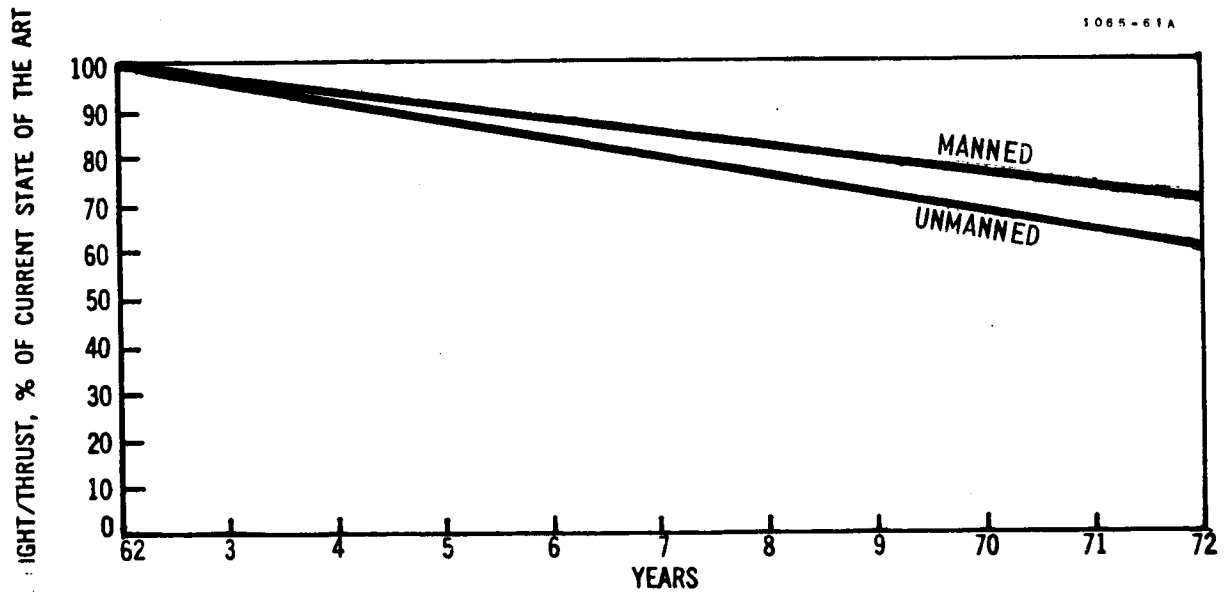


FIGURE 11-17. CONTROL ROCKET SYSTEM WEIGHT
(Source: Ref. 5)

Propellant Tanks. Future developments should allow the use of plastics in propellant tanks, which usually are of metal construction today. The compatibility of plastic materials with the various propellants is the major problem. Several propellant expulsion systems for use under zero gravity and multidirectional accelerations will receive further development. These include: gas-to-liquid, flexible bladder and metal bladder systems. New plastics having the combined properties of good flexibility and non-permeability will be used for many applications. Significant development is needed on metal bladders of both the bag and bellows types. The goals are to obtain lighter weight, corrosion resistance and unconventional shaping. It is expected that in the next ten years propellant-gas separators will be developed for more efficient separation of liquids from the pressuring gas when no physical membrane separates them. Solid grain propellants used for pressurizing gas production will undergo a major development program to enhance their use as pressuring agents for small rocket systems.

Reliability. The currently high reliability of control rocket systems will increase moderately over the forecast period. Man-rated systems will have the highest requirements with a goal of 0.99997. The major effect of this high reliability requirement, on sophisticated vehicle systems, is to raise the number of test hours to a marked degree. To meet this need the number of test facilities for control rockets will increase substantially in the next ten years. Reliability testing will affect system costs more than any other item, as shown in Figure 11-16. Development and test facility costs will increase, but at a slower rate than testing. Manufacturing cost will decrease because engines and systems components will be standardized to allow their use on several vehicles.

Modular Concept. Wide use will be made of the modular concept in packaging control rocket engines. Replacement of defective units by complete system modules is the preferred maintenance approach for long endurance space vehicles. Where possible, control rockets will be supplied from the main propellant tanks and plug-in type engines will lend themselves to easy maintenance.

Weight. New materials and design refinements will allow a reduction in the weight of control rocket systems of all sizes. The projected trend is shown in Figure 11-17. (End quote from Reference 5.)

11.1 HYPERGOLIC REACTION JETS

This category of jets is discussed in paragraph 11.0.8 "Control Rockets". Storable hypergolic propellants, such as N_2O_4 plus UDM hydrazine are simultaneously metered into a combustion chamber to produce a short pulse of thrust. Figures 11-10 through 11-13 show the expected performance by comparison with cold gas jets and subliming solids.

In some concepts hypergolic reaction jets were specified for applications which in reality may require cold gas jets "cap-pistol" solid jets, subliming solids or plasma jets. This was done for simplicity and no significant technological deficiency choices were likely to appear, whatever jets were specified, therefore this inaccuracy is considered relatively unimportant for the present purpose.

11.1.1 Reliability

The critical elements have to do with tank expulsion and propellant metering. Failure or fouling of either the oxidizer valve or the fuel valve can cause jet failure. However, much design attention has been devoted to details of jet reliability, particularly for manned vehicles (see paragraph 11.0.8). The most noticeable effect of reliability requirements is cost increase.

11.1.2 Higher Order Control Techniques

Unlike aircraft whose geometry and flight environment vary only slightly, space vehicles will be changing shape, size, speed and environment both drastically and rapidly. In some missions simple programmed gain, first order control systems would not remain stable under all conditions. Higher order, adaptive systems are coming into use. An example is the FCS for the X-15. The higher order systems modify the loop gains as functions of total environment and vehicle geometry to effect stable operation whatever the external influences. Research and development is required for such nonlinear systems.

11.1.3 Reliability, Long Storage

Many s/c subsystems will be required to remain dormant for months or years, then fire correctly upon receipt of a discrete command. Such difficulties as lubricant jelling or freezing, diffusion welding, or chemical reaction contamination may cause fouling of valve mechanisms as a consequence of non-use. Some periodic dithering of valves during dormancy may be necessary to avoid such risks. The same problem is considered in propulsion subsystems.

11.1.4 Dead Zone

The smaller the dead zone of the limit cycle (see Figure 11-5) the better the attitude accuracy, but the more frequent are corrective impulses and the higher the propellant expenditure. Where extremely stable conditions are infrequently required, but operationally critical, it becomes necessary to adjust the dead zone by programmer. As a consequence, jet impulses and sensor system thresholds must also be adjusted to maintain loop stability. A large dead zone (± 1 to ± 5 degrees) may be used for coasting periods, then tightened to an arc second or less during observational periods. Often in such cases two separate S/C subsystems with different dead zones are used; for example, reaction jets for coarse control and reaction wheels or solar soil panels for fine control.

11.1.5 Attitude Resolution

Some attitude reference system usually is included as part of the S/C subsystem. In the Apollo CM a strapdown analog "Attitude Gyro Coupling Unit" (AGCU) provides coarse attitude reference.

In local vertical stabilization, the attitude reference resolution is limited by the threshold of the horizon sensor (or gravity gradient system).

In space stabilized systems, star trackers can provide attitude resolutions within a small fraction of an arc second, but these are of little use in a local vertical system, except for azimuth reference.

Regardless of the sensors used, attitude resolution of the vehicle system cannot be better than the limit cycle. (See paragraph 11. .4.)

11.1.6 Propellant Limited

Attidue control systems which depend upon reaction jets are limited by the quantity of stored propellant to a finite total change in angular momentum. When the propellant is exhausted, the system is out of business. This almost happened during one of the Mercury flights. Had inadequate propellant remained to permit stabilization during retro fire, that astronaut might still be in some orbit.

Whenever a vehicle must be stabilized continuously for periods of many months or even years, the use of reaction jets implies storage of large volumes of propellant. In such cases solar soil panels, gravity gradient dampers, or reaction wheel systems usually are required instead.

11.1.7 Specific Impulse

(See Figure 11-10.) Hypergolic reaction jets have a specific impulse advantage over other chemical jets. However, improvement in achievable specific impulse (closer to theoretical limit) would tend to alleviate the problem discussed in paragraph 11.1.6. Use of plasma jets and ionic rockets for S/C reactions can greatly extend operating life.

11.2 COLD GAS REACTION JETS

Usually nitrogen (but also other materials) is stored under high pressure; the valve permits short spurts of the material to expand through a nozzle to space. This is a very simple and usually dependable method for exerting reaction torques upon vehicles. It has the advantage of a single material and no combustion.

Figures 11-10 through 11-13 show some of the parametric forecasts.

11.2.1 Fuel Consumption, CGJ

Since the achievable specific impulse is low (50 to 100 seconds) the total impulse is severely limited by the finite quantity of propellant. Astronaut EVA maneuvering units, which employ cold gas jets for safety reasons, are limited to a few seconds of total operation. Extending the useful time to an hour would require prohibitively large propellant tanks. High I_{sp} jets with good safety of operation (no tail scorching) are needed for EVA units and small vehicles.

11.2.2 Reliability, CGJ

Potentially the CGJ is the most reliable of jet systems. However, in expanding, the cold gas takes up heat and refrigerates its immediate environment (it resembles a CO₂ fire extinguisher in this respect). Valve spools have been known to jam due to icing. The presence of trapped moisture in the tanks can severely aggravate this problem, because water expands upon freezing and freezes at a relatively high temperature.

Attention is required to the thermal properties of valve materials to minimize freezing.

11.2.3 Specific Impulse

(See paragraph 11.2.1.)

11.3 THRUST VECTOR CONTROL

The function of controlling magnitude and direction of a $\Delta \vec{V}$ during engine firing may fall within the scope of the G&N subsystem or the S/C subsystem depending upon purely arbitrary choice of terms. Usually, however, the function falls within the S/C area when a simple point/fire thrusting technique is used.

The S/C subsystem may under certain conditions provide attitude (or rate) error signals as commands to the gimbal servos of a gimbaled rocket engine, or it may simply hold attitude with the reaction jets alone. In the latter case it is mandatory that the thrust vector from a fixed engine pass closely through (or near) the center of gravity of the vehicle; otherwise the main engine might produce torques of such magnitude that the reaction jets could not maintain stability.

Thrust cut-off discretely may be generated by a pre-set timer, or by a roll axis accelerometer and integrator with appropriate electronics (or by a stern pointed doppler radar and computer in case of vertical landing or lift off, or by a pair of doppler receivers and a velocity computer in Earth orbit).

Stated simply, the TVC is an elementary, simplified, strapdown guidance system. Often, as in the Apollo CM, it serves as backup for the primary inertial guidance subsystem to increase probability of mission success. In some cases TVC is defined to include the inertial platform or strapdown IMU.

11.3.1 Research and Development Required TVC

In the case of the lunar flying vehicle, the TVC may be required to sense and oppose the surface gravity vector and exert thrust to maintain stability during either hover or lateral motion. In cases of high speed (ballistic) vehicles, it will be required to exert braking thrust near the end of the trajectory to permit a soft landing.

For the most part it will be subject to direct orders from the pilot, but the stability and hover balance functions should not be left to him. R&D will be required to define the minimum adequate system for operation in the low-g, vacuum environment.

For Mars landing vehicles, it will be necessary to gain better environmental data before finalizing the design. This is particularly important to help decide what (if any) aerodynamic control surfaces might be used.

11.3.2 Reliability, TVC

In operations near the surface, even a momentary malfunction of TVC could prove fatal to the astronauts. Reliability in excess of 0.999 is desirable.

11.3.3. Throttleability, TVC

When TVC is applied to exert a contra-gravity force during hover or low speed flight, a pulse-fall-pulse-fall cycle would be most annoying to passengers. If the total vehicle weight is to be precisely balanced with thrust for stable flight, the engines must provide "infinite" resolution throttling for smooth altitude rate control. Step throttling, while superior to pulse-fall-pulse, would be generally unacceptable requiring a sawtooth altitude profile during "horizontal" flight.

11.3.4 Engine

Usually a gimballed engine will be required to permit rapid compensation for shifts in center of gravity. The choice of gimbals or fluid injection vectoring needs experimental design studies, particularly for TVC at several different thrust levels.

One problem will be the overall engine efficiency at various thrust levels. If combustion chamber size and nozzle size are constant, there will be an optimum thrust level for efficiency; usually this will be different from the instantaneous, required thrust.

Since engine failure in flight would be fatal, the required reliability is however close to 1.000 the technology can come. Less reliability represents a sort of astronautical Russian roulette.

11.4 THERMO-ELECTRIC THRUSTORS (From Reference 6)

The plasma jet is a hybrid thermal-electrical propulsion system. In both operational concept and performance, it is located in the transition range between the purely thermal jets (chemical and nuclear rockets) and the totally electrical engines (ion drives). Although the plasma jet relies upon the expansion of a hot plasma for the bulk of its thrust, its energy is supplied electrically, rather than from chemical combustion or nuclear fission and fusion.

Hans Gerdien, in the early 1920s, was one of the first to construct a true plasma jet, or "fluid-constricted-electric arc". These interesting devices

remained little more than laboratory curiosities until very recently when they became potent instruments for high temperature research. In connection with reentry-nose-cone studies, the plasma jet has been called upon to produce tens of thousands of degrees for periods up to a few minutes in duration.

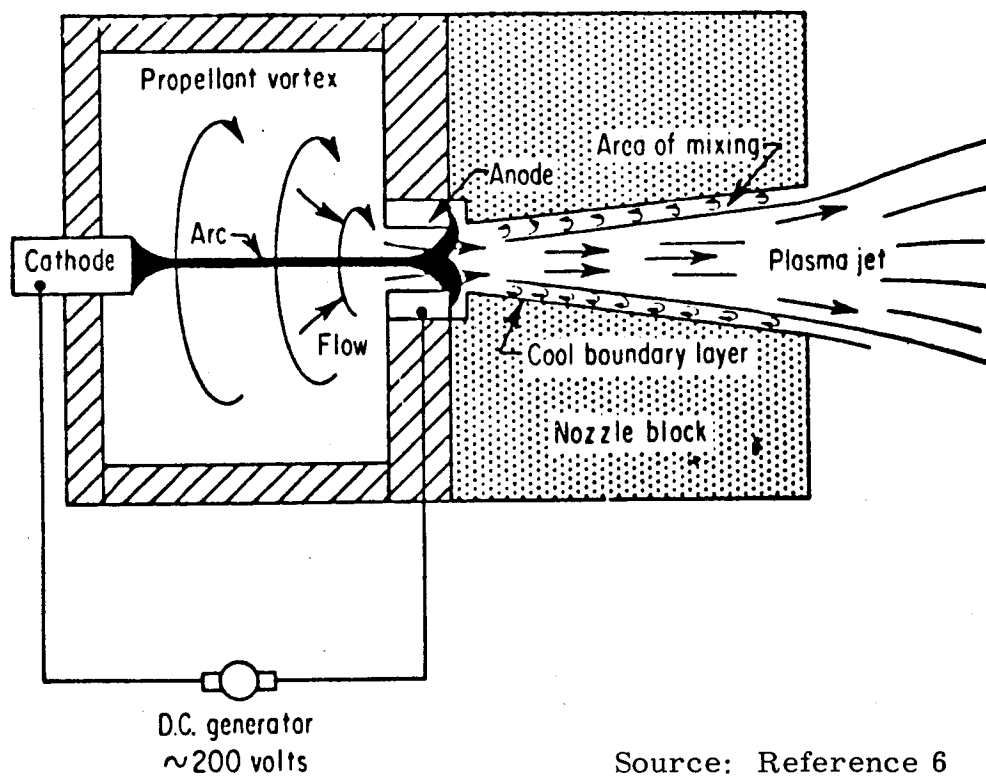
11.4.1 Description of Operation

A somewhat idealized schematic of the plasma jet is shown in Figure 11-18. It consists of a cylindrical chamber with electrodes at either end. The negative electrode may be a plate with an orifice in it or an actual nozzle. The positive electrode is rod-like, possibly with an automatic feeding mechanism to replace material that is consumed in the hot arc.

During startup, a dc voltage is built up across the electrodes. At some potential difference, depending on the pressure and proximity of the electrodes, an arc will be struck between the electrodes, and the region in the chamber core will become a good electrical conductor by virtue of the presence of positive ions and electrons in the arc. In actual practice, the arc generally is started by placing the electrodes very close together at first and then separating them after the arc has been struck. In the arc, electrical energy is being consumed in the ionization process, in molecular dissociation, and in increasing the temperature of the material in the region of the arc. Energy leaves the arc in the form of electromagnetic radiation, as kinetic energy of escaping particles, and in the potential energy of ionization and dissociation. So far, only an ordinary arc discharge in air has been described. Such an arc, using the chamber shown would produce little thrust in itself, but it does represent an admirable heat source.

To make a plasma jet from this device, a propellant-coolant must be introduced along with a means of protecting the surrounding solid structures from the high temperatures created in the arc. The propellant-coolant can be made to perform the latter task by introducing it at high tangential velocity around the periphery of the cylindrical chamber. In this fashion, the introduced fluid will cool the walls and also act as a radiation shield. As the propellant fluid swirls closer and closer to the arc, it naturally gets hotter and hotter as it absorbs heat energy from the arc. It is first vaporized (if it is a liquid), then dissociated, and finally ionized. Each of these three processes tends to cool the outer surface of the arc. The cooling causes the current to concentrate in the hotter, more conductive central regions of the arc. This constriction and resulting concentration of energy makes the arc much hotter than it would be without

1065-50A



Source: Reference 6

FIGURE 11-18. SCHEMATIC OF A PLASMA JET WITH NOZZLE

the fluid vortex surrounding it. Temperatures as high as 50,000°K have been obtained with modern plasma jets. With the generation of high temperatures and the formation of large quantities of vapor, conditions exist for production of thrust through the expansion of the hot, gaseous coolant-propellant. The orifice or nozzle comprising the negative electrode will permit the escape of some of the hot plasma, and a propulsion system is created. The acceleration of the hot fluid is accomplished thermally; pressures of several atmospheres exist in the usual plasma jet chamber. Electromagnetic forces play a very small role in accelerating the propellant in this propulsion system.

(Table 11-5 lists parameters of a typical plasma jet. --Ed.)

11.4.2 Research and Development Required, TE Thrustors

One of the greatest drawbacks of the plasma jet in space propulsion is its present short operating lifetime. Plasma jet operation shows that the expelled propellant includes many particles and atoms of the electrode materials. The hot plasma rapidly erodes away the orifice and nozzle where the jet is formed unless protective measures are taken. Most present-day plasma jets can operate for only a few minutes before the electrodes must be replaced. The pinch effect, may be used in large plasma jets to reduce electrode consumption and thus lengthen its operating lifetime, but the use of electrolytic electrodes, transpiration cooling of the orifice and nozzle, and the development of ablating materials may eventually extend the operating lifetimes of plasma jets to 100 hours and more.

As the temperature of the plasma jet arc rises, its electrical resistance drops, causing it to draw more and more current. In the Earth-bound plasma jets now operating, the (required) current limiting resistance is usually a large and heavy component.

The fact that the plasma jet is basically a thermal engine limits the maximum obtainable specific impulses to about 2,500 seconds. Above this value, temperatures are so high that the propulsion unit efficiency drops to low levels because of the loss of power by radiation and through the expulsion of uncombined ions and atoms. Since the plasma jet is also electrical in the sense that it requires electrical energy for the production of its high temperatures, the thrust-to-weight ratio suffers because of the heavy electrical conversion equipment. The maximum thrust-to-weight ratio for an advanced plasma jet will be about 10^{-3} at 1,000 seconds specific impulse.

The ultimate operating lifetime of the plasma jet is still open to question. While it seems certain that they can be run for hours without seriously

TABLE 11-5. SAMPLE PLASMA JET
PARAMETERS
(From Reference 6)

State of the art	1965
Propellant	Hydrogen
Thrust	50 newtons, 11.2 pounds
Specific impulse	1,000 seconds
Exhaust velocity	9,800 m/second
Thrust-to-weight ratio*	7.5×10^{-4}
Mass flow rate	0.0051 kg/second
Beam power	245 kw
Power input	1 Mw
Propulsion unit efficiency	24.5 percent
Chamber pressure	1 atm
Power-supply mass, 5 kg/kw	5,000 kg
Propellant mass, 100 hr	1,840 kg
Average efflux temperature	3,500°K

* Excluding vehicle, but including propellant for 100 hours of operation.
No recovery of combination energy.

eroding the nozzle and orifice, further tests must be completed to show whether plasma jets will be able to operate successfully for the thousands of hours required for space flight. The degree to which the energy locked up in ionization and dissociation may be recovered cannot be answered conclusively at the present time. The answers to both of these questions will greatly affect the applications of the plasma jet in space.

The plasma jet will probably be most useful in satellite missions where bursts of thrust, at high specific impulse, will be needed for attitude control and orbit adjustment.

One of the problems of developing a plasma rocket is that of containing the high temperature cloud of ionized particles that make up the plasma. Under AFCRL contract, studies developed the so-called "stable pinch", a technique whereby magnets force the plasma stream in a desired path through the nozzle. This optimizes thrust and prevents the walls of the nozzle from melting. This technique is capable of producing specific impulses in the range of 2,000 seconds. A device of this type was tested by Republic Aviation. (See Figure 11-19). (End quote from Reference 6)

Relaxation oscillations within the arc can become a serious electrical noise source in some applications.

Proper shielding and power supply filtering equipment may be essential.

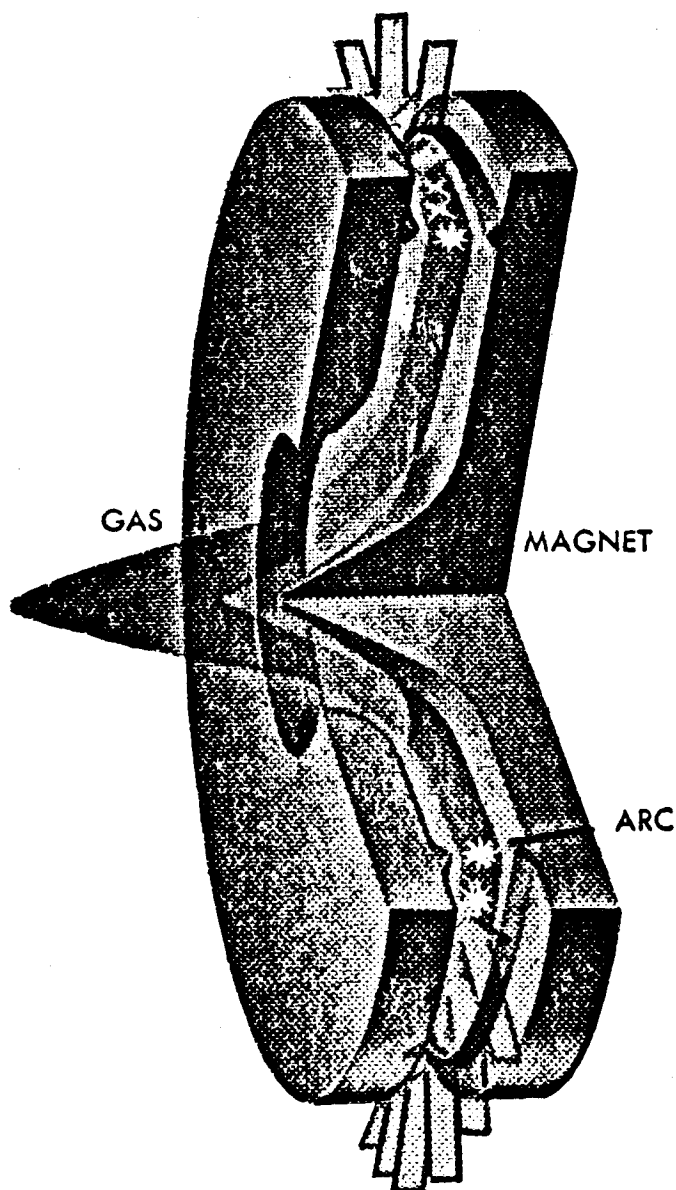


FIGURE 11-19. AIR FORCE PLASMA ROCKET
(Source: USAF Brochure)

Cut-away view of stable pinch nozzle. Gas is forced into passageway between magnets; arc creates plasma, and magnetic repulsion pinches plasma to keep it from eroding walls of nozzle.

11.4.3 Nonlinear Performance

The temperature and impedance of the arc are dependent upon chamber pressure as well as upon the power supply. Simply varying propellant flow does not guarantee linear throttling. The arc supply voltage may be lowered but must not go lower than the ionization extinction potential (at which point the arc would abruptly cease).

Rapid pulsing of the arc with the PRF controlled in unison with propellant flow could permit limited throttling range with reduced efficiency and reduced ISP.

11.4.4 Reliability, TE Thrusters

Until electrode and chamber erosion problems have been satisfactorily solved, MTBFs of plasma jets probably will remain below 100 hours.

11.5 CONTROL MOMENT GYROS

Refer to paragraph 11.0.3.

11.5.1 Reliability, Gyro

The control moment gyros are electromechanical, spinning wheel devices. MTBFs in excess of 10,000 hours are theoretically possible, but actual hardware may fall far short of this. The use of self-generating gas spin bearings, brushless dc torquers and sleeve bearing gimbals may permit very reliable operation.

11.6 AERODYNAMIC SURFACES

Aerodynamic control surfaces have been suggested as a possible flight control means for the Mars Excursion Module (MEM). Present state of knowledge of the Martian atmosphere poses a feasibility question. These surfaces might take the form of pyrolytic graphite spoiler tabs mounted to the vehicle by loose, high-temperature sleeve bearings.

11.6.1 Reentry Heating

The heating rates and radiation cooling possibilities of the Martian environment during entry are not adequately known to permit confident design work to proceed. Voyager landings should improve this situation.

Ablation of the tabs is quite likely.

11.6.2 Rotary Bearings

Tab mounting bearings and control arm bearings must operate at whatever temperatures obtain as a result of atmospheric friction. This may be a minor consideration or it may become the pacing problem. Certainly it will be difficult to maintain close tolerances on bearing clearance; loose bearings probably will be essential, and could cause vibration difficulties.

11.6.3 Stability, Dynamic Range

Refer to paragraph 11.1.2, "Higher Order Systems". The expected variation of environmental parameters during Martian entry and the unknown environment could make an adaptive control system mandatory in order to maintain vehicle stability during the entire entry operation. A proper mix of reaction jet and aerodynamic control may be required to cope with the pressure/speed ranges from high velocity in vacuum to low velocity in the surface atmosphere.

11.7 SOLAR SAILS

Solar sail tabs were used effectively for midcourse attitude control of Mariner IV. In cases where the vehicle is always sun oriented during midcourse flight, rather simple thermomechanical (bi-metal) sensor/actuators for the sail tabs are possible.

Solar radiation pressure is relatively constant at constant solar attitude. Torques are varied by varying the effective sail area normal to the sun line. This is accomplished simply by tilting the sail tabs on hinges. Four such tabs permit two-axis orientation toward the sun. The resulting torques are small and continuous, and with pairs of sails on opposite lever arms, torque resolution can be infinite. This permits control within very small dead zones.

The method is flight proven state of the art.

11.8 ELECTROMECHANICAL DRIVE

Two applications of importance are called for in the concepts: 1) control of parachute steering by electromechanical capstans to permit controlled recovery of instrument packages from high altitude rockets, and 2) torquing of solar cell panels against local vertically stabilized satellites.

Although the applications are quite different the mechanizations are similar. A low power, high speed servo motor will be severely geared down to provide high torque at low speeds (one revolution per orbit to a few revolutions per minute for parachute steering).

11.8.1 Low Cost

In the research rocket application cost must be kept low because of the great number of rockets needed. Servos for toys and for radio tuning have been built for less than \$2.00 (materials cost for motor, gear box and relay amplifier). At the low speeds involved the servos can be greatly overdamped, and worm gears permit positive locking. Die cast or stamped gears and framework cost a few cents per unit.

11.8.2 Weight

Total weight of the parachute capstans and their servos probably can be held to less than one-half pound each for the research rocket capsule recovery (four are required).

Total weight of solar cell panel torquers probably can be held to less than five pounds (one required).

11.8.3 Size

The physical size of the servo mechanisms for the parachutes will be comparable to the size of a Baby Ben alarm clock. A toy, permag dc motor, one three-state relay, a transistor relay amplifier and a worm drive gear train and shroud line spool (capstan) will occupy about three cubic inches (49 cc).

For the solar panel torquers a somewhat more powerful servo could be used, if it were essential to rapidly return the panel to solar orientation during attitude maneuvers. However, there is little loss if the panel requires several minutes to overtake an attitude change. Fast tracking servos would be wasteful for such an application. Volume of the required mechanism probably can be held to less than 10 cubic inches (164 cc).

11.8.4 Dynamic Range

The torque and speed of servos for both applications are essentially constant. There is little need for proportional control. A constant speed, bang-bang relay servo with a fairly large dead zone should be quite adequate. Dynamic range is considered noncritical by the writer.

11.8.5 Vibration

The parachute servos could even chatter wildly without serious effects.

Vibrations in the solar panel application could introduce noise into observational equipment by causing spacecraft vibrations within the attitude control dead zone. This would reduce optical instrument resolution. Close tolerance gearing with proper lubrication can avoid this trouble. The additional cost is allowable in the satellites.

11.8.6 Threshold Rate Sensor

Since a damping factor of two or more is allowable in such low performance servos, it is doubtful that a rate feedback will be needed at all.

11.8.7 Accuracy, Pointing

Pointing of solar panels is not critical ($\pm 5^\circ$ of the sun line is adequate and easily provided by a low grade servo). Short term errors of as much as 90 degrees for several minutes would not be serious, since the storage batteries can take up the power load, until orientation is restored.

11.8.8 Reliability

The simple servos should operate for several years unattended in the solar panel application.

MTBF of one hour would be adequate for the parachute capstans.

11.8.9 Bearings, Solar Panels

The solar panels represent a large inertial mass against which the satellite is torqued by the servo. Bearing friction is the major load to be overcome by the servo. Seizure of the panel support bearings could cause the solar panels to lose track of the sun with consequent serious reduction in auxiliary power output. Since rotation is always in the

same direction, electrical output from the solar panels must be transferred to the spacecraft either by slip rings or directly through the bearings (use of conductive, solid lubricant bearings should permit this).

11.9 ELECTRIC PROPULSION SYSTEM

As discussed in paragraph 11. .6, reaction jets are severely propellant limited due to low specific impulse. Long life satellites with large auxiliary power subsystems could use ionic rockets to advantage for both station keeping (orbit maintenance) and attitude control. In the case of synchronous satellites stabilized to local vertical, attitude disturbances are slight; low, continuous torque such as that possible with ionic rockets should be adequate for S/C actuation.

Since specific impulses in the tens of thousands are possible, the propellant limitation is considerably reduced with ionic rocket control motors.

11.9.1 Research and Development Required, Electric Propulsion

Ionic rocket research is in the early experimental flight test stage. Adequate motors should be available within three years of a firm order.

11.10 REACTION WHEELS

Reaction wheels and particularly reaction spheres are often considered as angular momentum storage and exchange devices. In contrast to control moment gyros, the precession torque is avoided by free suspension of reaction spheres. Momentum is transferred to the sphere by magnetic torquing to accelerate it. Where oscillatory disturbance torques are to be overcome, the sphere is quite effective. Where rotation is nearly always of one sense, a limiting speed is eventually reached and momentum must be dumped from the sphere with reaction jets.

However, in concept EO05 reaction wheels very likely could be interpreted to mean control moment gyros (see paragraph 11.0.3).

11.10.1 Wearout, Reaction Wheels

(Refer to paragraph 11.5.1, "Reliability, Gyro".)

11.10.2 Vibration, Noise, Reaction Wheels

Gas spin bearings are needed, but no bearing is completely free of vibration noise. In view of the stability requirements for solar scan, some problem may exist.

11.11 SERVO CONTROLLED TO SIDEREAL RATE

The problem referred to is one of space stabilizing telescopes mounted to the lunar surface. There is some physical libration of the moon due to torques from the Earth and sun. These would manifest as minor variations in the moon's rotation rate and would amount to a few arc-minutes per month (oscillatory). This libration in sidereal rate is quite small and averages out to zero over one year. However, it may be desirable to make time exposure photographs for as long as 14 days, and this minor libration would cause streaking of star images. If it were not for this consideration, a simple, equatorial clock drive would adequately stabilize the telescope.

The velocity servo input could take the form of a variable frequency oscillator whose frequency is controlled by the star angle error output of a narrow field, image dissector star tracker. This tracker would be mounted to the equatorial drive table and manually trained to center a distant, key star. The star tracking dead zone could be as small as 0.0001 arc-second (optics limited, perhaps to 0.001). Any variation in star position would cause an appropriate increase or decrease in the oscillator frequency. The oscillator would drive a synchronous clock motor through a power amplifier. The clock motor drives the telescope mount in right ascension through a worm gear. The average clock speed would torque the telescope mount in right ascension at average moon's rate. Variations in speed would permit compensating the slight variability in moon's rate.

The telescope would be position servo driven, with respect to the equatorial mount, in both right ascension and declination by moderate speed (gun laying type) dc servos with digital pickoffs. After positioning upon command these servos probably would be shut down to avoid hunting noise. Positive worm gear drives with zero backlash are essential. Vernier velocity servos with the synchronous clock drive might be applied for planet tracking motions. No insurmountable problems are seen.

11.11.1 Precision Angular Measurement

Image dissector star trackers (1965 state-of-the-art) are adequate for sensing angular error. Telescope position in both right ascension and declination can be read out to ± 0.1 arc-second with large (30 inch diameter) dynagon or inductosyn servo pickoffs. The greater angular precision (0.001 arc-second) should be obtained from the stars themselves as an angular reference. As long as stability is adequate, it should not be necessary to measure right ascension and declination better than ± 0.1 arc-second, and even this seems unnecessarily precise.

11.11.2 Servo Threshold and Null Angle

Thresholds of the telescope positioning servos are relatively unimportant for star observation (the servos are shut down), but might become a source of trouble in tracking planets for time exposures. Digital servos with angular dead zones of less than one arc-second (using Dynagon pickoffs for position and rate feedback) have been built with only 3 inch diameter dynagons. By using larger Dynagons, any dead zone desired is achievable, especially in such a very slow servo loop as this. Dead zone of the sidereal velocity servo can be as little as 10^{-4} arc-seconds/day.

11.11.3 Material Degradation

Most of the critical electronic and electromechanical elements would be sealed in pressurized shields. Degradation of telescope bearings due to lunar dust abrasion may be a factor which causes gradual increase in stability dead zone.

11.11.4 Reliability Servo

MTBFs of 500 to 1,000 hours should be fairly easy to achieve. MTCFs of many thousands of hours should be achievable with maintenance and periodic inspections.

11.11.5 Temperature Distortions

The sidereal velocity servo can compensate for mount position angle leads or lags due to thermal cycling, but it could not compensate for shifts of the optical axis due to thermal gradients.

A sunshade or "tin umbrella" may be needed to keep the entire telescope shielded from solar radiation.

11.11.6 Uncertainty in Lunar Surface Vibration

A seismometer actuated shutter could interrupt exposures at the onset of any surface vibration.

It may be necessary to keep men and machines at a few kilometers distance during observations if the lunar surface proves to be somewhat unstable.

Certainly it is not practical to attempt servo compensation for surface vibrations.

11.12 ADAPTIVE FLIGHT CONTROL SYSTEM

System EO08 T Reusable Booster

The flight environment ranges from low subsonic sea level flight to Mach 22 stratospheric flight. This range involves variations in vehicle configuration (before and after second stage release) and variable wing geometry.

A flight control system somewhat more sophisticated than the X-15 autopilot will be necessary to assure stability during all phases.

(See also paragraph 11.1.2.)

11.12.1 Higher Order Adaptive Techniques

Mathematical analyses, system designs, computer software and sensors will require research and development. Theory basic to the required design is now in the elementary stage of development.

11. 13 GRAVITY GRADIENT

(Quoting from Reference 2.)

"Getting a communications or weather satellite to point continually toward Earth is a tricky business, especially if short-lived active systems such as jet thrusters are ruled out. But gravity-gradient stabilization does the job nicely, and in a gentle way that takes little or no effort to comply with the inevitable laws of physics. That's why its future is so bright.

If the Moon were a perfect sphere, chances are we would be equally familiar with its whole surface, not just half of it. But the Moon isn't completely symmetrical, and as a result, one side continually faces us as it orbits the Earth. This is gravity-gradient stabilization at work.

This same principle is now being put to use on lesser Earth satellites, where a passive, continually Earth-pointing orientation system is desired, such as for communications and weather reporting. Key to the principle is the fact that the Earth's gravitational field is not constant-- it diminishes as one moves out from the Earth. Thus, unequal gravitational forces act on a vehicle that has any size at all, and if it is nonspherical there will tend to be a preferred orientation with respect to the Earth. An orbiting rod or dumbbell, for example, would tend to align itself parallel to the local vertical, and a disc would tend to align parallel to the orbit plane.

The stabilizing forces acting in such a gravity-gradient system are extremely weak-- on the order of a ten-thousandth or hundred-thousandth of a pound. But that's all that is needed, for they are acting continuously and against equally slight disturbances. Where most active systems (for example, cold gas jets) keep a satellite pointing in the right direction by fast-acting brute force, gravity-gradient is soft and slow, but capable of pointing accuracies of one to three degrees in a nearly circular orbit. For even greater accuracies, gravity-gradient becomes weight prohibitive and would require a perfectly circular orbit.

In one system, an extendible rod device makes possible large moments of inertia. In Advanced Technology Satellite, the one-half inch rods will hold 10-pound weights 100 feet out from main satellite body.

A likely use for gravity-gradient stabilization is in a communications satellite, where the requirements are for a lightweight (25 pound or so), long-lived (many years) system. By being able to use only one antenna continually pointed toward Earth, the communications engineer has, in effect, been given at least a fourfold bonus in available antenna gain, compared to what he would have with simple spin orientation.

In the case of weather satellites, studies show that at the lower altitudes (500 nautical miles or so), a 25- to 35-pound gravity-gradient system can readily satisfy the few-degree pointing accuracy and several-day damping time required and be lighter and more long-lived than active systems.

Normally, gravity-gradient stabilization is not considered practical at altitudes of 100 to 400 miles, because aerodynamic drag is an overpowering disturbing force. However, it is possible to design a gravity-gradient system which, by having an aerodynamic configuration, capitalizes on this drag. Such a system is expected to be flight tested within a year, and could open up bio-satellites and other low-orbit scientific vehicles to gravity-gradient orientation.

In the bio-satellites, for example, the stabilization requirements are two-fold: a slow oscillation rate to insure a zero-g environment, and a known final orientation of the spacecraft to allow deboost of the retrievable capsule. Since the natural oscillatory frequency of such a gravity-gradient configuration is low-- on the order of one-half to two cycles per orbit-- and the oscillation amplitude would be only a few degrees, the near-zero-g requirements can be met. Moreover, precise orientation of the satellite is needed only at the end of the mission so that there is time for the decay of initial pointing errors.

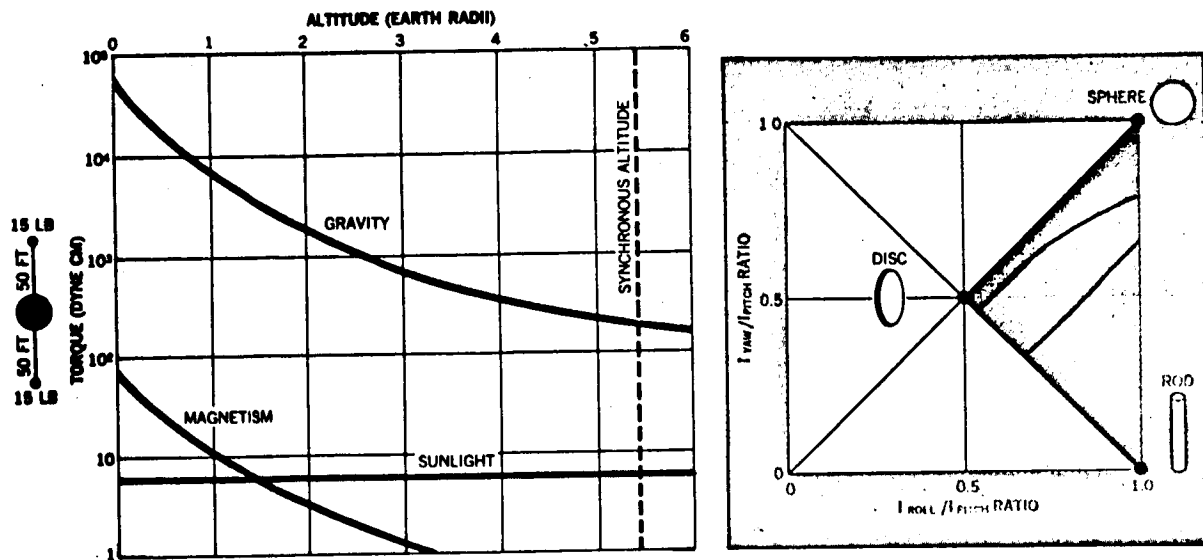
As a generalization, then, it can be said that the gravity-gradient state of the art has advanced enough in the past one or two years to assure the feasibility of stabilizing any spacecraft at altitudes from 100 nautical miles up through synchronous orbit. Weight of such a system for synchronous orbits would be about 35 pounds for satellites weighing up to 1,000 pounds, and the near-circular orbits can be met with present boosters.

11.13.1 Damping, Gravity-Gradient

The one-to-three degree accuracy, however, can be achieved only if the vehicle is properly configured, and if the system has an effective damper. This latter requirement comes in because a gravity-gradient satellite will always tend to oscillate about local vertical, either because the oscillation (or tumbling) is introduced when it is freed from the launch vehicle, or because it encounters disturbing torques that have now become relatively significant, such as the Earth's magnetic field or solar radiation pressure. The function of the damper, then, is to absorb the energy of oscillation. (In the case of the Moon, it is believed that possibly its viscous fluid core damped out most of what motion there was. But a slight, slow oscillation still exists today.)

TABLE 11-6. GRAVITY-GRADIENT STABILIZATION SYSTEMS
(Source: Ref. 2)

Damper Reference Force Field	System Name	Sponsor	Damper Mechanism	Comments
Gravity	Passive Gravity Altitude Control	Bell Labs	Magnetic hysteresis	Unique damping concept; configuration is sensitive to solar pressure and orbit eccentricity
	Vertistat	General/ Dynamics Astronautics	Viscous fluid	Good at low (500 to 1,000 miles) altitudes; damper concept difficult and vertical rod is in payload field of view.
	Gravity Anchored Passive	General Electric	Eddy current	Uncluttered payload field of view and good damping; oscillation decay time is exponential rather than linear
	Ames System	NASA-Ames	Undefined	Valid for all altitudes and orbit inclinations; vertical rod is in payload field of view.
	Articulated Gravity	University of Toronto	Undefined	Rapid oscillation decay time; sensitive to solar pressure at higher (above 6,000 miles) altitudes.
Magnetic	Magnetically Anchored Gravity	General Electric	Viscous fluid and eddy current	Simplest design; oscillation decay time lengthens with accuracy required, and is dependent on orbit inclination.
Magnetic and Gravity	APL System	Johns Hopkins Applied Physics Lab.	Lossy spring and magnetic hysteresis (using Earth's magnetic field).	Good for low altitudes; weight prohibitive at high altitudes.
Solar	Passive Orienta- tion and Damping	General Electric	Eddy current	Can be configured to orient solar cells; not suited for dawn-dusk orbits and decay time lengthens with accuracy required.
Aerodynamic and Magnetic	Aerodynamic Magnetically Anchored Gravity	General Electric	Viscous fluid	Only known approach for very-low altitude orbits; decay time is sensitive to orbit inclination.



SATELLITE CONFIGURATION greatly affects performance of gravity-gradient system. Drawing at left shows typical design torques for 15-sq-ft vehicle oriented about two axes. Torque due to solar pressure has been minimized by limiting distance between center of mass and center of pressure to only 1 in. Satellite's net magnetic dipole has been kept to reasonable level of 400 pole cm. In choosing vehicle configuration (right) designer is limited by certain ratios of moments of inertia (I) about the various axes, given which way he wants vehicle to point. Dark blue area defines such "monostable" configurations—that is vehicle will be stable but will not know difference between "up" or "down." "Bistable" area (dark gray) is forbidden region in which vehicle would have two additional and unwanted stable positions. Configurations along heavy black lines are also forbidden, as orbital frequency or its harmonics will match, and therefore resonate with, natural oscillatory frequency of a vehicle so configured. This in effect would add error to the system. Unshaded areas are physically unrealizable shapes, or regions of instability.

FIGURE 11-20. GRAVITY STABILIZED SATELLITE CONFIGURATION
(Source: Ref. 2)

As far as the actual damping mechanisms are concerned, four types are now under consideration-- viscous fluid, magnetic hysteresis, eddy currents, and mechanical hysteresis (lossy spring). The particular choice is a somewhat arbitrary one. (Paragraph 10.16 compares damping methods under the heading of "Lossy Spring Damper".)

While the main function of the damper is to eliminate the large oscillations that are usually imparted to a satellite when it is freed from its launch vehicle, there is also another function-- to help suppress the smaller oscillations that are imparted by unwanted disturbing torques. (End of quote Ref. 2)

11.13.2 Attitude Accuracy

Refer to Figure 11-21. Attitude oscillations gradually decay toward zero, but random disturbing torques can require re-settling. If attitude accuracy better than ± 3 degrees is required, it probably is advisable to augment the control with an active system during those times when tight tracking is required.

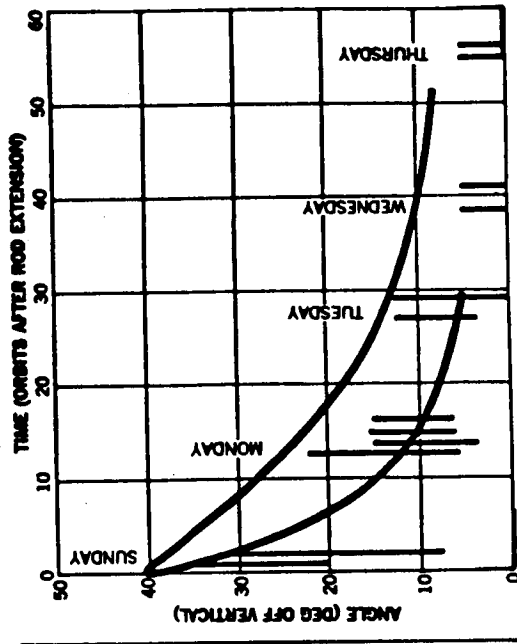
11.13.3 Settling Time

Refer to Figure 11-21. Perhaps research in damper methods can lead to methods for a slight reduction in settling time.

TABLE 11-7. NOMINAL ATS GRAVITY-GRADIENT SYSTEMS
(Source: Ref. 2)

	Orbit	
	6000 nm	Synchronous
Main boom length (ft) ^a	100	100
Main boom tip weight (lb) ^a	2.5	10
Damper boom length (ft) ^a	90	90
Pitch moment of inertia (slug ft sq)	3,812	13,476
Roll moment of inertia (slug ft sq)	3,402	11,723
Yaw moment of inertia (slug ft sq)	410	1,413
Spring constant (dyne cm/deg)	21.77	5.36
Damping constant (dyne cm/deg/sec)	27,166	25,030
Pitch restoring torque (dyne cm/deg)	158	39
Roll restoring torque (dyne cm/deg)	240	59
Yaw restoring torque (dyne cm/deg)	7.2	1.8

^aFor each of four booms. ^bTip to tip.



OSCILLATION DECAY with gravity-gradient stabilization can be predicted in a continuous fashion by a computer, but results from actual measurements in space take a cruder form. Curves at left show a typical computer prediction for the Advanced Technology Satellite orbiting at 6000 miles. Discontinuity in yaw indicates end of one complete yaw revolution of satellite. At right are data from magnetically anchored satellite flown early this year. Blue lines show predicted performance envelope, while black lines show actual performance. Length of lines is measure of uncertainty of attitude sensor.

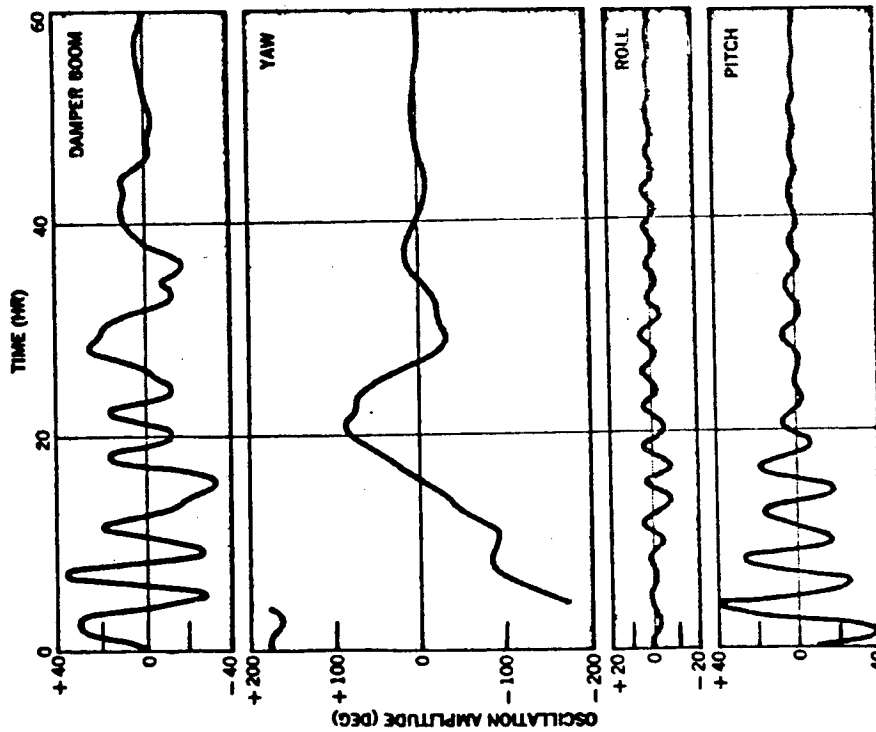


FIGURE 11-21. GRAVITY STABILIZED SATELLITE OSCILLATION DECAY
(Source: Ref. 2)

11.14 MAGNETIC TORQUING

Selective magnetization of long ferromagnetic rods can permit controlled torquing against the Earth's magnetic field. Orthogonal arrays of ferromagnetic and diamagnetic rods can permit passive, two axis stabilization to magnetic field. (Early "Transit" satellites used this method.) Magnetic torquing is used as one means for dumping momentum to damp gravity-gradient systems.

11.14.1 Accuracy

Accuracy is limited by the sensors more than by magnetic torquing. Continuous analog torquing permits infinite resolution within the range allowable by the Earth's field.

11.14.2 Limited Torquing Rates

Little can be done to increase torques. In principle, superconductive electromagnets might achieve gains through the intense fields possible, but the need for cryogenic temperatures makes the method generally impractical with today's technology.

11.14.3 Earth Field Compensation

The Earth's magnetic field is neither isometric nor constant. Open loop torquing would require calibration based upon position and stored map data on the field.

11.15 LOSSY SPRING DAMPING (OF GRAVITY-GRADIENT SYSTEM)

This is best explained by a comparison with other methods (quoting Reference 2).

"There are four basic force fields which can provide the anchor or referencing torque-- magnetic, gravity, solar wind, or atmospheric drag, the latter being found only at very low altitudes. Magnetic anchoring is achieved by using a freely-moving magnet as the heart of the damper reference mechanism. Solar and atmospheric anchors are based on the same principle, relying on a tear-drop-like configuration to point into the atmospheric or solar wind. This provides the damper reference torque.

Gravity anchoring (or lossy spring damping) is somewhat more complex. If a long rod in orbit is horizontal, it is in a position of conditional stability; that is, it has no torques acting on it. However, if it is disturbed from the local horizontal, it will seek local vertical. By coupling this horizontal boom to the main gravity-oriented body through a hinge and spring, the horizontal position can be made truly stable. By doing so, however, the natural frequency of this horizontal damper boom is made different from that of the main body, and any disturbance of the main body will couple into oscillations of the damper boom. Since the bodies have different natural frequencies, relative motion can be supported, and a damper in parallel with the spring can be made effective.

The relative strengths of the four reference force fields are of course not constant, depending on altitude. At 420 nautical miles, the aerodynamic field has decreased to the point where it is comparable to the solar field, but both are only one percent of the gravitational field. At 1,000 nautical miles, the gravitational field has become weak enough so that the essentially constant solar field is now becoming a more significant percentage of it.

But this isn't the whole story since we are really concerned with torques, not forces. By properly designing a vehicle, we can make the center of mass and center of pressure identical, thus eliminating solar torques. Or we can use bigger magnets to accentuate magnetic torques. In choosing a damper reference force field, then, we must know how to sensitize the vehicle to the desired orienting force, while minimizing its sensitivity to the other fields. For a gravity-anchored system, for example, we would need a magnetically clean vehicle and center of mass-center of pressure coincidence.

The gravity-anchored damper system has the highest oscillation decay rate for equivalent pointing accuracies. It is insensitive to orbit inclination and introduces no damper-induced errors. But the system is also one that is more complex and difficult to mechanize.

Magnetic orientation provides the simplest configuration, is the easiest to mechanize, and will work in the aerodynamic regime (below 500 nautical miles). But the greater the pointing accuracy required, the longer the decay time will have to be. Also, performance is sensitive to orbit inclination, being best in close to polar orbits.

Solar orientation, of course, is best when a subsidiary requirement is that some part of the vehicle point toward the Sun, for example, solar cells. But this requires a complex configuration, and there is a damper-induced error, causing the accuracy-decay time trade-off also found in magnetic orientation.

Aerodynamic referencing of course has to be used in the aerodynamic regime because the drag force is so overpowering, but it is probably a difficult concept to mechanize.

11.15.1 Attitude Interference

The lossy spring damper is not suitable for vehicles which require active attitude maneuvering. The damper booms interfere with stability of the active system.

11.16 AUTOMATIC GRAVITY LEVEL

This refers to a set of servo actuated screw jacks driven by signals from a pair of potentiometer levels or a pair of strapdown accelerometers. It permits adjusting the vehicle base to local horizontal upon landing. After leveling, the screw jack servos are shut off and the entire upper section of the vehicle turns in azimuth to acquire a reference star. It centers the designated azimuth star (at the prescribed time) then shuts down the azimuth drive. The vehicle now represents an aligned base for an equatorial clock drive telescope mount with right ascension and declination servo drives between the telescope and the sidereal table.

Leveling is the first necessary step of the automatic telescope alignment process.

11.17 SUN/STAR TRACKER

Suntrackers are chiefly used to roughly position solar cell panels (to about ± 1 degree of the sun line) in many vehicle designs; solar cells are the sensors.

In the AOSO concept, an extremely precise "rotary slab" sun tracker is being developed for controlling vehicle scan raster motions to ± 5 arc-seconds.

Two general types of star trackers are available, those which servo point a telescope's optic axis directly at the chosen star (± 5 to 15 arc-seconds) and photomultiplier image dissector trackers which provide coordinates of star position error off of scan center within their field of view. These latter can be made to operate at the diffraction limited resolution of the optics. They can yield star error signals to a small fraction of an arc-second. Combinations of the two types have been proposed.

11. 17. 1 Reliability Star Tracker

Servo slewed trackers have MTBFs in the 500 to 5,000 hour region.

Photomultiplier trackers have MTBFs in the 1,000 to 10,000 hour region.

Failures can occur in lock-on when vehicle vibration causes the star image to dance in and out of the field of view. Erosion of lenses by cosmic dust may be a problem but insufficient data is available to evaluate significance of this. Servo trackers are subject to the usual electromechanical problems and image dissector trackers are subject to the usual electronics limitations; scan can be disturbed by magnetic fields from nearby inductors; signal circuits are subject to errors from induced noise pulses.

11. 17. 2 Research and Development Required Sun Tracker

The rotary slab sun tracker for controlling AOSO scan is being developed, but as it is an electro/mechanical/optical device of high precision, it requires some time to come to maturity. Tests must be developed to prove performance and reliability. It is conceivable that an electronic sun scanning method may be developed to replace it.

The critical angle of incidence prism sun tracker developed by M. I. T. for Apollo G&N represents a fairly simple approach to a highly accurate sun tracker for navigation and attitude reference.

11. 17. 3 Research and Development Required Raster Motion

Two rectangular scan rasters are planned for scanning the solar disk: a small area scan (5 arc-minutes x 5 arc-minutes) and a large area scan of the entire solar disk. In both of these the present plan is to scan with vehicle attitude motions produced by reaction wheels or control moment gyros. If some simpler scan method can be developed, the vehicle's attitude control requirements perhaps can be relaxed.

11. 17. 4 Star Identification

There presently is no hardware optical system which can look at the sky, identify its own pointing attitude, then slew to track a particular star. Human beings and probably several birds and animals can perform this feat with little difficulty after training because of their capability for pattern recognition. Many constellation map matchers, astroglobes, star angle comparators, total sky scanners, Betelgeuse and Canopus identifiers, etc., have been proposed, but for some reason or other each has been side tracked.

Mariner depends upon performing a roll motion around the sun line to pick up Canopus within a conical scan at an angle which depends upon celestial longitude of the vehicle. Although, with much coaxing from DSIF the system is workable, and mechanically simple, it could hardly be termed totally automatic.

This technical deficiency is related to one titled "R&D Required, Pattern Recognition".

11.18 HORIZON SCANNER

Refer to discussion in Section 4 (Guidance and Navigation).

In many of the concepts where this term is called out, the device really intended is a planet tracker-scanner. The horizon scanner serves only to define "local vertical" at altitudes within less than 20 planetary radii. The planet tracker scanner provides local vertical to the planet from early mid-course to touch down; during late mid-course and approach, it also provides a measure of subtended visual angle. From this, altitude and altitude rate can be estimated.

Horizon sensors or scanners with a resolution of ± 0.1 degree (or worse) have been used as a means for stabilizing satellites to local horizontal in orbit. Better accuracy is much needed for navigation as well as S/C, and serious research to improve knowledge of the horizon has begun.

Many planet tracker-scanners have been proposed; none have been developed.

11.18.1 Reliability, Horizon Scanner

MTBFs range from 300 to 3,000 hours. Some missions will require MTBF of 2 to 5 years.

11.18.2 Accuracy, Horizon Scanner

Horizon scanners are limited by variation of definable horizon to about ± 0.1 degree in Earth orbit. Needed is local vertical reference to ± 0.1 arc-second or better for navigation in orbit.

11.18.3 Horizon Definition, Mars

It is conceivable that due to lower atmospheric density, the horizons of Mars will be more stable, and that eventual local vertical visual or IR reference to about ± 5 arc-minutes will be possible after the exact shape of Mars has been determined.

11.18.4 Horizon Definition, Venus

Quite probably the local vertical reference to Venus will be worse than ± 0.25 degree, due to its dense atmosphere and the uncertainty of its thermal balance.

11.19 ATTITUDE GYRO

Refer to paragraph 11.1.5.

11.19.1 Reliability, Gyro

MTBFs of strapdown attitude reference subsystems, based upon miniature integrating rate gyros, can exceed 3,000 hours. Gyro MTBFs in excess of 10,000 hours have been quoted.

11.20 RATE GYROS

The primary sensing means in the majority of S/C subsystems is the miniature, floated gimbal rate gyro. Tiny two ounce "low cost" rate gyros used in today's space vehicle S/C subsystems can outperform the best quality gyros available in World War II.

11.20.1 Threshold

The hour hand on your watch turns 360 degrees in 12 hours (0.0083 degree per second). The minute hand moves 360 degrees in one hour (0.1 degree per second). Present rate gyros can respond to rates as low as 0.01 degree per second.

Stabilization of vehicle rates to drift rates lower than 0.01 degree per second will require a major advance in the rate sensing art.

11.20.2 Reliability, Gyro

MTBFs as high as 10,000 hours have been quoted. Guaranteed minimum lifetimes of 1,000 hours are common practice.

11.21 HOLLOW SPHERE

Refer to paragraph 11.10 "Reaction Wheels" for comparative comment (wheels versus spheres versus gyros).

11.21.1 Weight

Total weight of a reaction sphere subsystem is dependent chiefly upon the maximum torquing rate required of the vehicle. For the "Manned Telescope Satellite" the subsystem probably would weigh less than 100 pounds, including a 20-pound sphere suspension and centering electronics and torquing electronics. High slewing rates would require reaction jets. The chief virtues of the reaction sphere are its infinite torque resolution, its independence from propellants, its zero cross-coupling effects and the possibility for executing whatever small attitude and body rate changes can be sensed.

11.22 EARTH/MOON SEEKER

The Earth/Moon "trojan" points are located at the apexes of two equilateral triangles (in the Moon's orbit plane) whose common base is the Earth-Moon vector. It has been proposed that satellites trapped in these libration points be used as Earth/Moon communication relay satellites for observatory stations located just behind the lunar limb (and invisible from the Earth).

The only attitude control requirement of such satellites is that they point antennas (± 5 degrees) at both Earth and Moon continuously.

Using IR horizon sensors (initially erected by on-board TV from Earth) this antenna pointing is quite feasible.

11.22.1 Reliability, Earth/Moon Seeker

The loss of local vertical to either Earth or Moon could make the relay satellites inoperative, requiring replacement of the entire satellite. Ideally, five years of uninterrupted operation is desirable.

11.23 RADIO DIRECTION FINDER

RDF (radio direction finder) is suggested to provide beacon homing for the parachuted payload of high altitude research rockets.

An alternative method for controlling the relay antennas discussed in paragraph 11.22 is to cause the satellite to stabilize upon Earth and Moon beacons with a pair of Marconi antenna radio direction finder (interferometers). The antenna beam zeros would thus be centered on the stations of interest rather than just the planets.

11.23.1 Reliability, Radio Direction Finder

Having no moving parts and being very old in the art, RDF systems of the Marconi type are well developed. They should be considerably more reliable than IR trackers.

11.23.2 Low Cost

For the parachute homing application, an adequate Marconi type RDF probably could be made to commercial toy-class standards for less than \$20 each. Signals from RDF would control the capstan servos (paragraph 11.8). Low cost is needed to make the entire system "expendable".

11.23.3 Size

It is estimated that the RDF for the satellite application would occupy four cubic inches (plus loop antennas), weigh three pounds and draw one watt of power. For the parachute system the RDF could be packaged in about two cubic inches, weigh about six ounces and draw about 100 milliwatts. The loop antennas would be wound of litzendraught wire on paper strips wrapped orthogonally around the instrument package.

11.23.4 Standby Reliability

The parachute RDF along with the entire rocket system would be stored in quantity and used as needed, like a transistor radio in a wholesaler's warehouse for several years-- it should "play" whenever it is turned on. Commercial, low cost, "toy" equipment traditionally has been more reliable in this respect than equipment built under Military Specification constraints. (Probably because volume production permits design optimization.)

REFERENCE LIST FOR SECTION 11

- 11-1 Brian F. Doolin, "Space Vehicle Attitude Control," Proceedings of the NASA-University Conference on the Science and Technology of Space Exploration, Chicago, Ill, 1-3 November 1962, Volume I, pages 381 - 386, NASA SP-11.
- 11-2 Richard J. Katucki, "Gravity Gradient Stabilization," Space/Aeronautics, October 1964.
- 11-3 John V. Foster, "Guidance and Control Components Research," pages 387 - 392, NASA SP-11 (see Ref. 1).
- 11-4 Dr. H. V. Noble, "Microelectronic Progress," Ordnance, May-June 1965.
- 11-5 "Aerospace Technical Forecast 1962-72," Aerospace Industries Association.
- 11-6 William R. Corliss, Propulsion Systems for Space Flight, McGraw-Hill, 1960.

Section 12

SIMULATION AND TRAINING

12.0 GENERAL COMMENTS

Simulation and training as defined for this study encompasses all equipment functions required to simulate actual mission conditions and equipment operations needed to train and maintain proficiency of crew members and maintenance personnel in the actual functioning of the system and subsystems.

A breakdown of comment by configuration and TD's was eliminated from this section in view of the little effect upon voter decisions of the whole subsystem area in the presently configured tree.

While the success of a mission will depend largely upon the quality of previous training and orientation of the personnel involved, it is doubtful that this technological area is likely to lead to in-flight experiments on Apollo. The chief technological problems at present are in simulating all environmental conditions simultaneously and in adequate sized test chambers. The most difficult factor of course is simulation of zero g.

The following is quoted from Ref. 1.

12.1 TRAINING SYSTEMS - MAN IN SPACE

Space flight trainers will receive more attention as space operations become more extensive. In contrast to aircraft practice, more space crew training will be conducted in flight simulators on the ground. Tremendous originality and design skill will be needed to construct machines which can truly perfect crew skills. Electronically operated displays and controls in the simulators will closely reproduce all normal operations and system malfunctions which could occur in space vehicles. There is a growing trend toward the use of digital computers to describe the dynamic equations of motion, but the requirement most often expressed is for analog to digital and digital to analog conversions to permit simulation of complex flight characteristics with greater fidelity. Additional study will be needed to establish the fidelity of simulation necessary to achieve desired research and training objectives.

As we progress from unmanned and minimally manned Earth-orbital and deep-space vehicles, to more manned missions and larger crews, there will be increasing requirements for space simulators, training for personnel performance tasks, and use of life support systems for space flight. Personnel response under high accelerations and zero gravity will continue to be simulated in such equipment as the NADC Centrifuge, Pensacola Disorientation Trainers and that available at the Ames Research Center. Training simulators which provide combined environmental stresses will become more important. Electronic equipment will be required for presentation of stimuli and analysis of the pilot's response, as well as the pilot's physiological status.

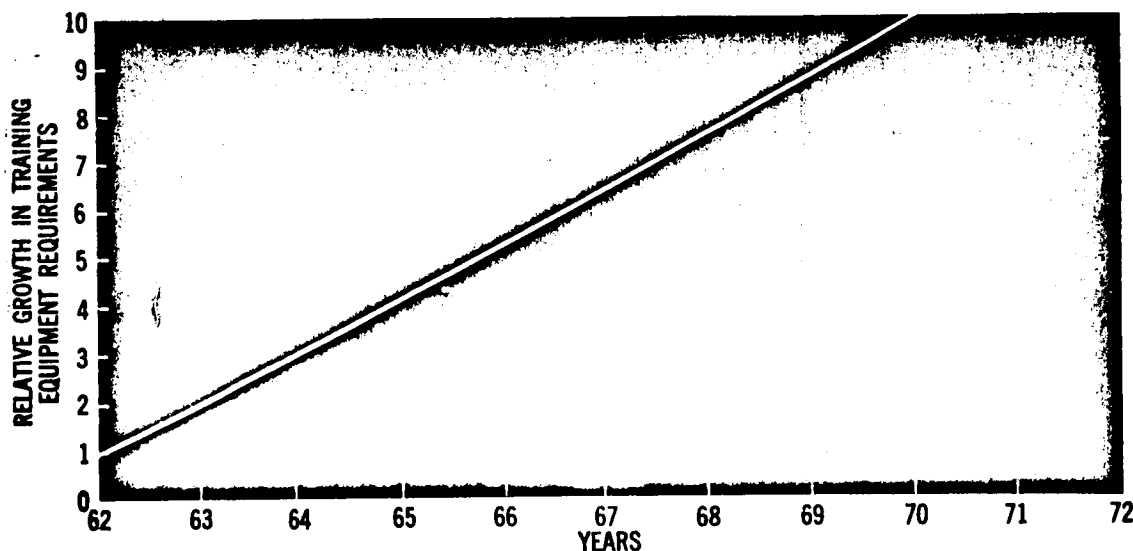


FIGURE 12-1. TRAINING REQUIREMENTS TREND
(Source: Ref. 1)

12.2 TRAINING SKILLS FOR THE HUMAN OPERATOR

Space flight, nuclear technology, and other scientific and engineering advances will increase the demand for human skills in operation and maintenance.

The first manned space vehicles could be operated by automatic systems during an entire flight, with the pilot serving primarily as a backup to the automatic controls. However, space vehicle design is moving rapidly away from this concept toward aircraft philosophy which calls for the pilot to be assisted by automatic devices, but to remain as a vital control and decision-making element in every phase of flight. Pilot control of all

phases of space flight, including booster launching into orbit, orbital flight and reentry is expected within this forecast period.

The long duration of most space missions compared to aircraft flights adds a major new skill to the requirements which an astronaut must fulfill. Space crews must perform maintenance and repair work on their vehicle's systems on flights lasting for weeks or months. Great research effort is needed to find the optimum balance between human and automatic control systems. This balance is a key factor in designing spacecraft of minimum weight and lowest cost.

12.3 SIMULATION FACILITIES FOR MAN-VEHICLE EVALUATION

As the manned space flight era progresses, more programs involving man-machine relationships become desirable. With the decision to advance our lunar landing attempt, a concentration of effort has begun in the necessary support areas. The use of high-fidelity flight simulation techniques for research and training purposes has become more necessary than in the past. More knowledge is needed of man's ability to control and navigate advanced vehicles and to cope with emergency situations, while under the combined stresses of space environments. Considerable data are available on man's capacity to perform under the stress of high acceleration, buffeting, temperature extremes, and certain ranges of atmospheric composition. More data are needed on his ability to monitor displays; to perform cognitive tasks under conditions of monotony, weightlessness, and sensory deprivation; and to control vehicular flight in the face of rapidly changing conditions. For such studies, real-time flight simulators are needed. Even for those areas of performance which are adequately known today, additional facilities will be needed for training and selection purposes.

Figure 12-4 shows the approximate needs for evaluation/training devices in eight categories over the next 10 years. The heavy curve is a weighted average of the individual curves, and takes into account relative expenditures in the various categories. Of necessity, the curves are only an approximation. It predicts that the present rate of expenditure will multiply by four or five times in the next four years. Any significant change in national space objectives, or presently unforeseen technical difficulties or breakthroughs, could drastically alter the shapes and magnitudes of the curves.

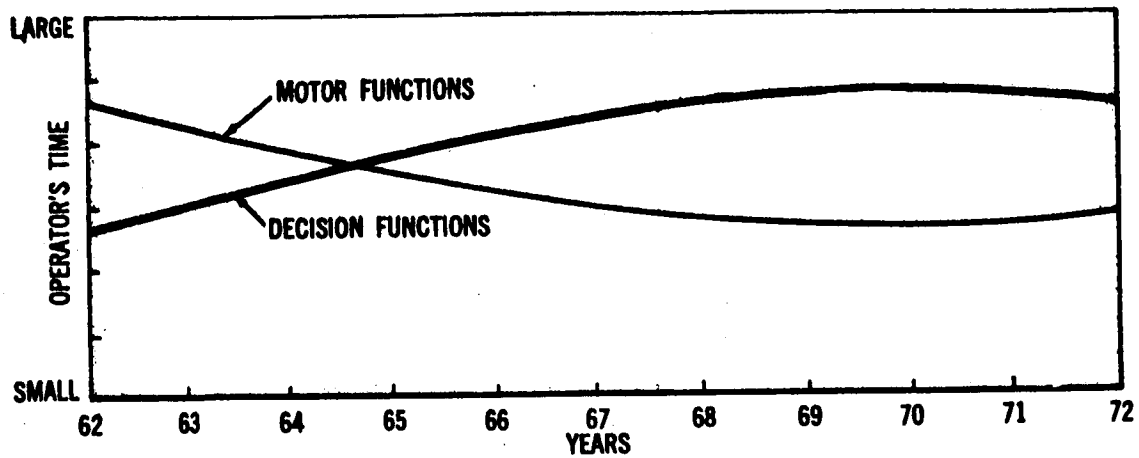


FIGURE 12-2. USE OF OPERATOR'S TIME
(Source: Ref. 1)

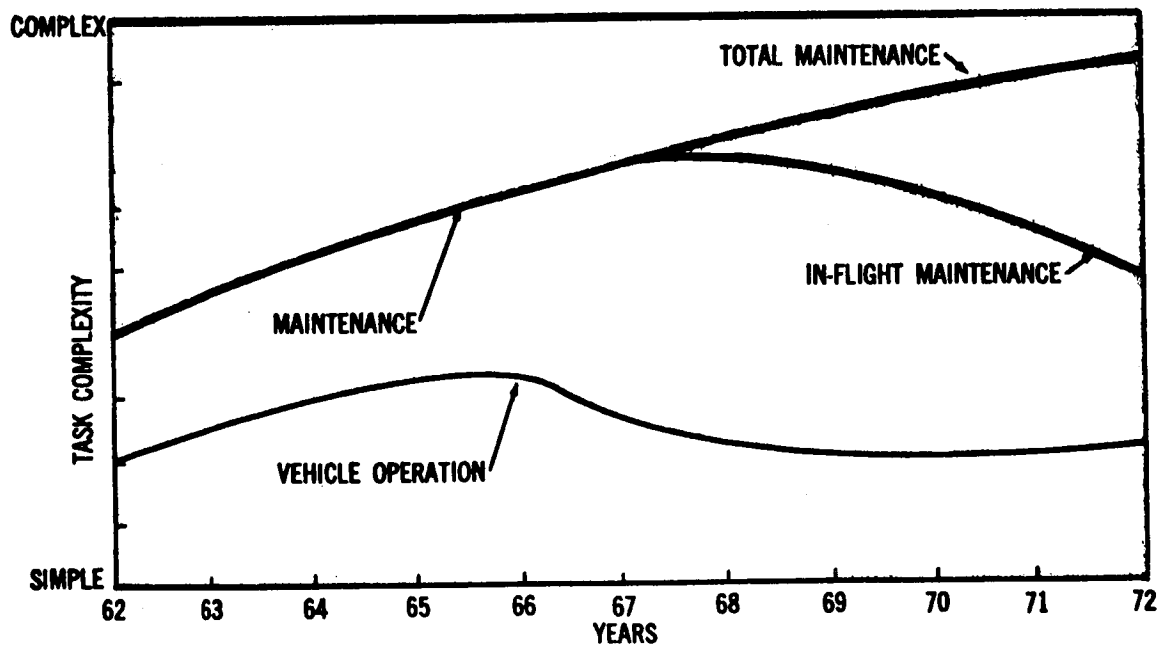


FIGURE 12-3. COMPLEXITY OF TASKS
(Source: Ref. 1)

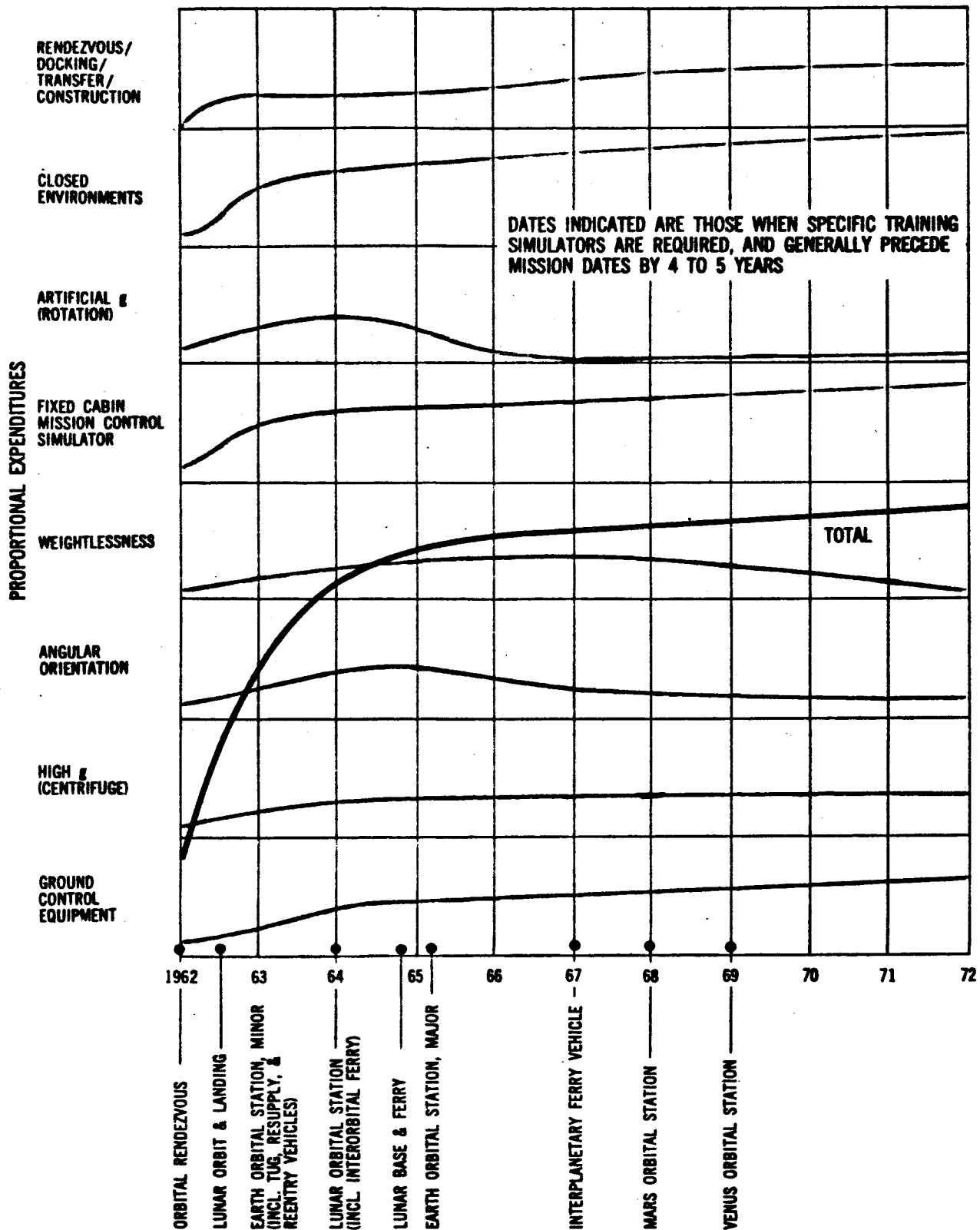


FIGURE 12-4. ESTIMATED YEARLY EXPENDITURES
FOR MANNED AEROSPACE SIMULATION DEVICES

(Source: Ref. 1)

12.4 FLIGHT SIMULATION AND GUIDANCE TESTS

More realistic physical environment simulation for the crew will be necessary to perform man-machine flight simulation tests. This simulation will include improved displays, improved application of external environments, and more realistic representation of instrumentation transients and accelerations. Greater use of real-time digital techniques in the computer portion of such tests are required by the increased control precision called for on the next generation of aerospace vehicles.

12.5 RE-ENTRY TRAINING

Re-entry drills during long flights may be essential to assure that astronauts retain the required skill necessary for their survival at the end of the mission.

12.5.1 Skill Retention

The manipulative skills may tend to diminish after several months in space. Onboard training simulators may be needed.

12.6 INFLIGHT SKILL RETENTION

Same as above.

12.6.1 Crew Mix

This refers to the problem of providing the optimum variety of skills and professions among a limited number of crew members.

12.6.2 Redundancy

If a particular skill is essential, then loss of the only man possessing it could be catastrophic. Multiple training permits back-up of each skill by another crew member.

12.7 ASSEMBLY TRAINING

Structures assembled in orbit must go together right on the first try. Simulation by such means as underwater assembly rehearsals is only partly effective.

12.7.1 Realism of Zero-g Work

Zero-g for extended training drills will be nearly impossible to simulate.

12.8 REFERENCE FOR SECTION 12

- 12-1 "Aerospace Technical Forecasts, 1960-1970 and 1962-1972,"
Aerospace Industries Association, 1725 DeSales Street N. W.,
Washington 6, D. C.

Section 13

VEHICLE AND STRUCTURE

13.0 GENERAL COMMENT

Vehicle is defined in this study as the framework and associated structural members that perform a carrying or platform function such as the booster structure, spacecraft airframe, roving vehicle, lunar base, etc.

Since vehicle structure itself is subject to infinite variation in detail, complete coverage cannot be expected. Vehicle structures in general are classified into types as follows:

Open frame (Mariner type).

Solid projectile with cavities.

Monocoque frame and hull (Apollo LEM for example).

Ablative shielded re-entry body (Apollo CM for example).

Inflated structures (Echo I for example).

Metallic balloon (Atlas fuel tanks for example).

Wound fiber.

Airplane type.

13.0.1 Environment

The following is quoted from Ref. 1:

A primary objective of space exploration is to perfect a definition of the space environment. Much more must be known of cosmic radiation, solar storms, meteoroid showers, the dust cloud around the sun and many other features of the natural environment above the atmosphere before man can venture with confidence to altitudes above 300 miles and eventually on the moon. In addition to natural phenomena, an

aerospace system is influenced by an environment it creates for itself-- often called the induced environment. Rocket engine vibration and boundary layer noise are examples. It is often difficult to distinguish between the effects of natural and induced environments, but together they determine the form and operating characteristics of aerospace vehicles.

Broad environmental trends are presented in these pages. It is probable that some of the predicted environmental values will change in the near future as the pace of space research and manned space operations increases.

Figures 13-1 through 13-3 show a graphical model of the assumed natural environments in near-Earth space.

Environmental Factors Affecting Design. The range of operational environments (natural and induced) which must be considered during the next 10 years extend from beneath the ocean's surface, through the Earth's atmosphere, through outer space, to the surface of the Moon and other planets.

Table 13-1 indicates the environmental factors which the designer of aero-space systems must consider for typical operational regimes. The anticipated range of values for many of these environmental variables is described in the pages which follow the table.

13.0.2 Natural Environments

Earth's Natural Environments. The variables of temperature, pressure, and contaminants such as rain, hail, sand, fungus, humidity, salt spray, etc. in the lower atmosphere of the Earth are described in detail in MIL specifications and it is improbable that their values will change appreciably in the next decade. Another of the Earth's natural environments, the ocean, is expected to have an increasingly significant influence on the design of launch and recovery systems. The variables involved such as temperature, depth, chemical content, currents, etc. have been the subject of considerable study, however, there is need for continued effort to obtain more accurate information.

Extremely Low Pressure. The Earth's atmosphere is considered to extend to 10 miles altitude. Between 10 and 120 miles above the Earth's surface, the pressures represent a partial space equivalent; and for most considerations, above 120 miles the pressures encountered constitute a total space equivalent. At altitudes above 120 miles extreme low pressures

GALACTIC COSMIC RADIATION:
1-10⁹ Bev./Nuclear: 85% protons.
14% α particles, interacts with at-
mosphere to form many new particles
plus photons.

SOLAR WIND:
50ev to 20 KEV at $\sim 10^4$ to 10^6 in.³

TOTAL SOLAR FLUX:
130 watts/ft.²

RADIATION TEMPERATURE SINK OF SPACE $\approx 7^{\circ}\text{R}$

VAN ALLEN RADIATION BELTS:
Inner Region--while relatively quiescent, observed to change in intensity by factor of two or three in sixteen months, associated with major geomagnetic and/or solar proton events of impulsive character.

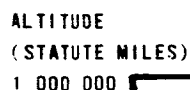
Outer Region--subject to changes by factors of 10 to 100 in a few days. Outer boundary normally at geocentric distance of 11 earth radii ($R_{\oplus}$) measured in electrons > 50 KEV, although can be "pressed in" to $8 R_{\oplus}$, counting rates subject to $\approx$ sharper peaks in high energy region of spectrum with maximum around $3.1 R_{\oplus}$.

ATMOSPHERIC IONIZATION
~10² TO 6x10⁷ PARTICLES. IN³

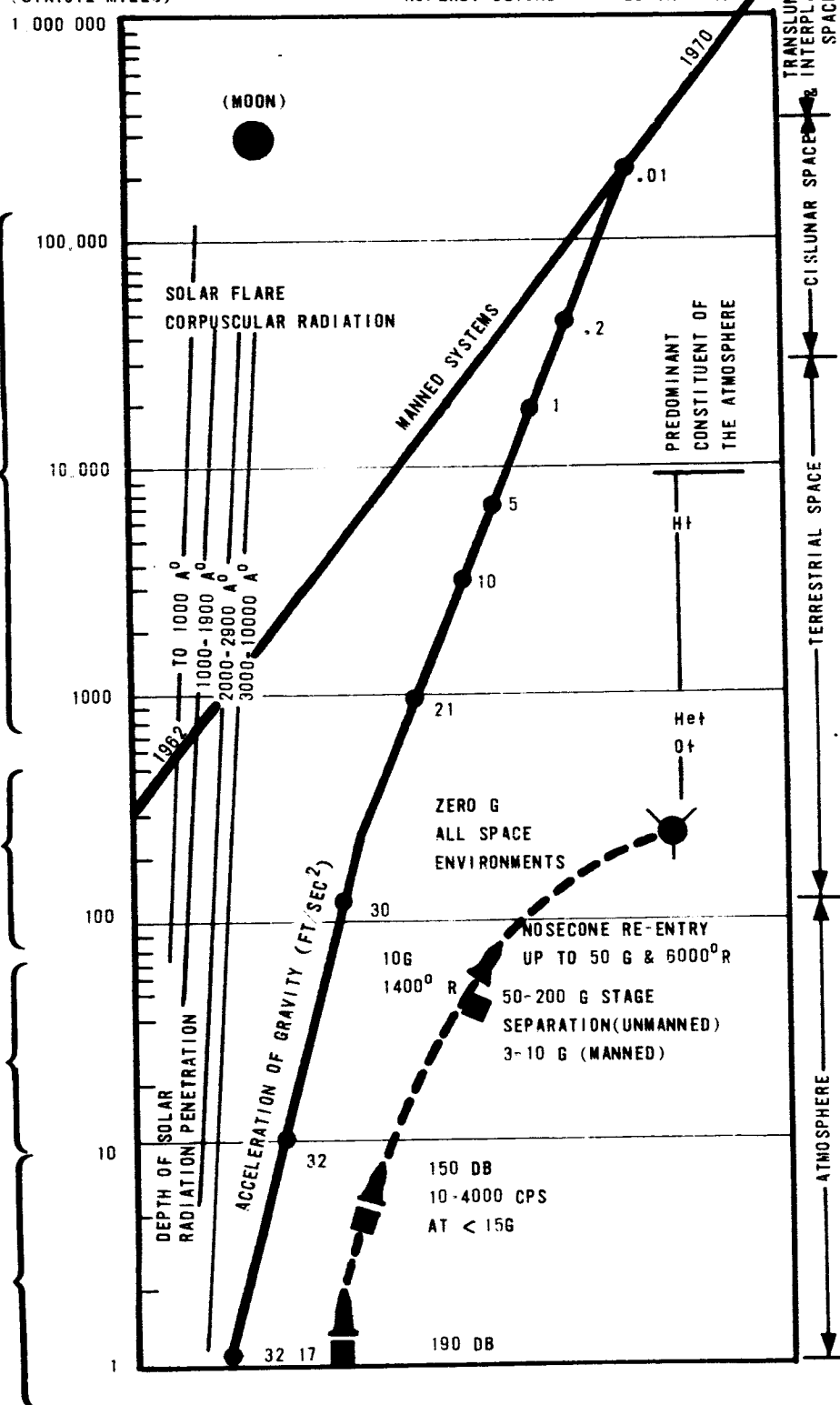
**FREE MOLECULAR MOTION:
THERMODYNAMIC, RAM AIR, ACOUSTIC
PROPAGATION, AND AERODYNAMIC
EFFECTS BECOME NEGLIGIBLE**

OZONE LAYER (O_3) UP TO ≈ 12 PPM

**TROPOSPHERE ENVIRONMENTS:
HIGH AND LOW TEMPERATURE
DYNAMIC PRESSURE
HUMIDITY, SAND, DUST
SHOCK, VIBRATION, ACOUSTICS
ACCELERATION**



UNMANNED SYSTEMS FUNCTIONING
PROPERLY BEYOND 10 MILLION MILES



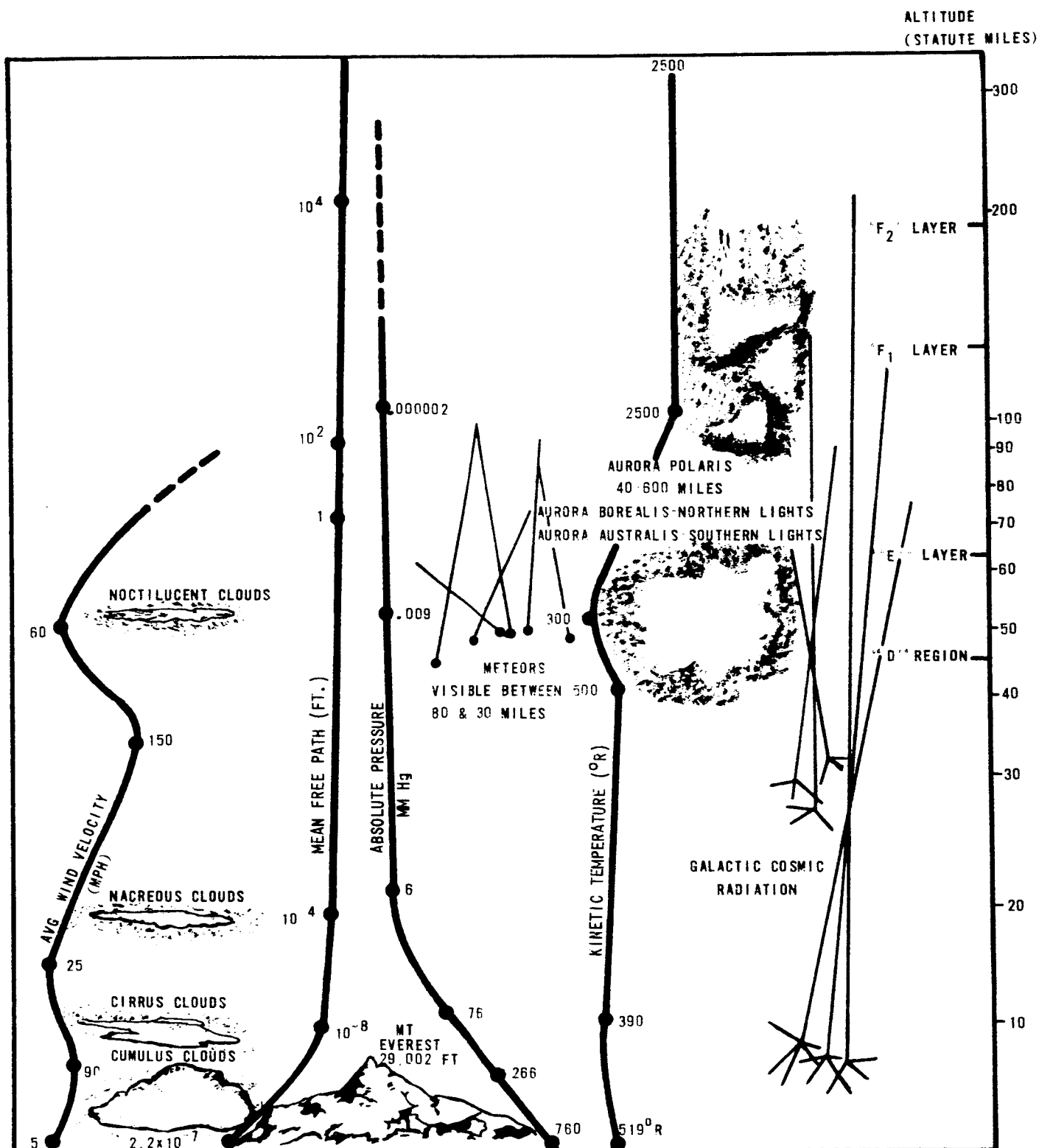
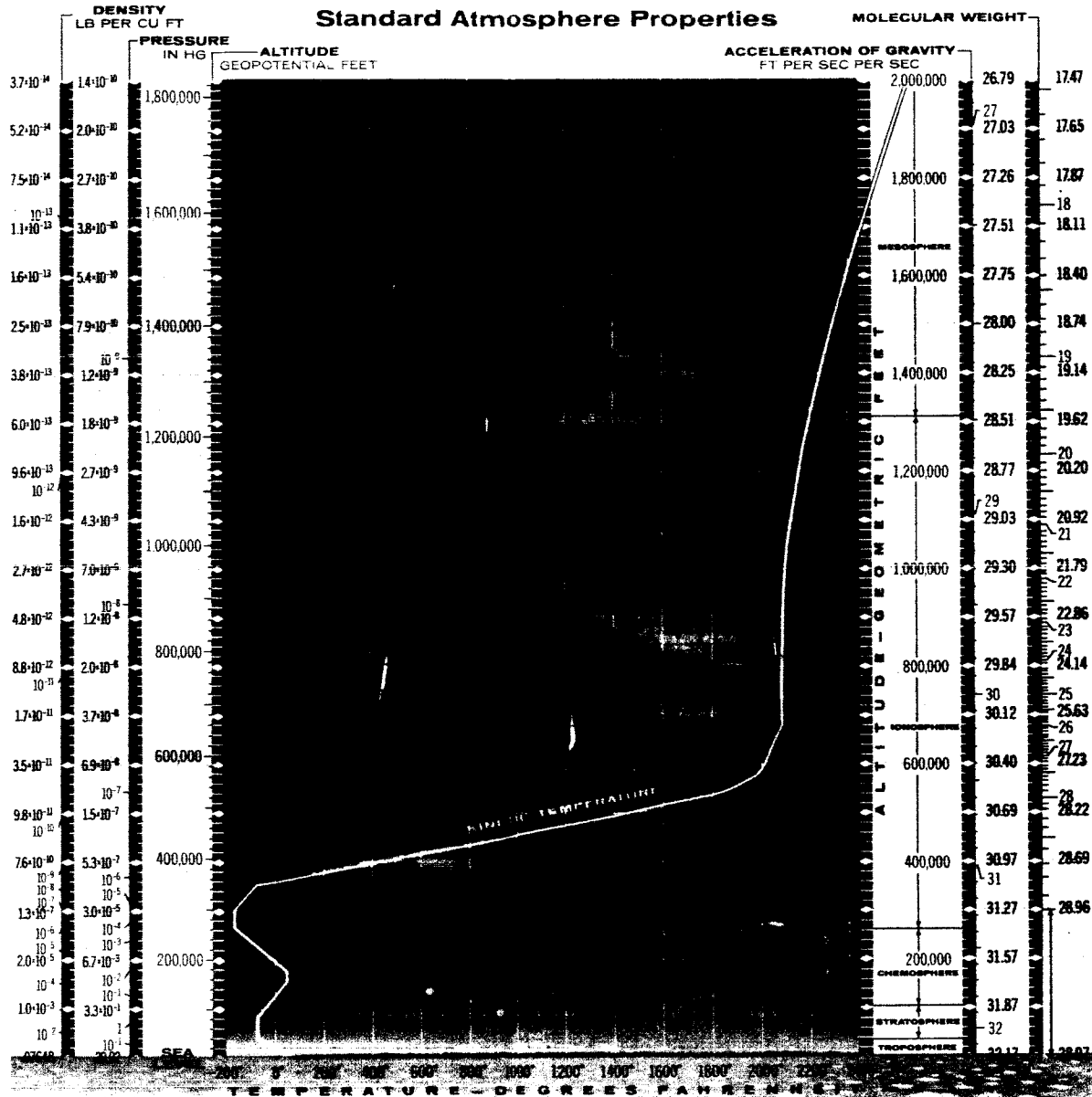


FIGURE 13-1. AEROSPACE ENVIRONMENTS

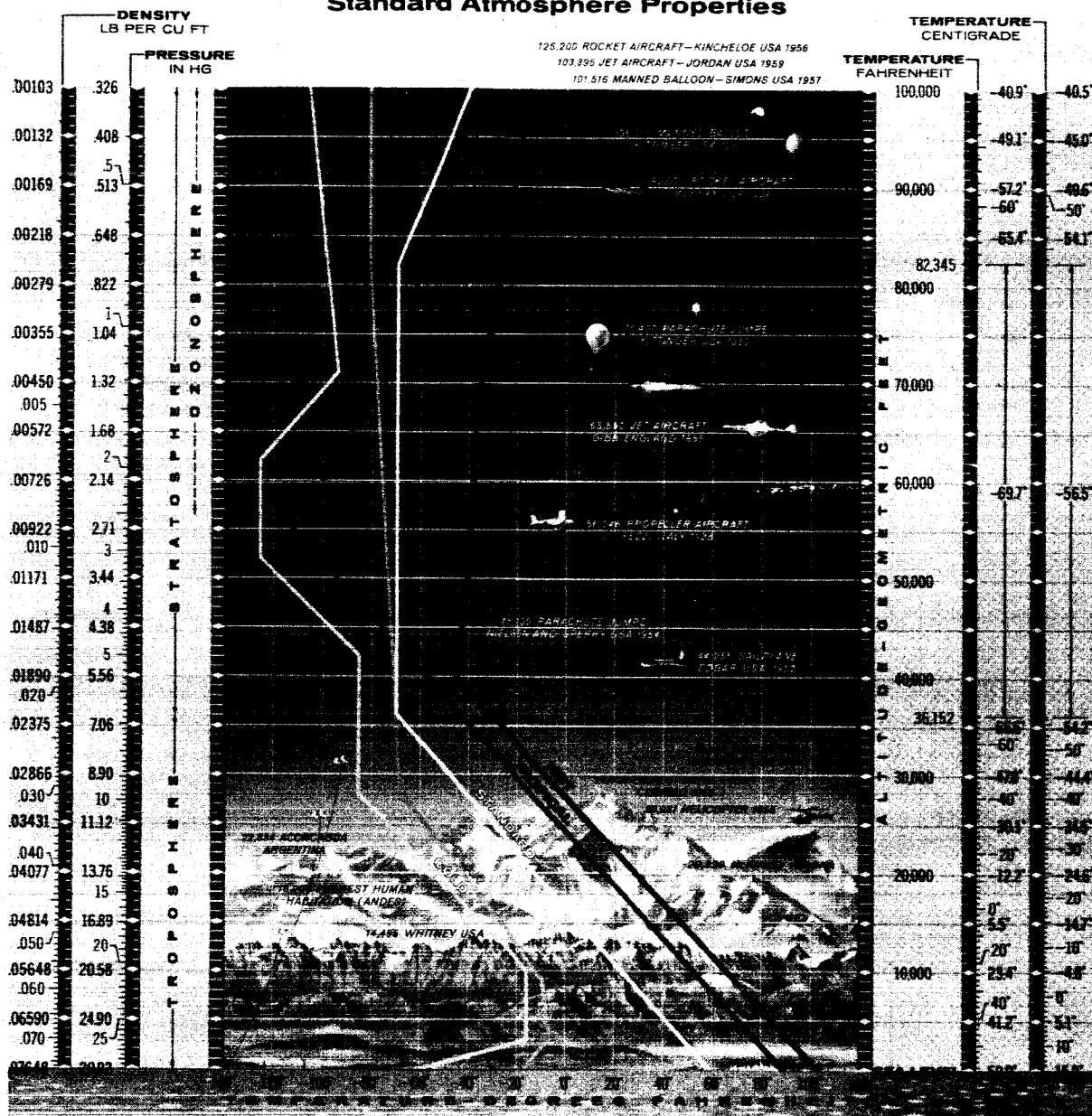


This chart presents standard values for the temperature, pressure and density of the air, based on the ARDC Model Atmosphere, 1959. The altitude column on the right presents geometric altitude. The energy required to lift an object 2,000,000 geometric feet is only 1,824,929 times that required to lift it one foot above sea level, because of the decrease in the acceleration of gravity with altitude. This relative amount of energy is shown in the column on the left as geopotential altitude. Kinetic temperature is a measure of the kinetic energy of

the molecules and atoms constituting the atmosphere. Its numerical value is determined by the assumed molecular weight of air as well as the assumed temperature lapse rate. The molecular weight of air is assumed constant from sea level to 299,518 ft. Its decrease above 299,518 ft. is attributed to the dissociation of oxygen and nitrogen. The molecular weights tabulated probably represent the distribution at high latitudes, such as in Northern Canada. The symbols D, E, F and G refer to the respective ionized regions of the atmosphere.

FIGURE 13-2. HIGH ALTITUDE CHART

Standard Atmosphere Properties



This chart presents the latest recognized standard values for the temperature, pressure and density of the air at altitudes up to 100,000 geometric feet. The pictorial inserts represent various steps in man's achievement in learning to reach and to exist at high altitudes. The data in the columns on the sides, and the central profile, all refer to the ARDC Model Atmosphere, 1959. The NACA Standard Atmosphere, 1955, and the ICAO Standard Atmosphere, 1954, are essentially consistent with the ARDC Model Atmosphere up to 66,008 geometric feet. The hot and cold

temperature profiles represent U.S. Military extreme maximum and minimum temperature criteria for the Northern Hemisphere. As defined in MIL-STD-210, it is assumed that the extremes will not be exceeded more than 10 per cent of the time. The tropical and polar temperature profiles represent U.S. Military standards for hydrodynamically consistent homogeneous atmospheres for the tropical and polar regions as defined in MIL-STD-210. The pressure and density values shown in the columns are consistent with the standard temperature profile only.

FIGURE 13-3(A). ATMOSPHERE CHART

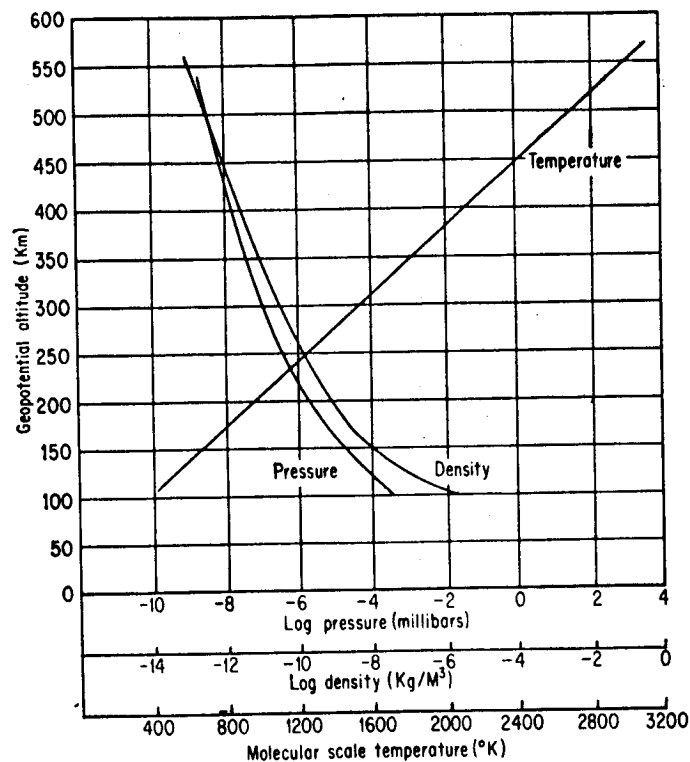


FIGURE 13-3(B). PROPERTIES OF THE EARTH'S
UPPER ATMOSPHERE ABOVE 10 KM
(Source: Ref. 3)

(Adapted from Smithsonian Contributions to Astrophysics, Orbital Data and Preliminary Analyses of Satellites 1957 α and 1957 β)

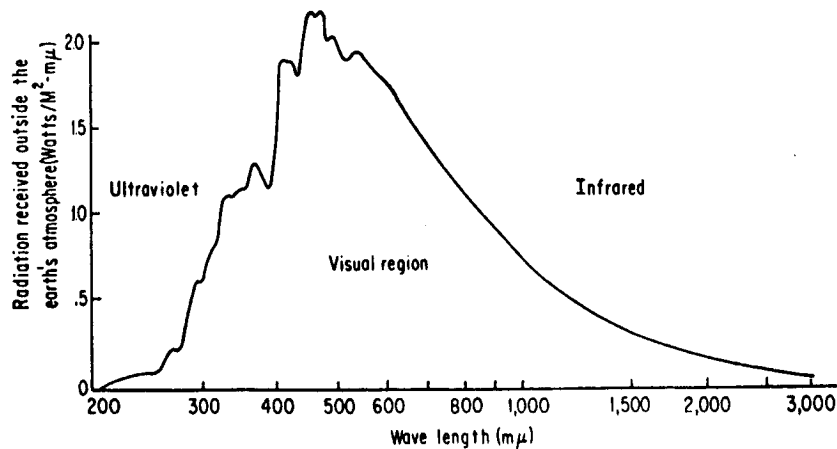


FIGURE 13-3(C). THE SPECTRUM OF SUNLIGHT
ABOVE THE EARTH'S ATMOSPHERE
(Source: Ref. 3)

TABLE 13-1. OPERATIONAL REGIME

Environment	Terrestrial			Transit ¹ (Launch & Re-Entry)	Space		Lunar Landing and Relaunch	Planetary Land- ing and Relaunch	Individ. Component Cond'n'd Env'ment
	Storage & Handling	Test & Check- Out	Earth's Atmos. Flight		Earth Satellite	Deep Space			
1. THERMAL									
a) Heat Flux			X	X	X	X	X	X	
b) Thermal Shock			X	X				X	
c) Temperature	X	X	X	X	X	X	X	X	X
2. RADIATION									
a) Van Allen					X	X		X	
b) Cosmic					X	X			
c) Nuclear, Man-Made					X	X	X	X	
d) Solar					X	X	X	X	
3. PARTICLE COLLISION, MECHANICAL									
a) Dust Clouds					X	X	X		
b) Meteoroids					X	X	X		
4. VACUUM			X	X	X	X	X	X	
5. MECHANICAL ACCELERATION									
a) Vibration	X		X	X	X	X	X	X	X
b) Shock	X		X	X			X	X	X
c) Acceleration	X		X	X	X	X	X	X	X
Weightlessness					X	X			
d) Acoustical Noise			X	X				X	
6. AERODYNAMIC									
a) Perturbations Affecting Stability & Control			X	X				X	
b) Plasma				X				X	
c) Wind, Wind Gusts & Wind Shears	X	X	X					X	
7. ELECTROMAGNETIC									
a) Geomagnetism			X	X	X				
b) Electric Static Chg. on Vehicle			X	X	X	X	X		
c) Radio Noise		X	X	X	X	X	X	X	X
8. WEATHERING & AGEING									
a) Humidity	X	X	X						
b) Salt Water	X	X		X					
c) Fungus	X								
d) Temperature	X	X	X		X	X			
e) Sunlight	X	X	X		X	X			
f) Sand & Dust	X	X	X				X		
g) Oxidation	X								
h) Outgassing					X	X			
i) Rain and Snow	X	X	X						
9. HYDRAULIC PRESSURE				X					

(Source: Ref. 1)

range from 1.5×10^{-6} mm Hg down to an estimated value of 10^{-11} or 10^{-12} mm Hg for solar system space and 10^{-15} to 10^{-20} mm Hg for intergalactic space.

Electromagnetic Radiation. Electromagnetic solar radiation is a continuum over virtually the complete range of wavelengths with a spectrum and energy output closely approximating that of a 6000°K black body. From $2000\text{\AA}$ down to $200\text{\AA}$ (vacuum ultraviolet) line spectra predominate. Ultraviolet radiation below $2000\text{\AA}$ represents a small part of the total solar radiant energy, but the higher quantum energies of this radiation can activate chemical reactions not encountered with the longer wavelength radiation. By far the major portion of the sun's energy is radiated between $2000\text{\AA}$ to $32,000\text{\AA}$ with the peak occurring around $4550\text{\AA}$ (blue). Radio frequency radiation is primarily of spatial origin, and its spectrum and intensity vary considerably with solar and stellar activity. The high quantum energy X-ray and gamma radiation of varying intensities emanate primarily from the attenuation and decay of high energy particles.

High Energy Particle Radiation. High energy particle radiation encountered in heliocentric space consists primarily of solar protons and electrons, and galactic cosmic rays. This radiation is absorbed by the Earth's atmosphere, but secondary radiation from cosmic rays is detectable on the Earth's surface. Two or more bands of magnetically trapped protons and electrons have been detected beyond the Earth's atmosphere. The outer band has mostly low energy particles, probably coming from the sun. A portion of the inner band has high energy particles which may result from decay products of cosmic ray collisions with atmospheric gas molecules. Cosmic rays consist of protons, alpha particles and heavier nuclei. These particles, originating from outer space, have low intensities, but extremely high energies. Their interaction with the atmosphere or other matter produces secondary particles and electromagnetic radiation. A similar reaction is believed to cause the auroral displays, when radiation particles collide with the upper atmosphere in the polar regions.

Dissociated and Ionized Gases. Dissociated and ionized gases are significant mainly because their chemical activity differs markedly from that of gases in their normal state (e. g., change of oxidation rate), and because they interfere with radio transmission. Such excited state reactions can occur thermally, electrically, or by the direct absorption of photons. There is a concentrated band of ozone in the atmosphere between 50,000 and 100,000 feet with maximum concentration between 70,000 and 80,000 feet. This component can present a serious material and personnel hazard in high altitude supersonic flight.

Meteoroids. Extraneous solid matter in space near the Earth varies over a wide range of size, mass, density and velocity. Information to date on the meteoroid environment is rather meager, but it has been estimated from satellite data during the past year that 2,000 tons per day enter the atmosphere. Published data are primarily the result of mathematical extrapolation based on several assumptions. It is known that most particles possess small size and mass, reaching a lower limit of approximately 10^{-12} grams and 10^{-4} cm diameter. Three classes of meteoroid composition are generally accepted: iron-nickel, stoney, and pithball (dust and frozen gases), with respective densities of 8, 3, 4, and 0.05 to 0.6 grams/cc. The overall meteoroid velocities in the vicinity of Earth range from 7 to 45 miles/second.

Penetration and erosion are the significant effects of spatial debris on space vehicles. The spatial density of this material is relatively low, reducing the risk of penetration to reasonable limits. Figure 13-4 gives current estimates of such risk for nominal aluminum skin thicknesses. Limited data from satellites are consistent with these estimates. The relatively wide range of values is due to present uncertainty about the impact phenomenon of hypervelocity particles. The upper limit of the curve assumes a direct conversion of the solid particle kinetic energy into heat energy on impact with the target material. This approach is believed to be conservative by a factor of three or four. The lower limit is based upon a numerical approximation of the equation of motion at impact, and appears to yield more reasonable data. However, there is very little substantiating test data.

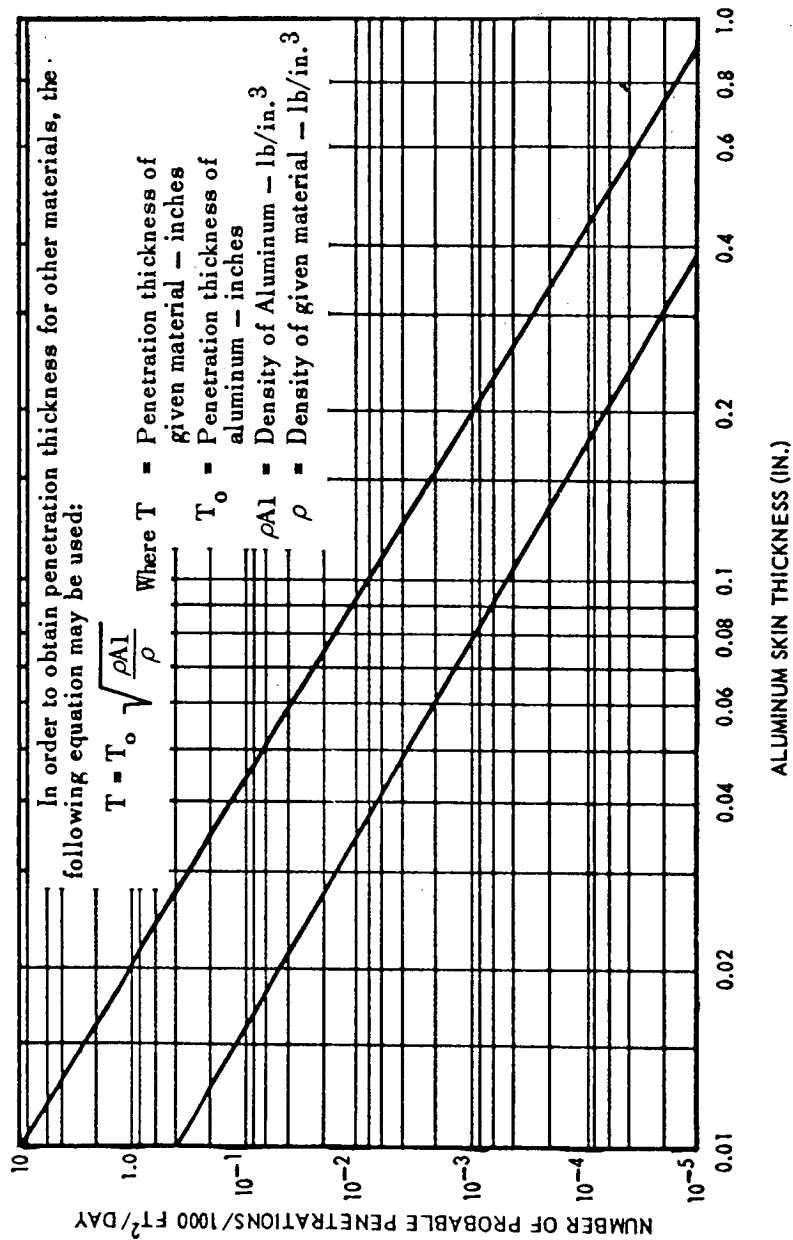


FIGURE 13-4(A). METEORIC PENETRATION RISK FOR SPACE VEHICLES
 (Source: Ref. 1)

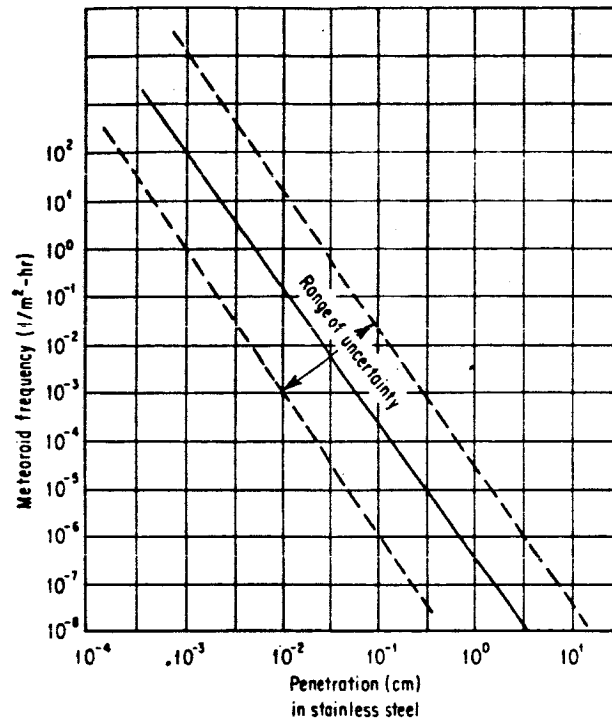


FIGURE 13-4(B). METEOROID FLUX VERSUS PENETRATION
DISTANCE IN STAINLESS STEEL
(Source: Ref. 3)

The wide range of uncertainty emphasizes the character of the data presently available.

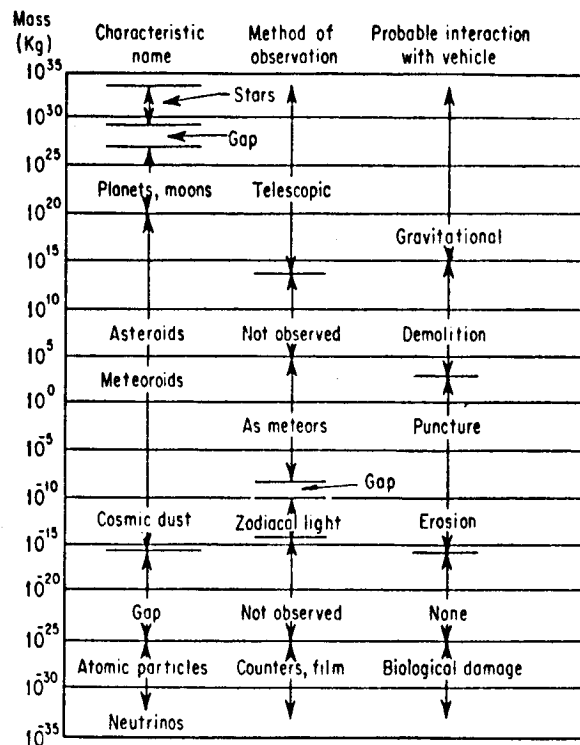
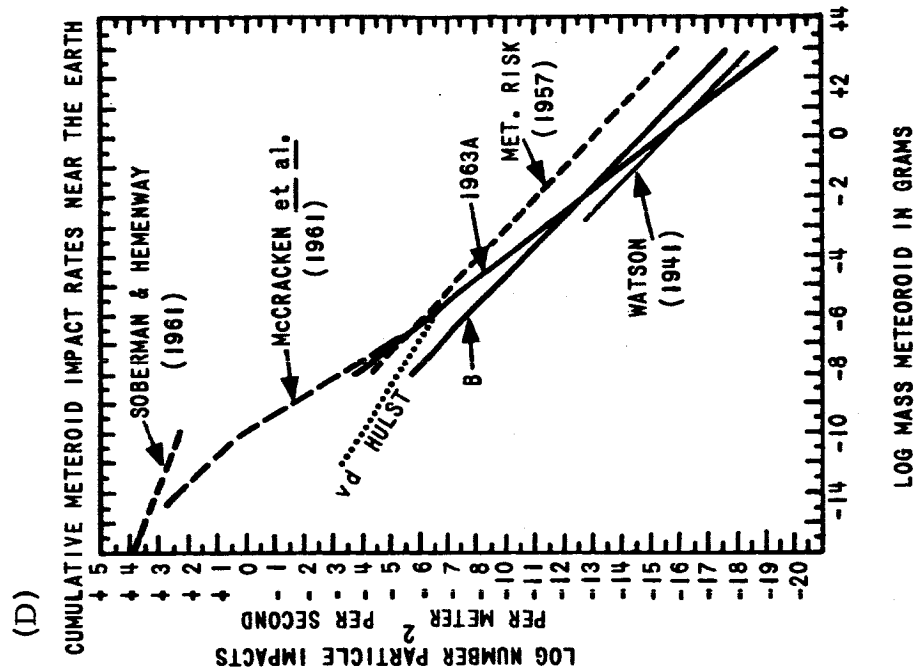
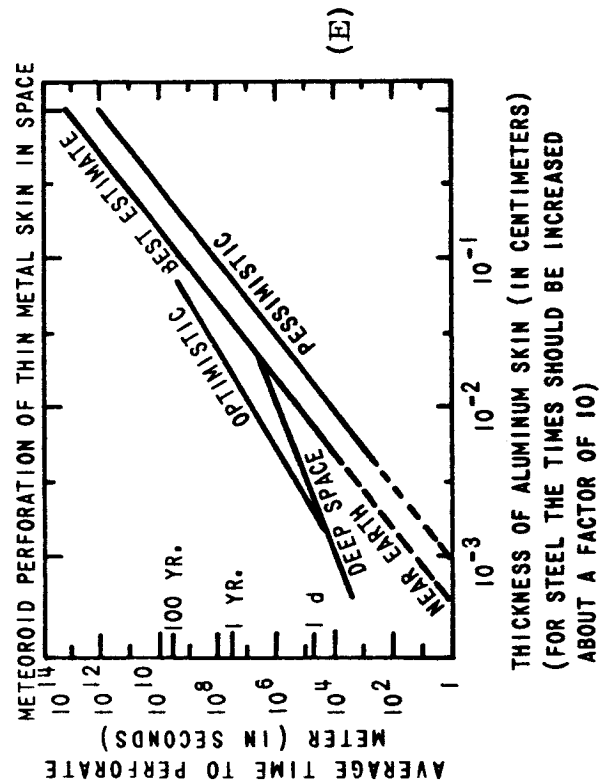


FIGURE 13-4(C). THE MASS SPECTRUM OF THE UNIVERSE
(Source: Ref. 3)

Methods of observation and the probable effects on the spaceship are indicated in the parallel columns.

(F) COMPARISON OF PENETRATION FORMULAS ON AL
AND Fe TARGETS

AUTHOR	ALUMINUM		IRON	
	p/m	$1/3$	p/m	$1/3$
(METEOROID OF $p = 0.44 \text{ gm cm}^{-3}$ AT $v = 22 \text{ km sec}^{-1}$)				
		$\text{cm gm}^{-1/3}$		$\text{cm gm}^{-1/3}$
HERRMANN AND JONES (11)	1.29	1.94	0.56	0.83
BJORK (12)	3.06	4.58	—	—
WHIPPLE (1)	—	6.04	—	—
BRUCE (15)	3.56	5.34	2.05	3.07
OPIK (16)	2.13	3.19	1.30	1.95



(Source: Ref. 2)

FIGURE 13-4. WHIPPLE'S REVISED FIGURES

TABLE 13-2. METEORIC FREQUENCY DATA

Visual magnitude	Mass, kg	Radius, μ	K.E., joules	Penetration in steel, cm	Frequency, 1/m ² -hr†
0	2.5×10^{-3}	49,000	1.0×10^7	9.9	
1	1.0×10^{-3}	36,000	4.0×10^6	7.3	
2	4.0×10^{-4}	27,000	1.6×10^6	5.3	
3	1.6×10^{-4}	20,000	6.3×10^5	3.9	
4	6.3×10^{-5}	14,000	2.5×10^5	2.9	
5	2.5×10^{-5}	11,000	1.0×10^5	2.1	3.3×10^{-8}
6	1.0×10^{-5}	7,800	4.0×10^4	1.6	9.5×10^{-8}
7	4.0×10^{-6}	5,700	1.6×10^4	1.2	2.4×10^{-7}
8	1.6×10^{-6}	4,200	5.9×10^3	0.83	6.0×10^{-7}
9	6.3×10^{-7}	3,100	2.2×10^3	0.60	1.5×10^{-6}
10	2.5×10^{-8}	2,300	8.0×10^2	0.43	3.8×10^{-6}
11	1.0×10^{-8}	1,700	2.9×10^2	0.31	9.5×10^{-6}
12	4.0×10^{-9}	1,200	1.1×10^2	0.24	2.4×10^{-5}
13	1.6×10^{-9}	910	3.9×10^1	0.16	6.0×10^{-5}
14	6.3×10^{-9}	670	1.4×10^1	0.11	1.5×10^{-4}
15	2.5×10^{-9}	490	5.1×10^0	0.079	3.8×10^{-4}
16	1.0×10^{-9}	360	1.8×10^0	0.056	9.5×10^{-4}
17	4.0×10^{-9}	270	6.6×10^{-1}	0.040	2.4×10^{-3}
18	1.6×10^{-9}	200	2.3×10^{-1}	0.028	6.0×10^{-3}
19	6.3×10^{-10}	140	8.2×10^{-2}	0.020	1.5×10^{-2}
20	2.5×10^{-10}	110	2.9×10^{-2}	0.014	3.8×10^{-2}
21	1.0×10^{-10}	78	1.1×10^{-2}	0.011	9.5×10^{-2}
22	4.0×10^{-11}	57	4.6×10^{-3}	0.0076	2.4×10^{-1}
23	1.6×10^{-11}	40	1.8×10^{-3}	0.0056	6.0×10^{-1}
24	6.3×10^{-12}	25	7.2×10^{-4}	0.0041	1.5×10^0
25	2.5×10^{-12}	16	2.9×10^{-4}	0.0030	3.8×10^0
26	1.0×10^{-12}	10	1.1×10^{-4}	0.0022	9.5×10^0
27	4.0×10^{-13}	6	4.6×10^{-5}	0.0016	2.4×10^1
28	1.6×10^{-13}	4	1.8×10^{-5}	0.0012	6.0×10^1
29	6.3×10^{-14}	3	7.2×10^{-7}	0.00089	1.5×10^2
30	2.5×10^{-14}	2	2.9×10^{-7}	0.00065	3.8×10^2
31	1.0×10^{-14}	1	1.1×10^{-7}	0.00022	9.5×10^2

(Source: Ref. 3)

† Figures in the Frequency column are cumulative. A factor of 1/2 multiplies the figures in the Frequency column to take Earth shielding into account.

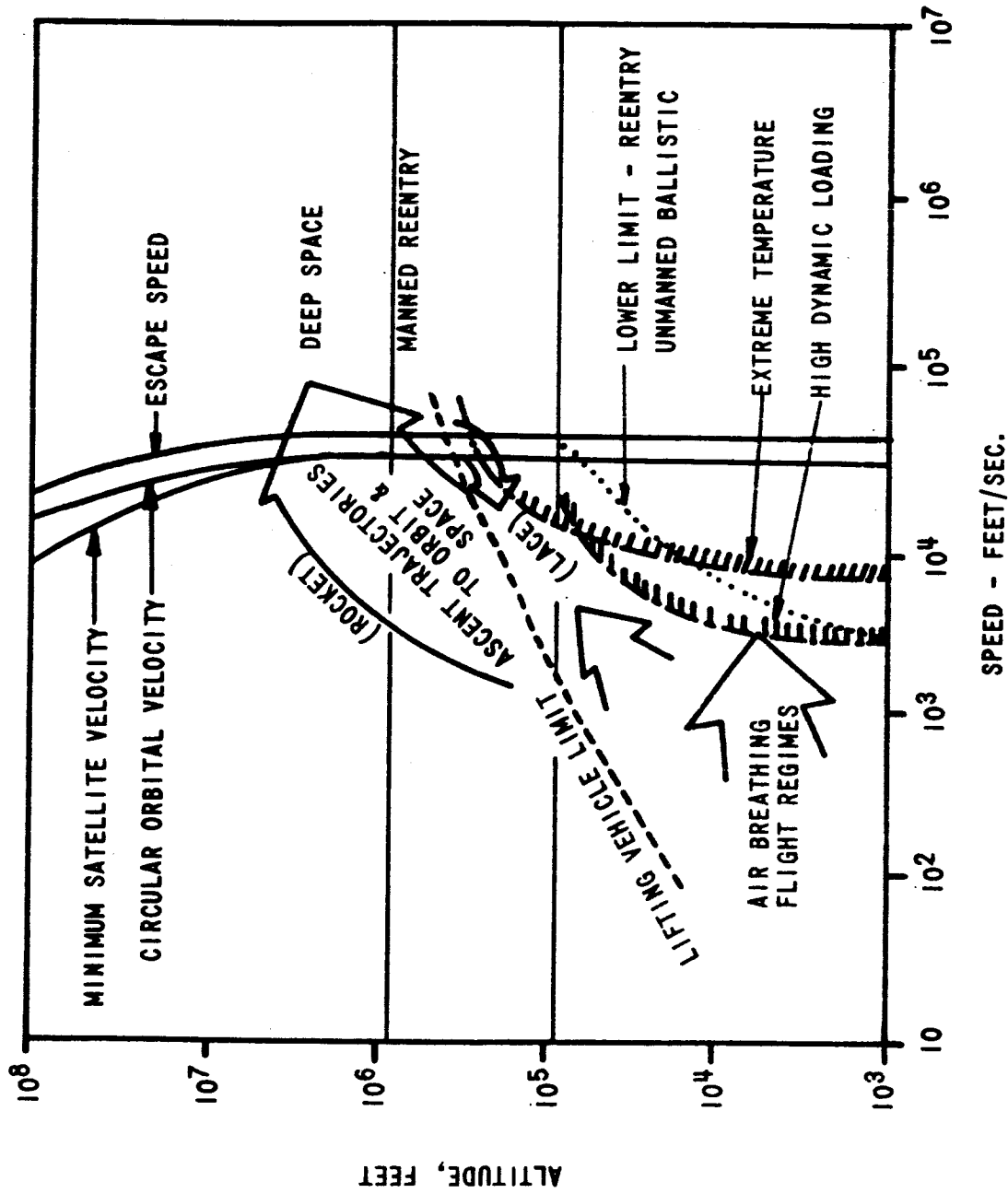


FIGURE 13-5. AEROSPACE VEHICLE OPERATIONAL REGIMES
(Unidentified Source)

13.1 ATMOSPHERIC RE-ENTRY (AND ENTRY) BODY

The following paragraphs pertinent to the problem are quoted from Reference 1.

13.1.1 Temperature Trends for Atmospheric Vehicles

The trend toward higher operating temperatures is evident from the mission profiles of present and projected military and exploratory vehicles.

Mach number for continuous flight atmospheric vehicles is expected to reach approximately Mach 6 during the next ten years. Greater speeds are foreseen for unmanned vehicles during the same period. Equilibrium structural temperatures at these speeds are shown on the chart (Figure 13-6). Higher temperatures will be experienced in certain locations on the vehicles during transient conditions.

13.1.2 Temperature Trends for Manned Re-entry Vehicles

Operating temperatures will increase more rapidly than in the past few years as the industry's products develop from boost glide and orbital re-entry to lunar re-entry systems. Predictions are less certain for these vehicles, however, since operating temperatures are so strongly dependent on configuration, re-entry corridors, and exploitation of technological advances. As rapidly as higher temperature materials become available, vehicles will be designed to use them.

Unmanned systems exist today which sustain temperatures higher than those shown in Figure 13-7. Operating temperatures for ballistic re-entries are keyed to choice of ablation materials, and vary over a considerable range as to high as approximately 5,000°F.

13.1.3 Anticipated Maximum Accelerations (Figure 13-8)

Manned ballistic and orbital return vehicles, starting with Project Mercury, will be subject to short time accelerations ranging from 10 to 20 g's. Later in the decade, satellite launched research (unmanned) space probe vehicles may be designed to withstand decelerations greater than 250 g's. Until that time, the highest load factors encountered above the ocean's surface will appear in high performance intercept missiles. Warheads of missiles designed to seek and destroy underwater targets will be subjected to even higher accelerations during water entry. (End quote from Ref. 1)

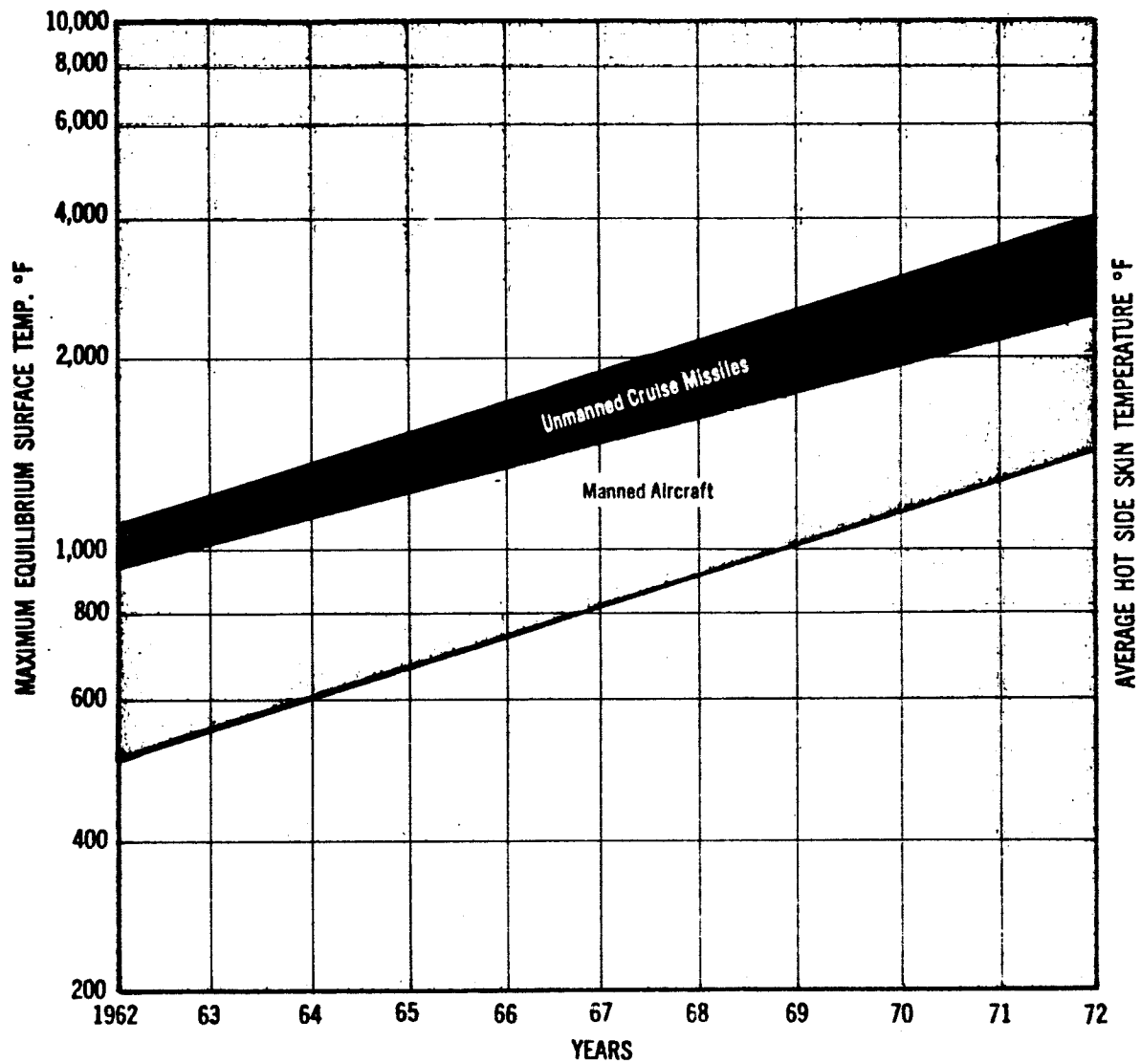


FIGURE 13-6. TEMPERATURE TRENDS
FOR ATMOSPHERIC VEHICLES
(Source: Ref. 1)

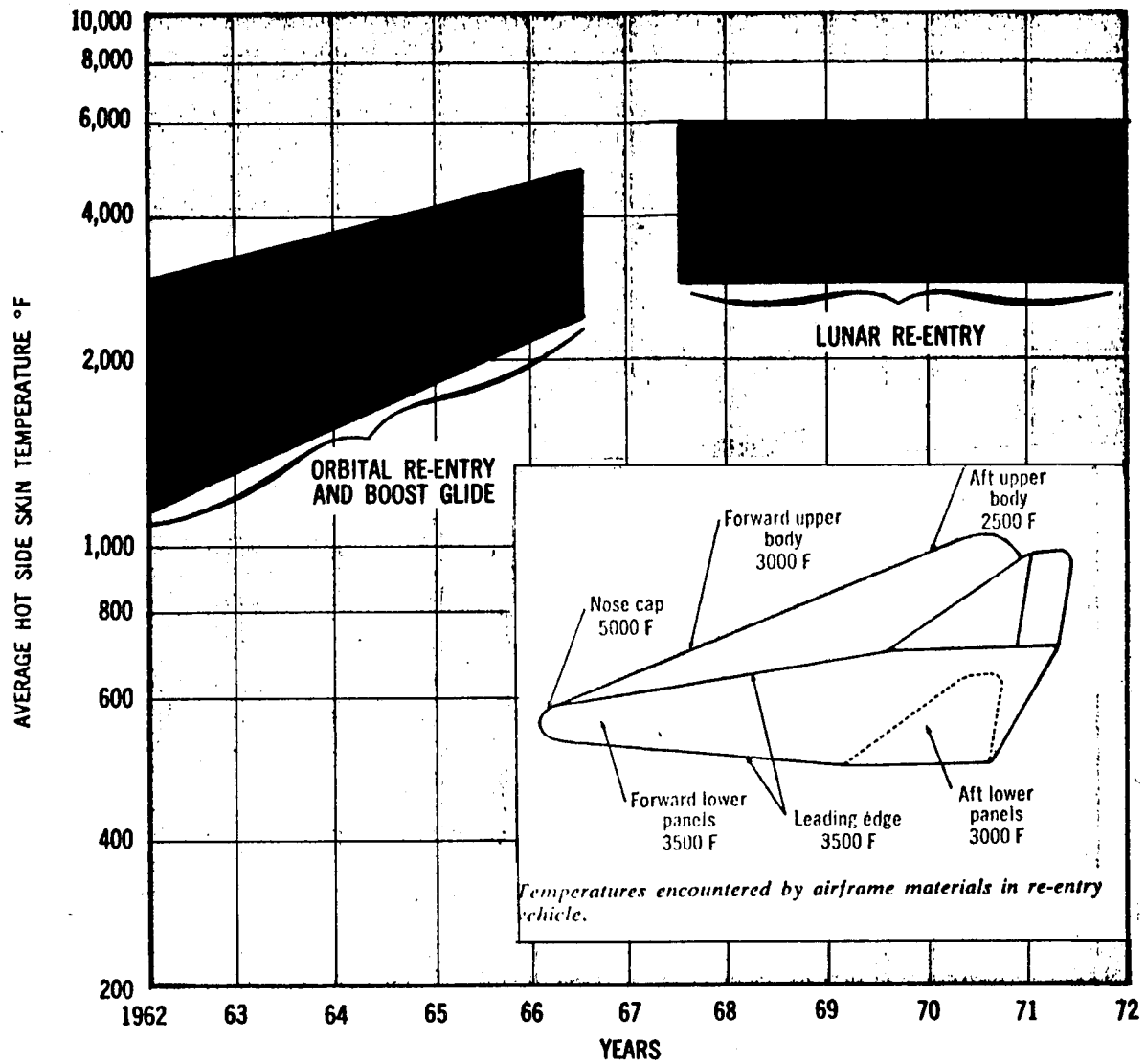


FIGURE 13-7. RE-ENTRY TEMPERATURES
(Source: Ref. 1 and 4)

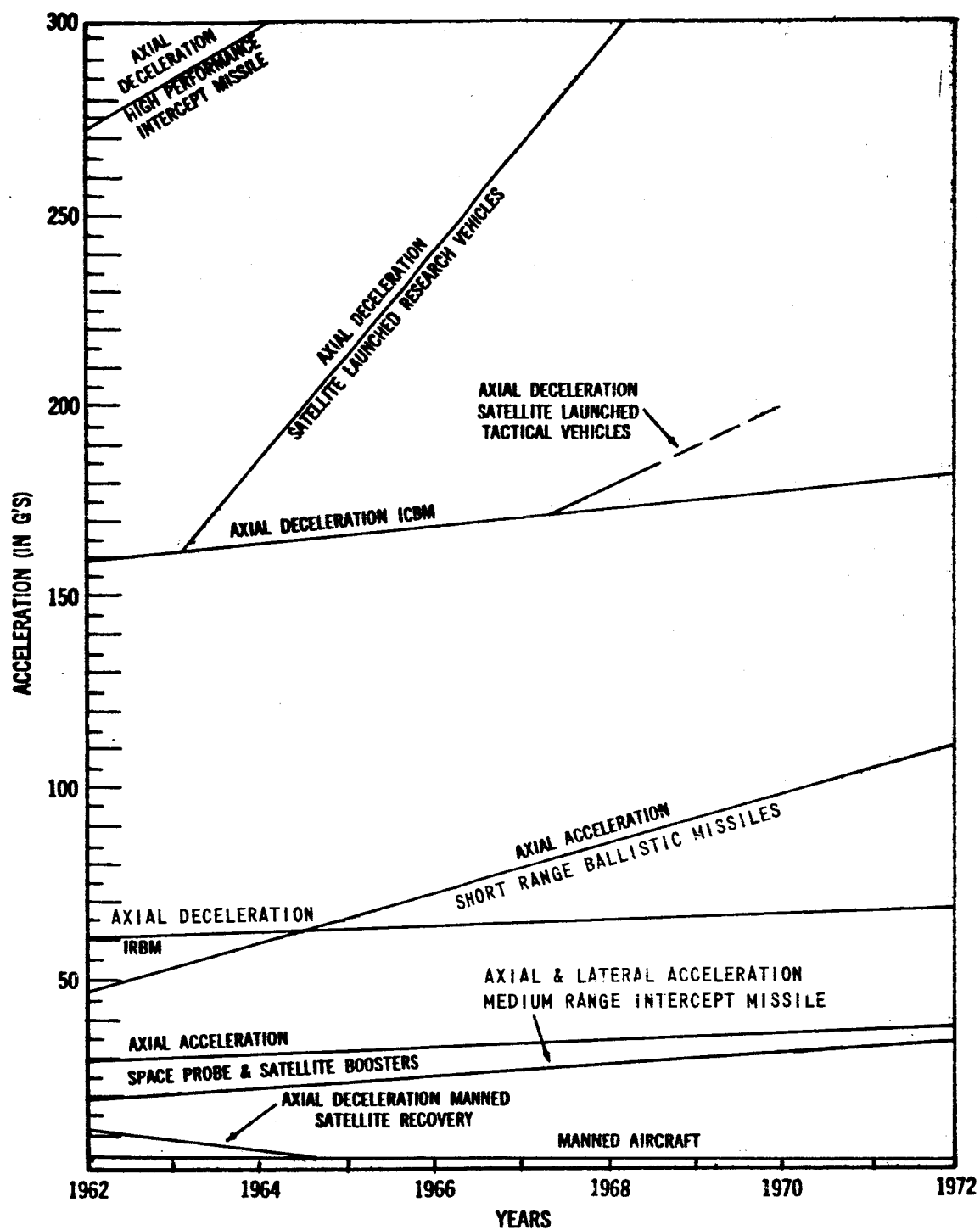


FIGURE 13-8. ANTICIPATED ACCELERATIONS
(Source: Ref. 1)

13.1.4 Temperature Trends for Atmospheric Vehicles

Figure 13-9 presents approximate bands of vehicle temperature operation. The manned aircraft band represents the technology currently predicted for future US aircraft. In 1962 predominantly all aircraft structure was composed of aluminum or magnesium alloys. The temperature band in 1962 ranges from 500°F to 900°F for Mach 2.0 aircraft. At short time temperature cycles to 500°F, all A1 2014-T6 loses 50 percent of ultimate tensile strength. Below 50 percent of ultimate design, aluminum and magnesium structure is subjected to permanent deformation and aero-elastic flutter. An example of this is found in the F4B which is Mach limited because of aerodynamic heating. In future structures such as the SST which has a speed range of Mach 3.0 and higher, the temperature band is between 1,250°F and 1,700°F (1967). This band represents the upper capability for 17-7 PH stainless. Beyond this point one must extrapolate the data based on the use of cermets and materials which provide a high temperature coating and lower the interface temperature to within the capabilities of stainless base structure.

The suborbital manned and unmanned temperature band has only one knowledgeable point; this is for a combination zirconium-graphite structure used on Dyna-Soar. The zirconium is fused to the graphite base structure and is designed to 300°F. This point is shown at 1968. The remainder of the data is analytical data taken from various re-entry vehicle analyses or extrapolated on the basis of past experience. (For further information on structural alloys, see Reference 4. - Ed.)

13.1.5 Typical Heating Rate Histories

Because of wide variations in the time histories of heat transfer to various vehicles and materials, it is helpful to examine Figure 13-12. The stagnation heating rates shown are typical of vehicles which will be operational up to 1972. The manned lifting vehicles operate at lower maximum heating rates, but sustain the heating loads for much longer periods. The ICBM and IRBMs re-enter at steep angles, decelerate rapidly and sustain large peak heating loads. The mass loss for the ballistic vehicles is very high. For the low lift and glider vehicles mass loss is negligible; however, clever insulation schemes are required to prevent the heat pulse from reaching the life support systems.

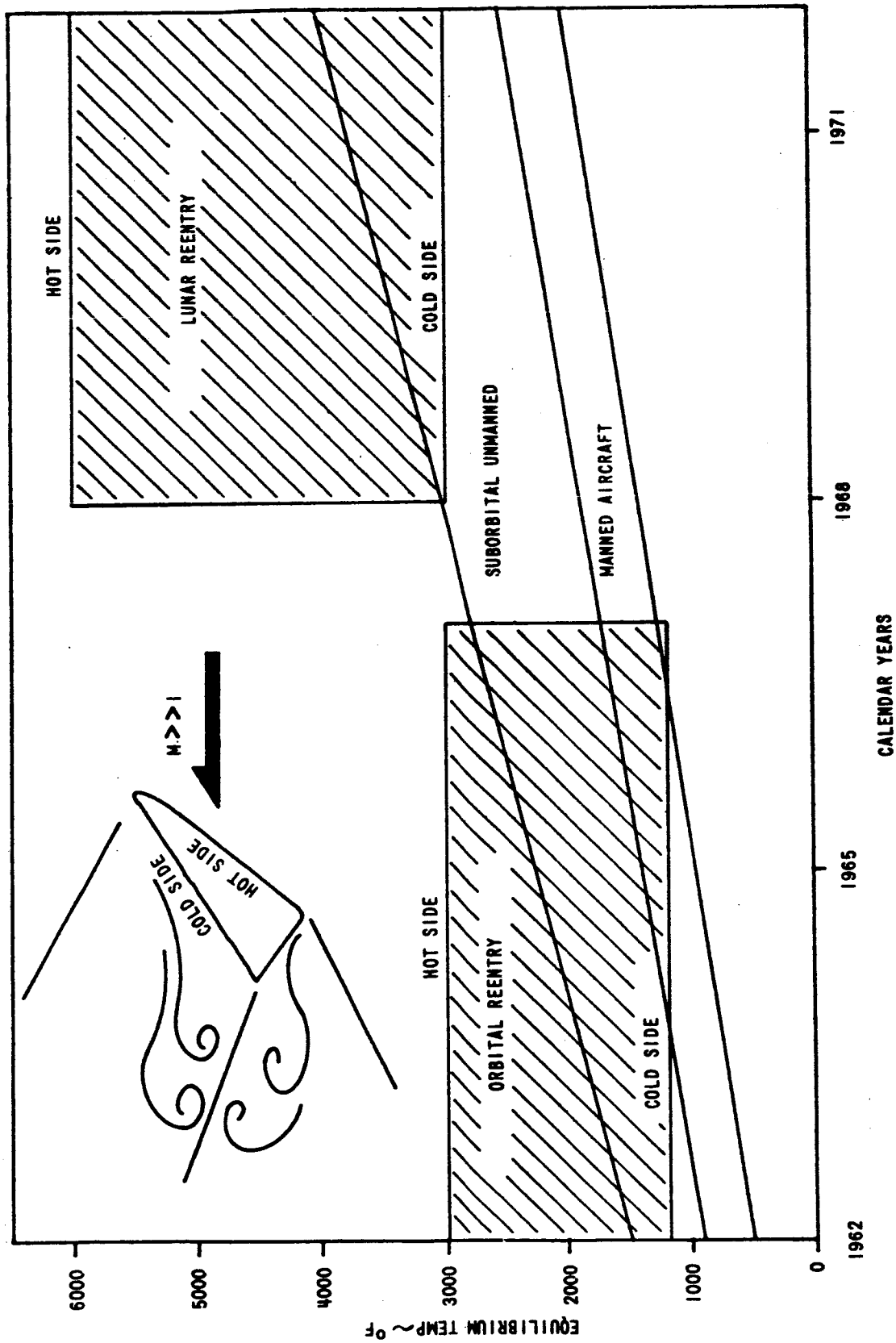


FIGURE 13-9. TEMPERATURE TRENDS FOR ATMOSPHERIC VEHICLES
(Source: Ref. 1, 4 and others)

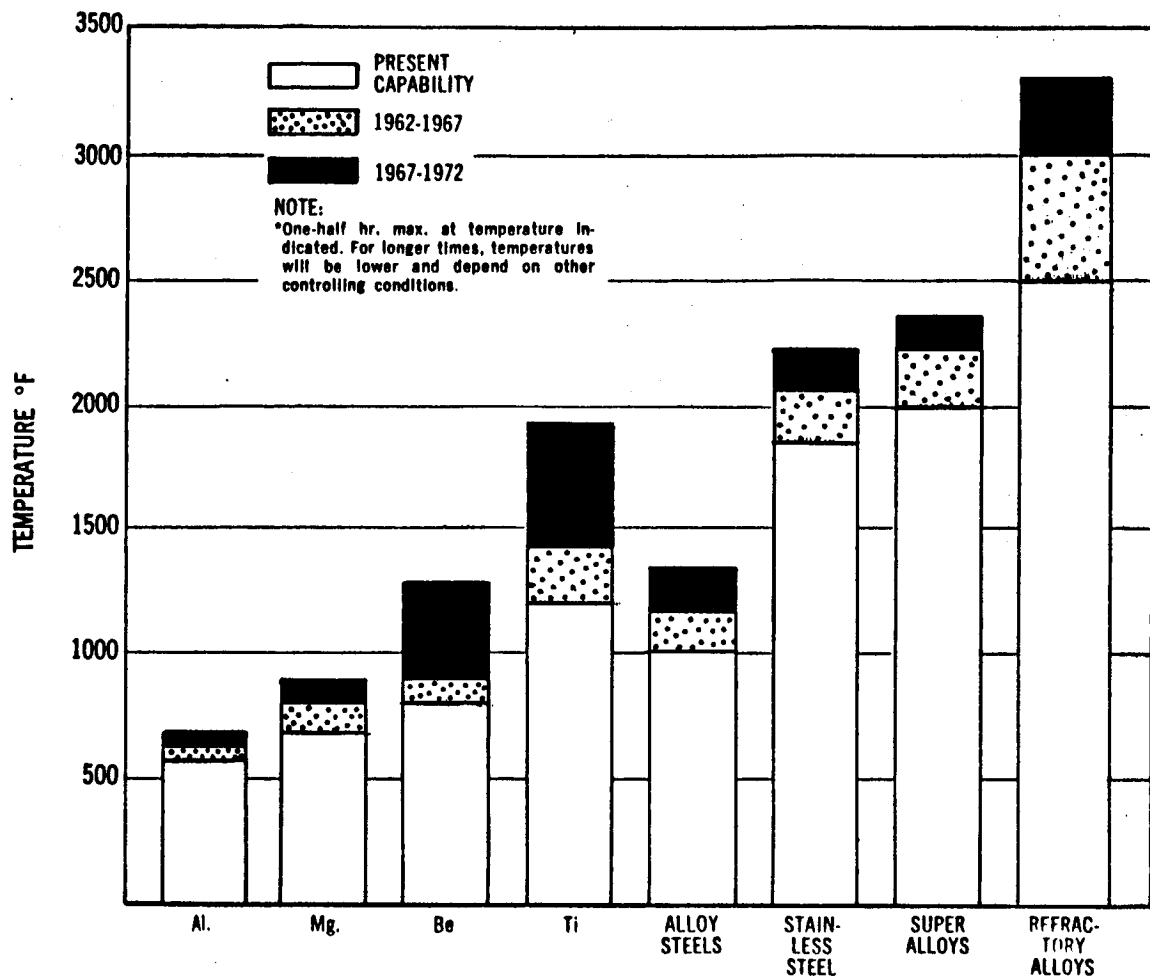


FIGURE 13-10. SHORT TIME* THERMAL CAPABILITIES OF METALS
(Source: Ref. 1)

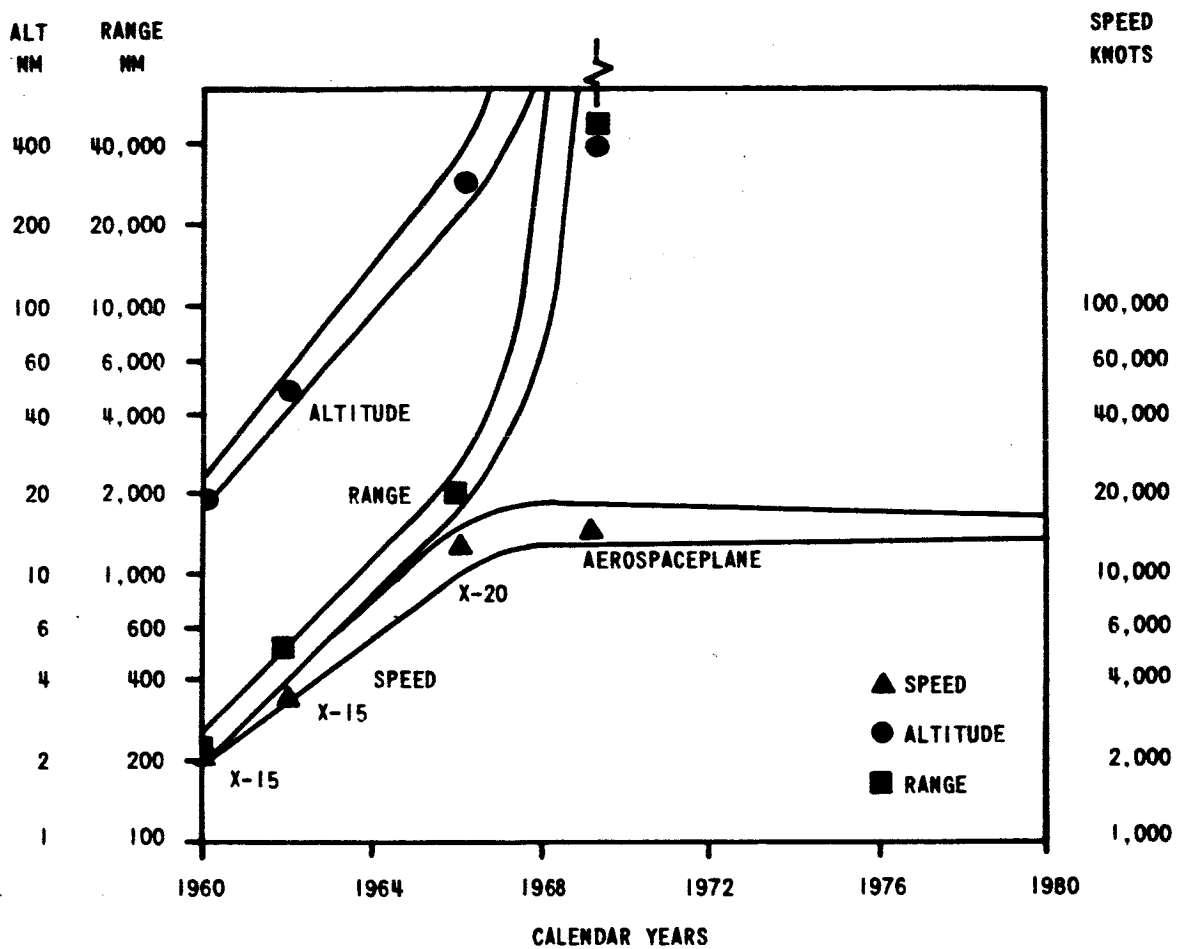


FIGURE 13-11. TRENDS OF MANNED MANEUVERABLE EXIT/REENTRY SYSTEMS
(Internal Honeywell source unidentified)

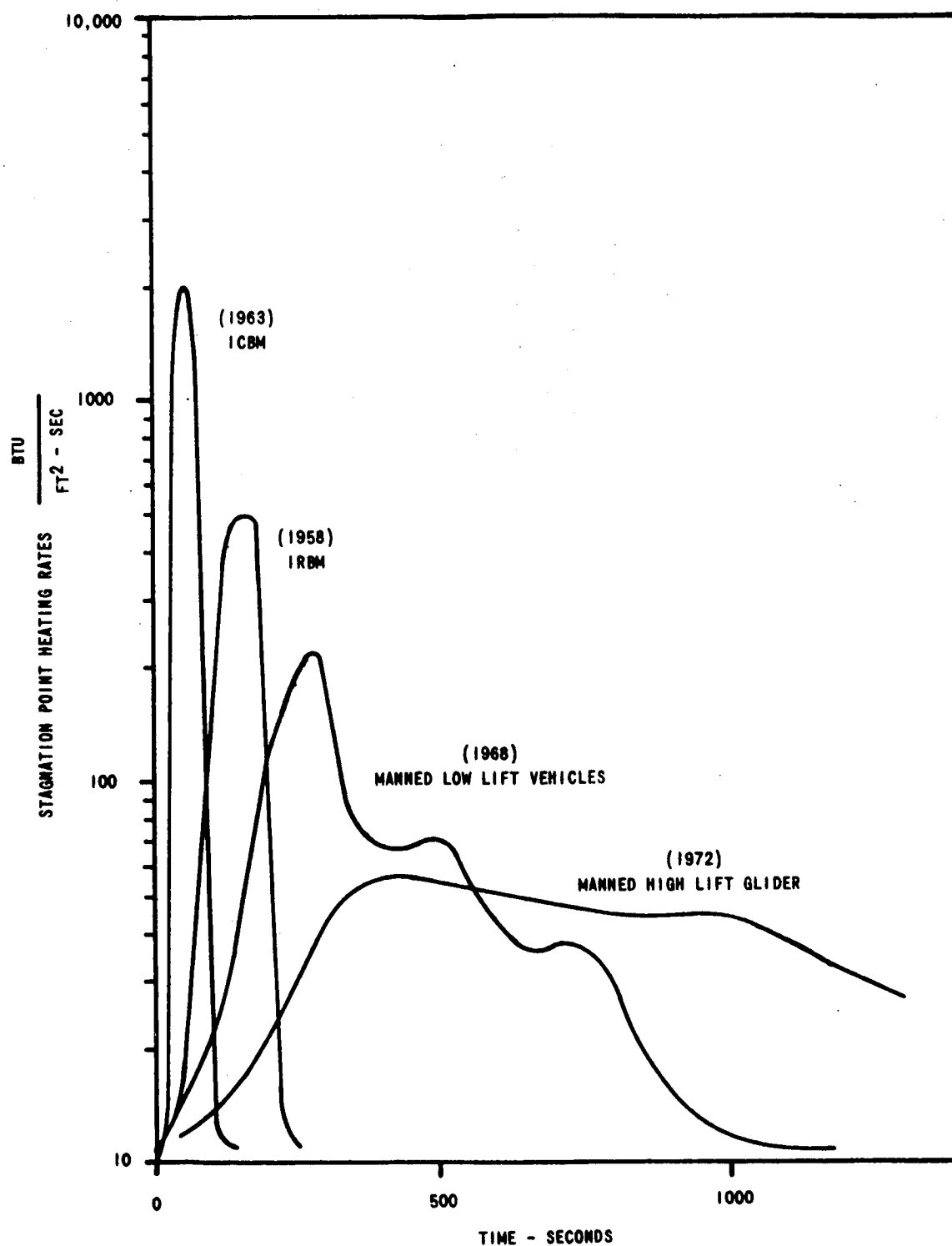


FIGURE 13-12. TYPICAL HEATING RATE HISTORIES
(Internal Honeywell Source Unidentified)

13.1.6 Thermal Protection Systems for Aerodynamically Heated Vehicles

Thermal protection systems for aerodynamically heated vehicles may be categorized as absorptive or radiative, depending on the method used to dissipate the aerodynamic heat. Absorptive systems are based on absorption of the heat by a suitable material carried within the vehicle, and such systems may be passive or active, depending whether a loss of mass is involved. Heat is absorbed by temperature rise, phase change and chemical reaction; in certain cases additional temperature rise occurs as the rejected liquids and gases mix with the hot boundary layer. The surface temperatures that develop with an absorptive system are controlled primarily by the reaction temperatures of heat absorbing material rather than by the intensity of heat input. For this reason surface temperatures have practically no significance when discussing re-entry unless the heat shield serves as a radiator. In Figure 13-13 the efficiency of various schemes are shown for different levels of heating. The efficiency of charring, or vaporizing ablation, and sublimation drops with increasing heating rates. At approximately 1,000 Btu/foot²-second transpiration and film cooling techniques become efficient methods of dissipating heat. Radiative systems resist the thermal environment by radiation of the convective heat input from a suitable high temperature surface. Normally such systems are passive and involve no significant mass loss or change in vehicle structure. A combination high temperature porous sintered material which transpires liquid or gas or a composite material which performs the same function has the greatest future potential.

13.1.7 Re-entry Design Capability

A broad picture of the role that the thermal shield plays in determining the feasibility of future entry vehicles is presented in Figure 13-14. Vehicle kinetic energy, heat shield capability, and vehicle heat load, all expressed in Btu/lb., are shown as a function of entry velocity. Of the tens of thousands of Btu's of kinetic energy possessed by each pound of the entering vehicle, the major portion is dissipated in heating the atmosphere with only a small fraction returning as a heat load on the vehicle surface. At the lower velocities heat transfer to the vehicle is primarily by convection, whereas at the higher velocities radiative transfer from the highly shocked air is predominant. As vehicle speeds increase the heat loads increase and the payload margin shrinks. Detailed knowledge of the heat load to be encountered by a given vehicle at the higher speeds is admittedly sketchy because of currently unresolved uncertainties in heat-transfer mechanisms.

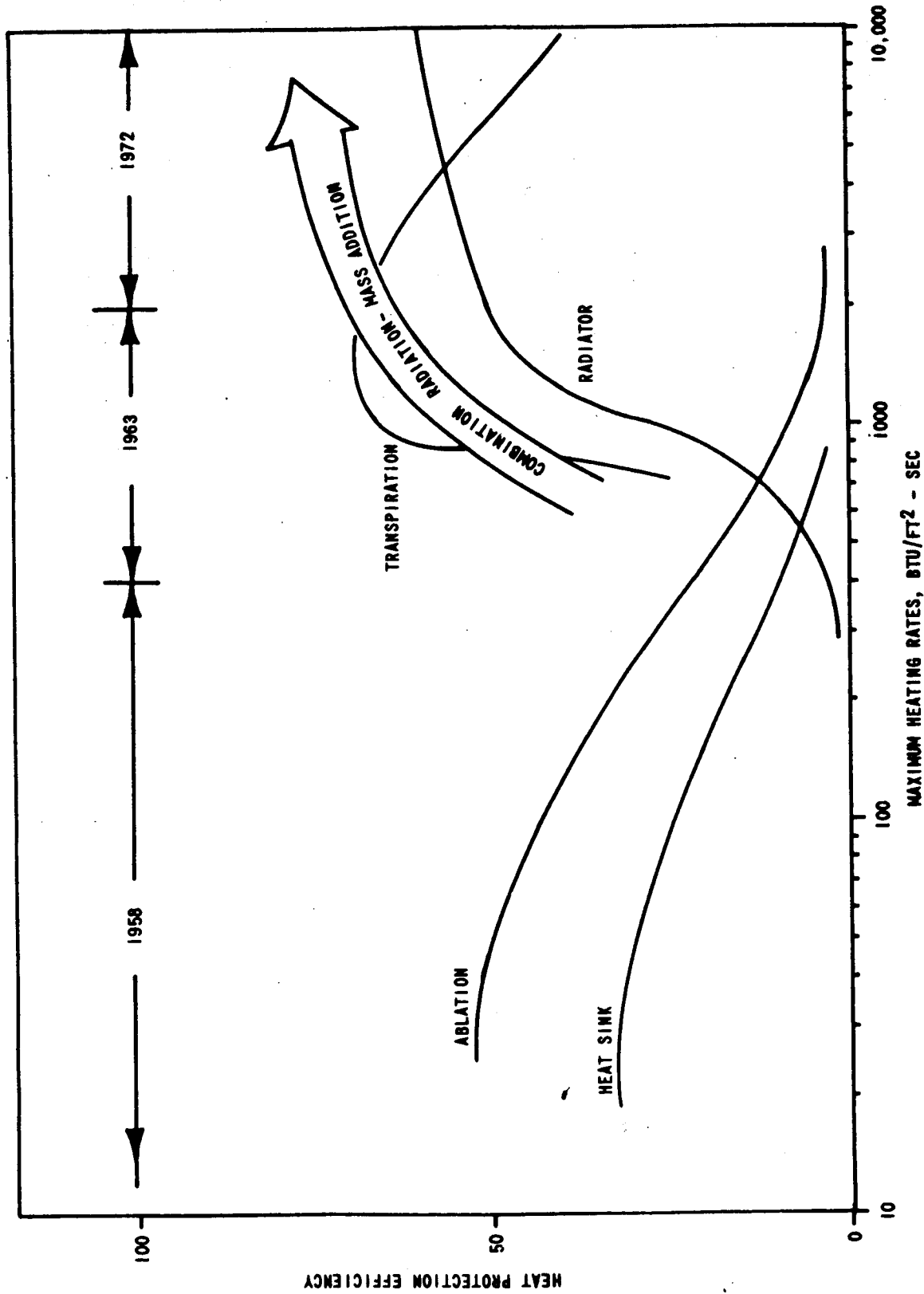


FIGURE 13-13. THERMAL PROTECTION SYSTEMS FOR
AERODYNAMICALLY HEATED VEHICLES
(Internal Honeywell Source Unidentified)

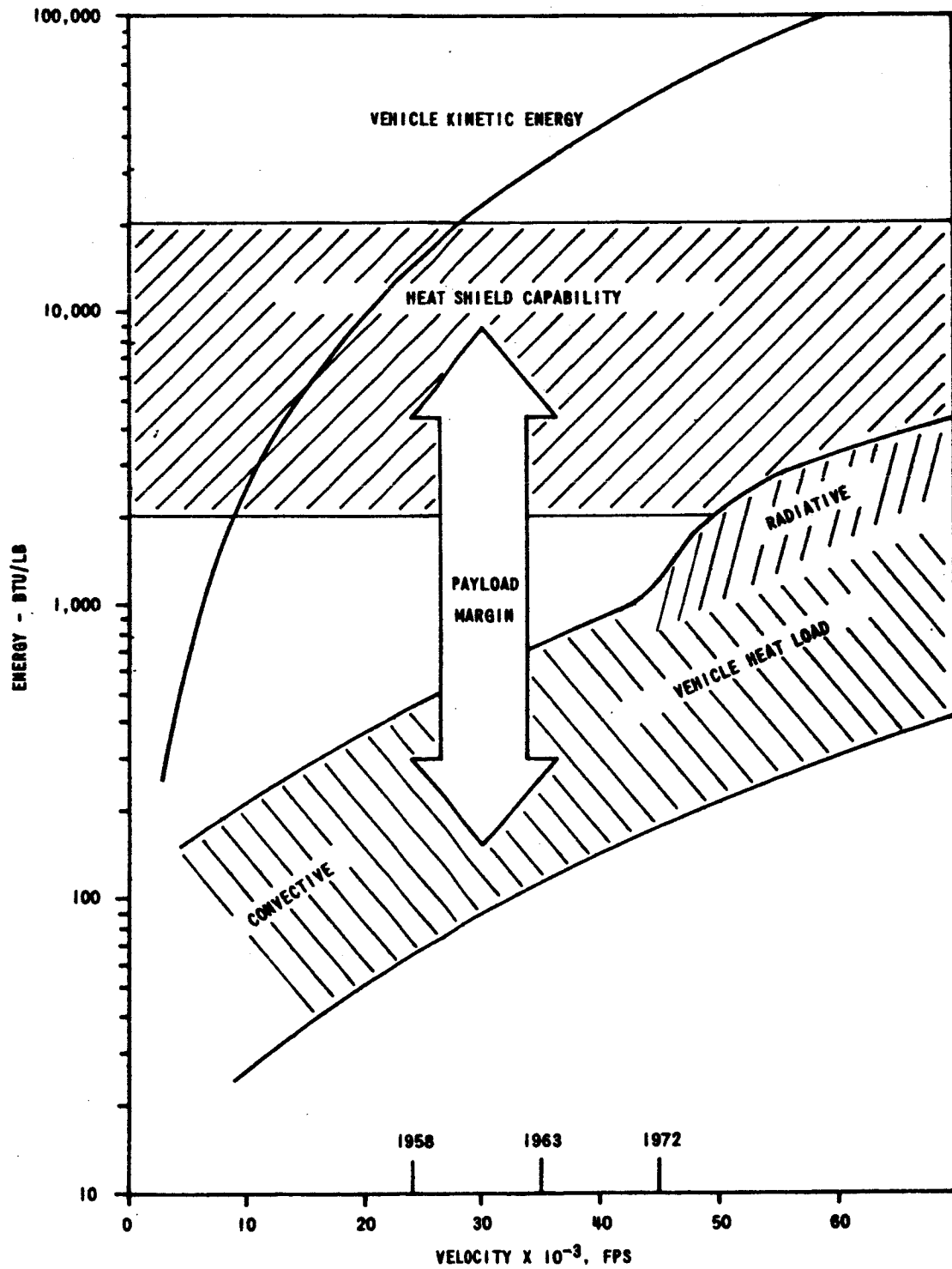


FIGURE 13-14. REENTRY DESIGN CAPABILITY
(Unidentified Source)

13.1.8 Relative Performance of Thermal Shields

The relative performance of several heat shield materials is shown in Figure 13-15. The effective energy of ablation is presented as a function of the imposed heating rate. Below about 40 Btu/foot² second the current limit of serviceability for non-ablating surfaces, the insulated shield provides the best performance. In vehicle areas where low heating rates permit this type of shield, a performance of 20,000 Btu/lb can be achieved.

In the intermediate range, a class of materials known as charring ablators have proven highly efficient. The improvement in charring ablators over low temperature ablators stems from the fact that their surfaces operate at much higher temperatures radiating a greater percentage of the total heat load.

At extreme heating loads, a degradation in performance is estimated for both the charring and low temperature ablators. Mass absorptive schemes will push the technology ahead significantly. A promising candidate for the high heating rate is graphite. The high latent heat of evaporation can be used most effectively when the heat pulse is very large.

13.1.9 Limiting Heating Trends in Materials

In Figure 13-16 the total heat input is plotted as a function of heating rates. The three divisions represent approximately the limits of metallic, ablating and advanced mass absorption schemes. The molybdenum heat shield will radiate about 40 Btu/foot²-second as shown.

The limiting heating value for the metallic shield is around 20,000 Btu/foot². Ablation materials which are currently operational radiate a large fraction of the incident heating and absorb upwards of 30,000 Btu/foot² and in certain cases probably a good deal more. Advanced absorption methods utilizing film and transpiration cooling potentially have very great capabilities. The heat absorption medium can be stored in a cryogenic liquid state and diffused to the surface as a gas. The surface may operate at a very high radiation temperature which is self-regulated by the gas diffusion across the boundary layer interface.

13.1.10 Proportion of Radiative and Ionized Heating to Convective with Increasing Ballistic and Lift-to-Drag Ratios

It is instructive to consider the percentage of heating shield which is caused by radiation and ionization in the gas during circular or supervircular entry. Figure 13-17 presents the ratio of radiative and ionized heating to convective

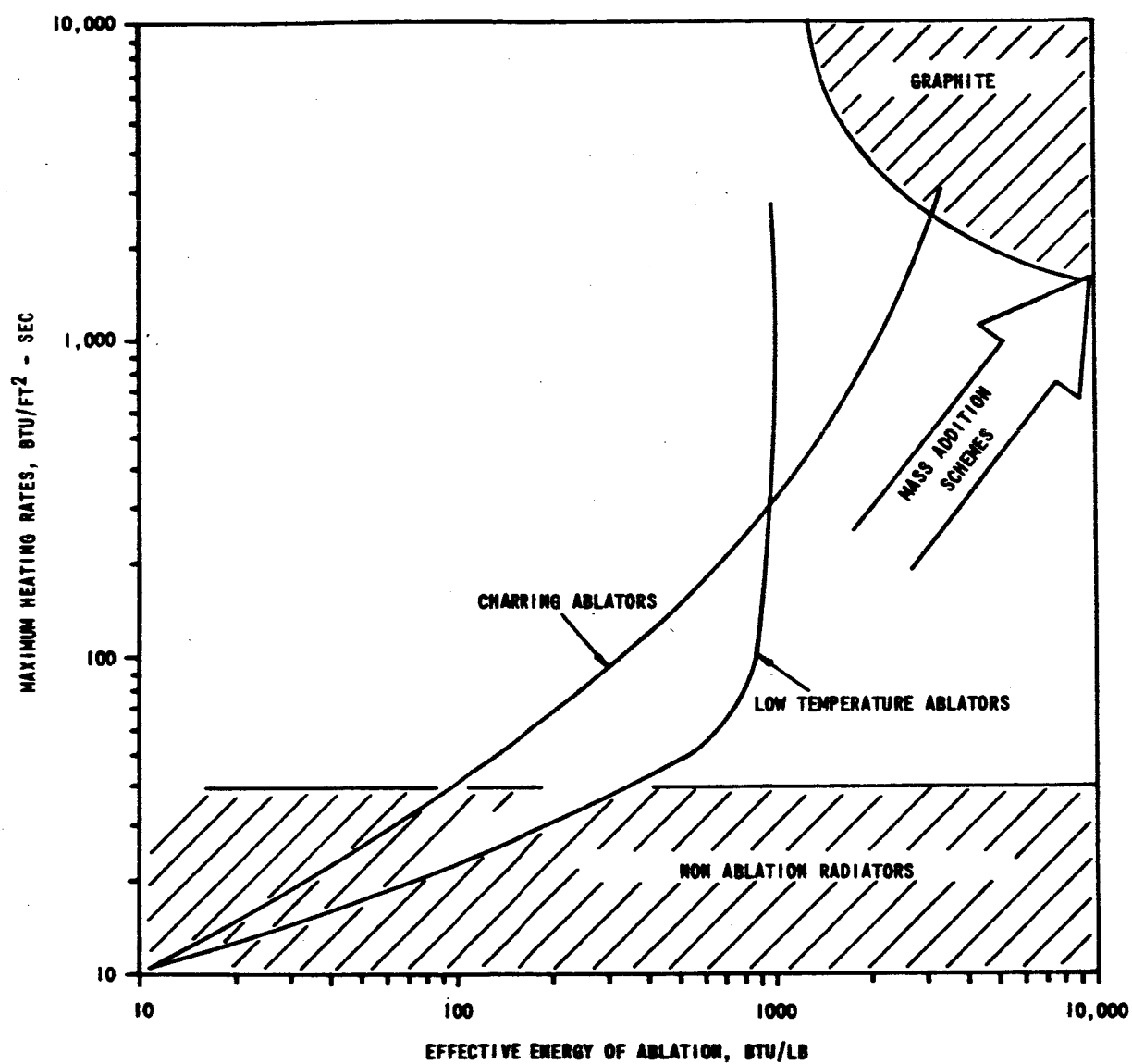


FIGURE 13-15. RELATIVE PERFORMANCE
THERMAL SHIELDS
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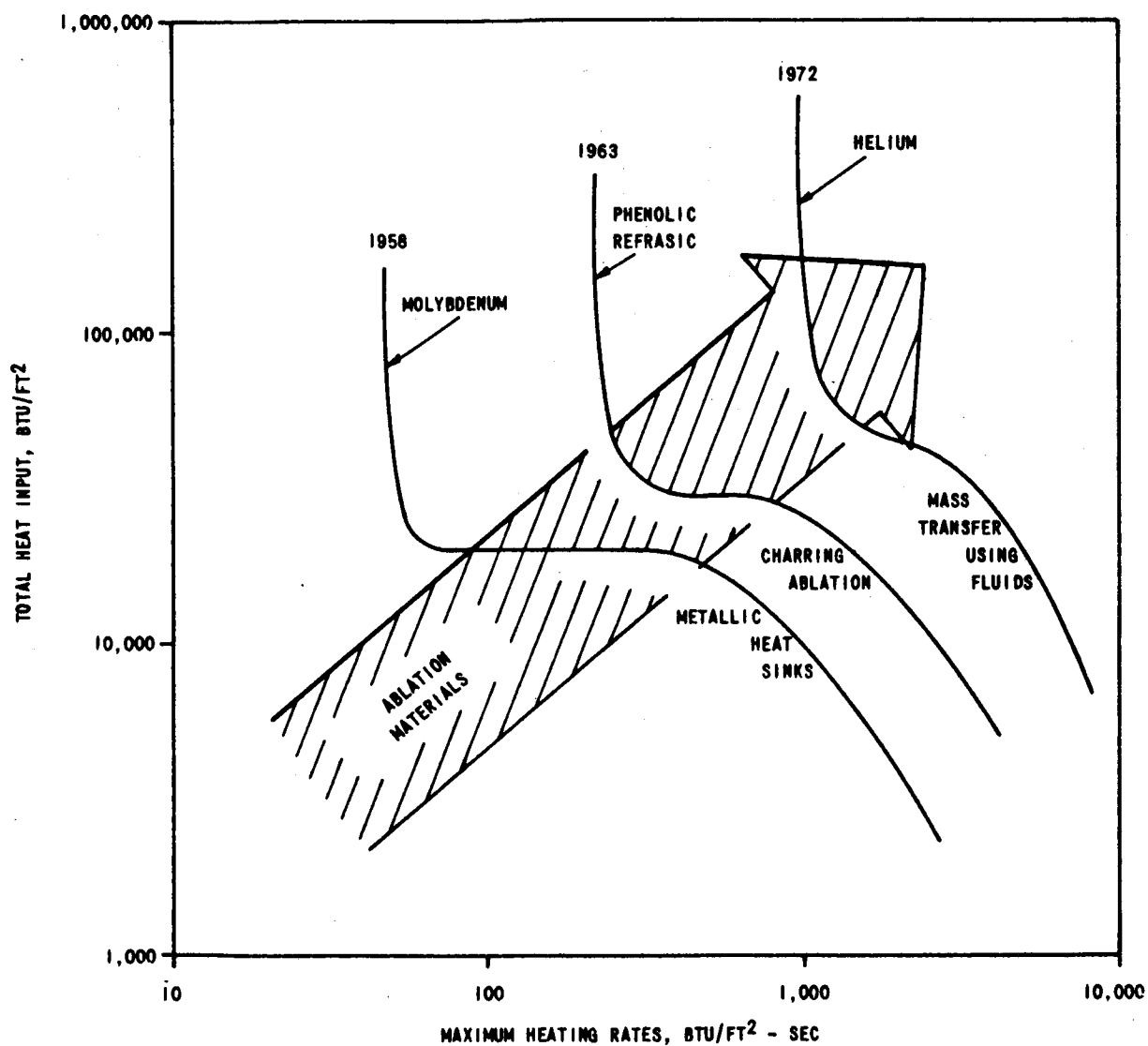


FIGURE 13-16. LIMITING HEATING TRENDS IN MATERIALS
(Unidentified Source)

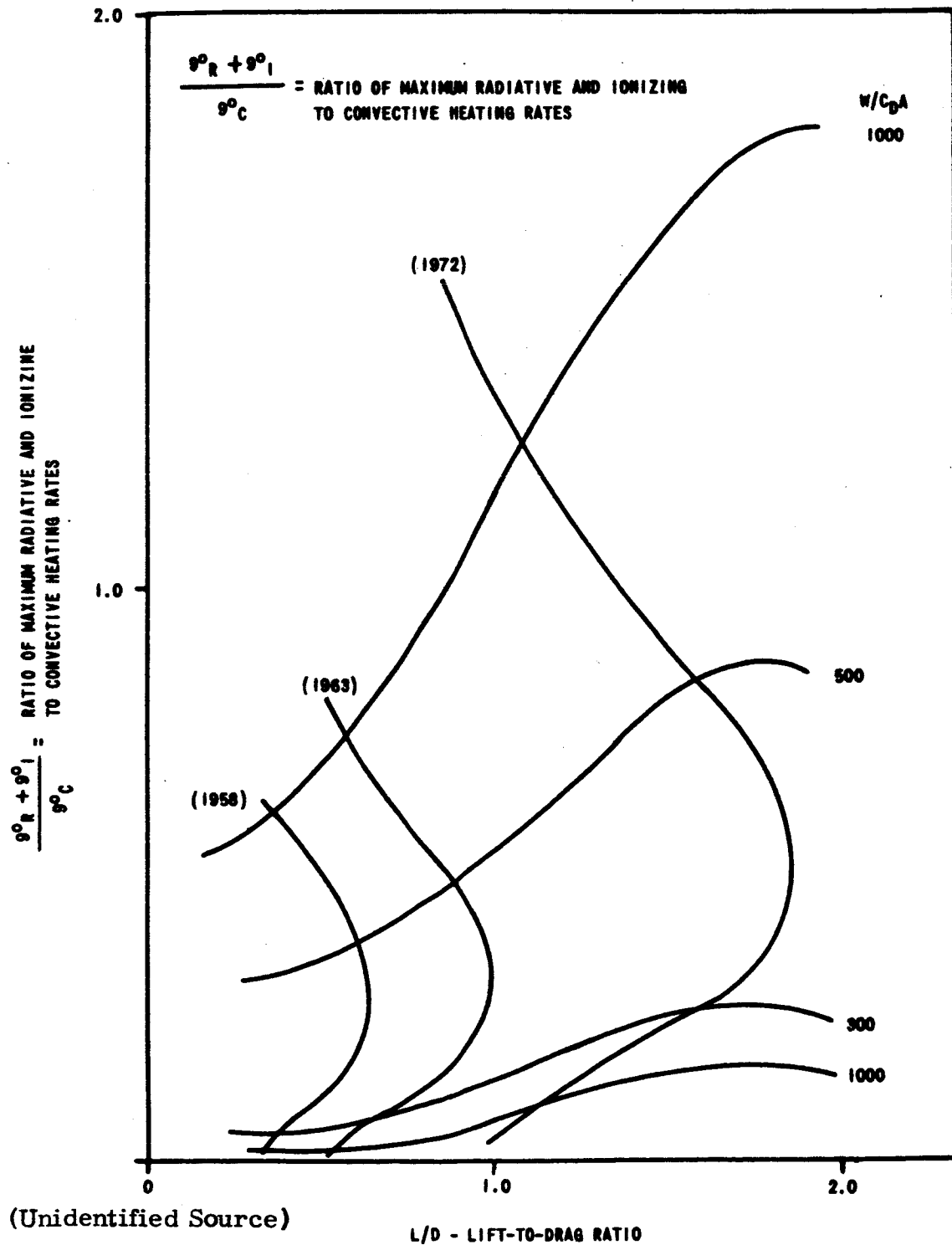


FIGURE 13-17. PROPORTION OF RADIATIVE AND IONIZED HEATING TO CONVECTIVE WITH INCREASING BALLISTIC COEFFICIENTS AND LIFT-TO-DRAG RATIOS

heating for various lift-to-drag and ballistic coefficients. As the ballistic coefficient, $W/C_D A$, or the lift-to-drag ratio L/D increases the effects of radiative heating increase due to the extremely high temperatures (20,000 to 60,000°K) in the shock layer. Large values of $W/C_D A$ imply slender bodies which decelerate deeper in the atmosphere. The effect of increasing lift and structural density on future vehicles will double and triple the total heating by compounding the importance of gas radiation and energies available due to ionization of the air.

13.1.11 Uncertain Characteristics of Martian Atmosphere

The Mariner IV refraction experiment indicates that the atmosphere of Mars is thinner than the previous pessimistic estimates. The feasibility of aerodynamic braking and parachute recovery are again open to question. Retro-rockets may be necessary to permit a Moon-type landing.

The drag versus altitude for a particular vehicle cannot be reliably predicted for Mars (or Venus) atmospheric piercing maneuvers.

13.1.12 Uncertain Characteristics of Venusian Atmosphere

Little is known regarding altitude/density/temperature profiles. Very probably ablative re-entry is feasible, but the JPL estimates of ambient temperatures (600 to 800°F at surface) present a severe structural design problem, particularly in view of the CO_2 "ice crystal" cloud hypothesis.

13.1.13 Meteoric Protection

Refer to paragraph 13.0.1 and Figure 13-4 (A through F).

13.1.14 Landing Shock

Due to the uncertain feasibility of parachutes in the Martian atmosphere, the final touchdown velocity, even with throttleable rockets, may be comparable to a 30-foot drop at the Earth. Crushable honeycomb struts or hydraulic damper struts may be required.

The possible landing shock on Venus presents a problem because of utter ignorance about its surface. Its gravity is estimated as almost that of the Earth (0.8 to 0.9) and may even be higher.

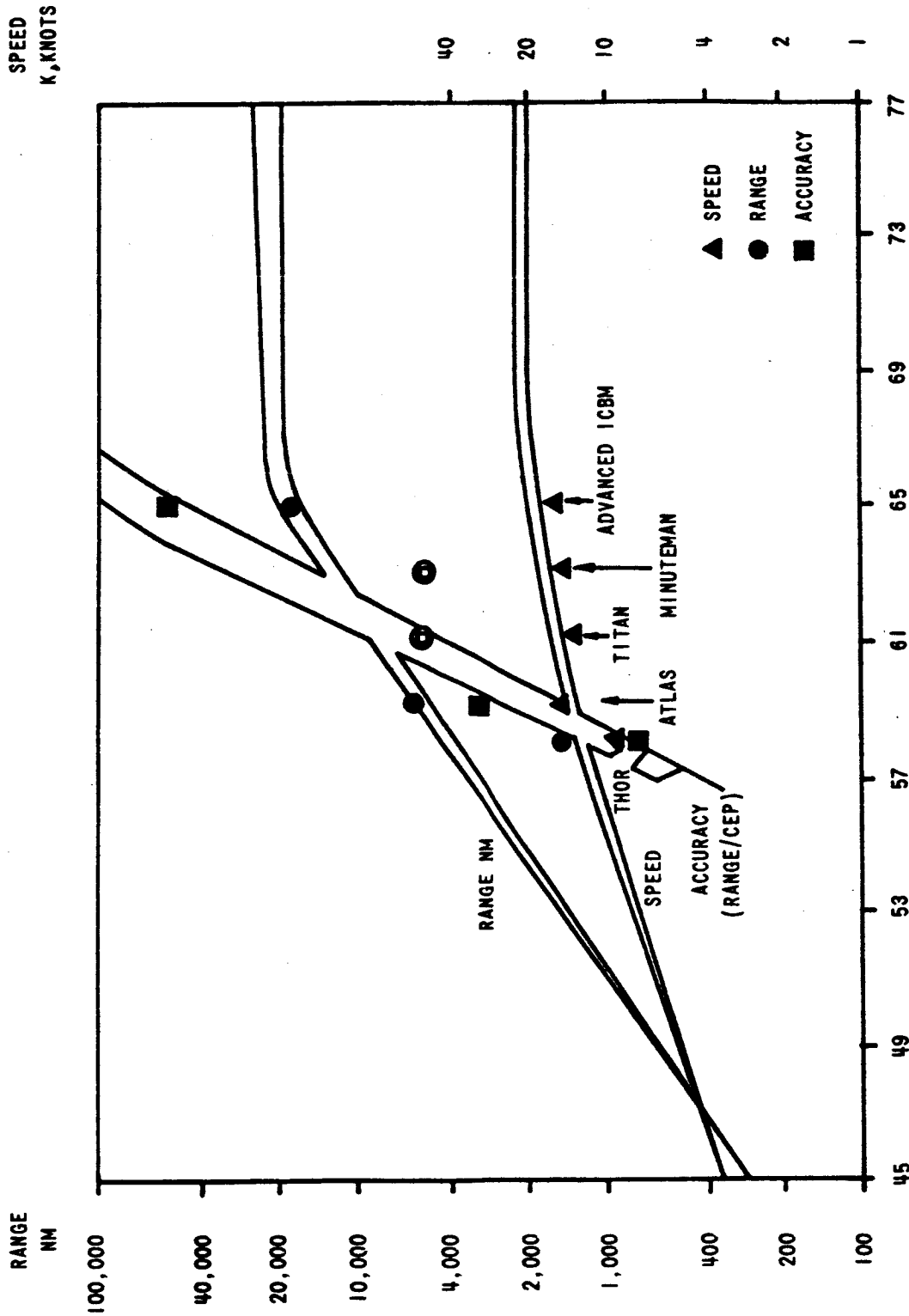


FIGURE 13-18. PERFORMANCE TRENDS
OF BALLISTIC SYSTEMS
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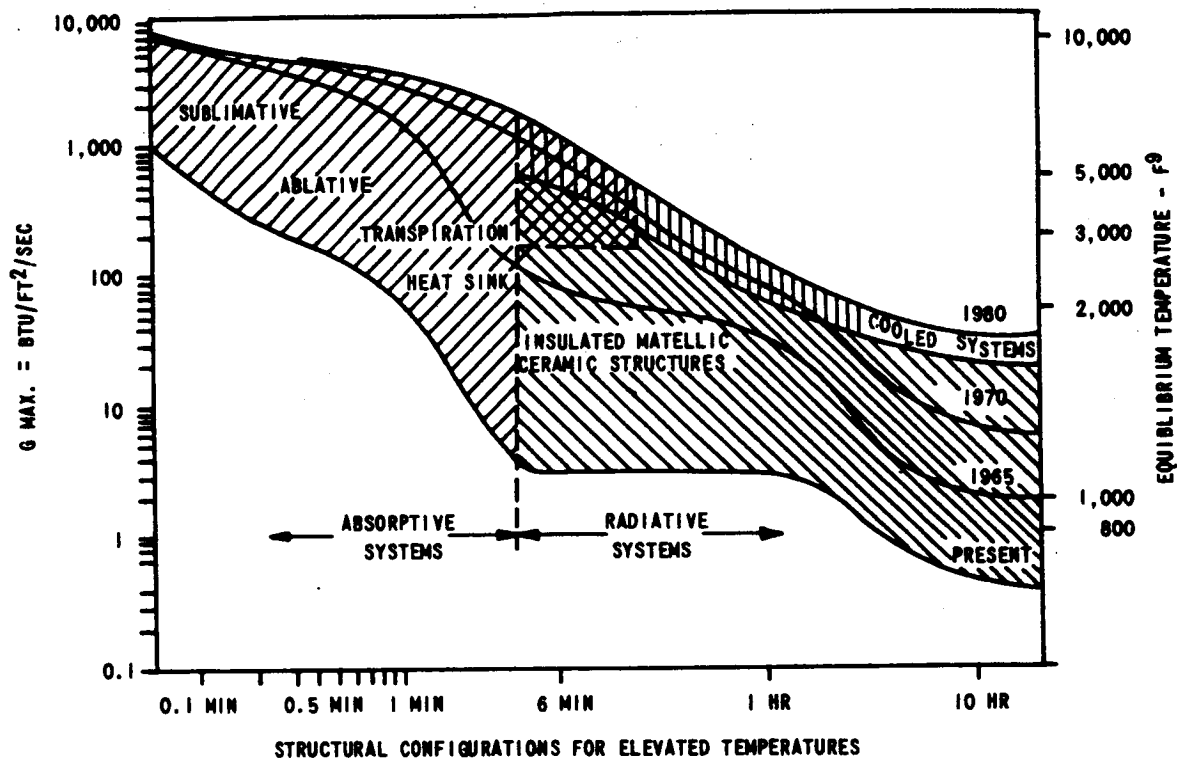


FIGURE 13-19. STRUCTURAL DESIGN TRENDS

(Unidentified Source)

Structural Configurations for Elevated Temperatures - The indicated progress trends represent significant advancements in structural technologies. Development of accurate analyses of unique structural configurations are mandatory requisites to hypervelocity flight. Our present-day structural capabilities permit sustained flight in low aerodynamic heating areas of high dynamic pressures and aeroelastic influences. Structural design approaches for the short duration, multiple usage and one-shot flight applications exist in the moderate temperature environments. By 1965, structural capabilities can be provided to achieve re-entry from high altitude orbits and to explore the atmospheric hypersonic flight regimes. The refractory metallic and nonmetallic configurations appear to offer the earliest solution for the research vehicles that will investigate the flight performance corridors of interest. Further scientific advances in the design of thermantic or cooled structural configurations will significantly extend the ability to obtain increased payloads, greater maneuverability and overall performance benefits for extreme altitude orbit vehicles.

13.1.15 Entry Heating

Refer to general discussions in paragraphs 13.1.1 through 13.1.10, with Figures 13-6 through 13-19.

13.1.16 Aerodynamic Configurations

Refer to paragraphs 13.1.6 through 13.1.10.

Several lenticular radiative entry body designs are being studied. The M-2 has completed wind tunnel tests and is ready for high altitude drop tests. The Start/Prime program will perform re-entry body research.

Apollo type ablative capsules are state of the art, but performance during re-entry awaits demonstration tests.

13.1.17 Structural Weight

Additional weight of entry heat shields penalizes payload. If safe radiative entry is feasible, payloads (and living space) can increase.

13.1.18 Sterilization

Unmanned vehicles might be sterilized by a sustained bake at about 130°F for several hours prior to launch.

Sterilization of manned vehicles probably is wasted effort; since no sterile men can go along, recontamination is immediate.

The heat of sterilization introduces electronic reliability and cryogenic materials problems. Chemical sterilization is rather messy and dangerous to personnel.

13.1.19 High Velocity Re-entry

This refers to the direct plunge into the Earth's atmosphere at return hyperbolic to parabolic velocity. Taking this dangerous risk could permit enormous reductions in launch boost requirements (or comparable increase in useful payload).

13.1.20 High Temperature Materials

Refer to Figures 13-10, 13-16 and paragraph 13.1.9. The general problem of structural strength at high temperatures is met with most of the available materials. Solutions of some of these problems for space could improve materials technology for many other applications. Needs include such things as springs and bimetal elements for operation up to 3,000°C. A very near-in need is for wire insulation which remains flexible and effective at 1,000 to 3,000°C.

13.1.21 Uncertain Characteristics of Martian and Venusian Atmospheres

For braking during grand tour flybys. See paragraphs 13.1.11 and 13.1.12.

13. 2 OPEN FRAME STRUCTURE

Examples are Ranger, Mariner and Nimbus.

13. 2. 1 Meteoric Protection

Each sensitive item requires its own enclosure skin. See paragraph 13.0 .1.

13. 2. 2 Material Degradation

Volatile components of complex materials sublime in vacuum. Plastics become brittle, glasses darken due to radiation, meteoric wear occurs, thermal cycling can cause stress hardening, etc.

13. 2. 3 Shielding for On-Off Operations

Once a nuclear mission rocket has been fired and shut down, it becomes hot due to the formation of short half-life isotopes. If this mode of engine operation is used, shielding of critical elements of the vehicle is essential.

13.2.4 Docking Shock

Visualize a pair of railroad locomotives gently "docking" at one foot per second. You may have been on such a train.

13.3 LIVING QUARTERS

The term refers to the inhabitable, pressurized capsules of a space vehicle or base installation. Factors to be considered are summarized in Figure 13-20.

Quoting Ref. 1:

"As the period of occupancy of a closed ecological environment is extended, the probability of induced fluctuation, which may jeopardize pilot safety, increases. Unlike terrestrial environment fluctuations, which return unnoticed to equilibrium as a result of natural forces, the artificial environment of a manned space vehicle must be influenced by the occupant in a manner which will produce an immediate balancing response. Over extended periods, unforeseen mutations of organisms (natural, chemical, radiation, or others) may make fungicides and pharmaceuticals ineffective, dormant pathogens may become active, and food regeneration materials may be subjected to micro-organism contamination. Recirculating air systems can be contaminated by volatile materials (lubricants, plasticizers) and by the toxic products of reactions between cosmic radiation and

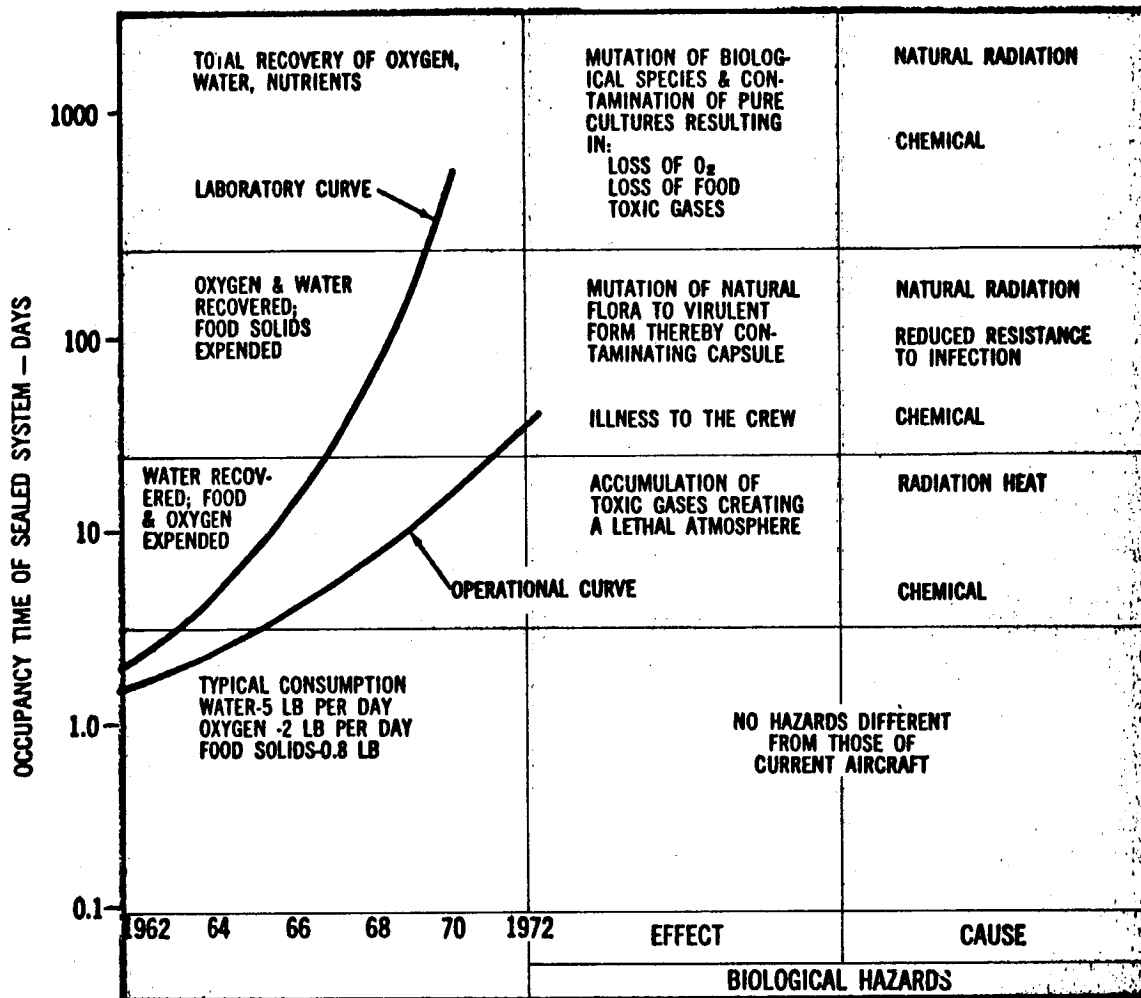


FIGURE 13-20. VARIATION OF BIOLOGICAL FACTORS WITH OCCUPANCY TIME
(Source: Ref. 1)

materials in a spacecraft. Although the astronaut will have several tools to apply in maintaining his life-sustaining balance, there always is the possibility of rapid environmental fluctuation." (End quote from Ref. 1) Figure 13-21 illustrates two estimates of required shielding weight as a function of mission time for a specific space vehicle design. The values may be roughly scaled for other concepts.

Figure 13-21 illustrates the change in structural weight (ΔW) versus mission stay-time on the lunar surface. Due to the limited data on meteoroid penetration, it is somewhat questionable as to what penetration criteria should be utilized; both Summers and Bjork's criteria are presented. This is based on Whipple's flux density data. The Bjork's criteria requires about one-half the protection as does Summers.

13.3.1 Portal Air Loss

Even laboratory type vacuum chamber ports leak enough to require continuous pump down for long duration vacuum experiments. Large hatches may leak very little when new, but with repeated use the seals become less perfect. There will be no zero leak hatches and the leakage loss must be supplied from stored gas tanks to maintain internal pressure.

13.3.2 Meteoric Protection

Refer to paragraph 13.0.1. Shielding which is adequate against meteoric particles is adequate for all radiation except neutrons, gamma and X-rays. Water, hydrogen and oxygen provide effective protection against neutrons but dense metals are needed for gamma and X-ray protection.

13.3.4 Biological Sterilization and Back Contamination

Refer to paragraph 13.1.18. The first reason for sterilization is to avoid contamination of other worlds. With returning astronauts the reverse problem may exist. Extraterrestrial, viral or fungus (or unknown) microscopic life forms capable of attacking man or his food sources conceivably could represent a hazard to the entire Earth.

Possibly, return quarantine decontamination of vehicles, astronauts and everything they touch may be required until more is known of exobiology.

13.3.5 Leakage, Inflatable Structure

The problem of leakage with inflatable structures is only slightly different than that for rigid structures, except that the frames are less rugged and tend to lose structural form and strength with deflation.

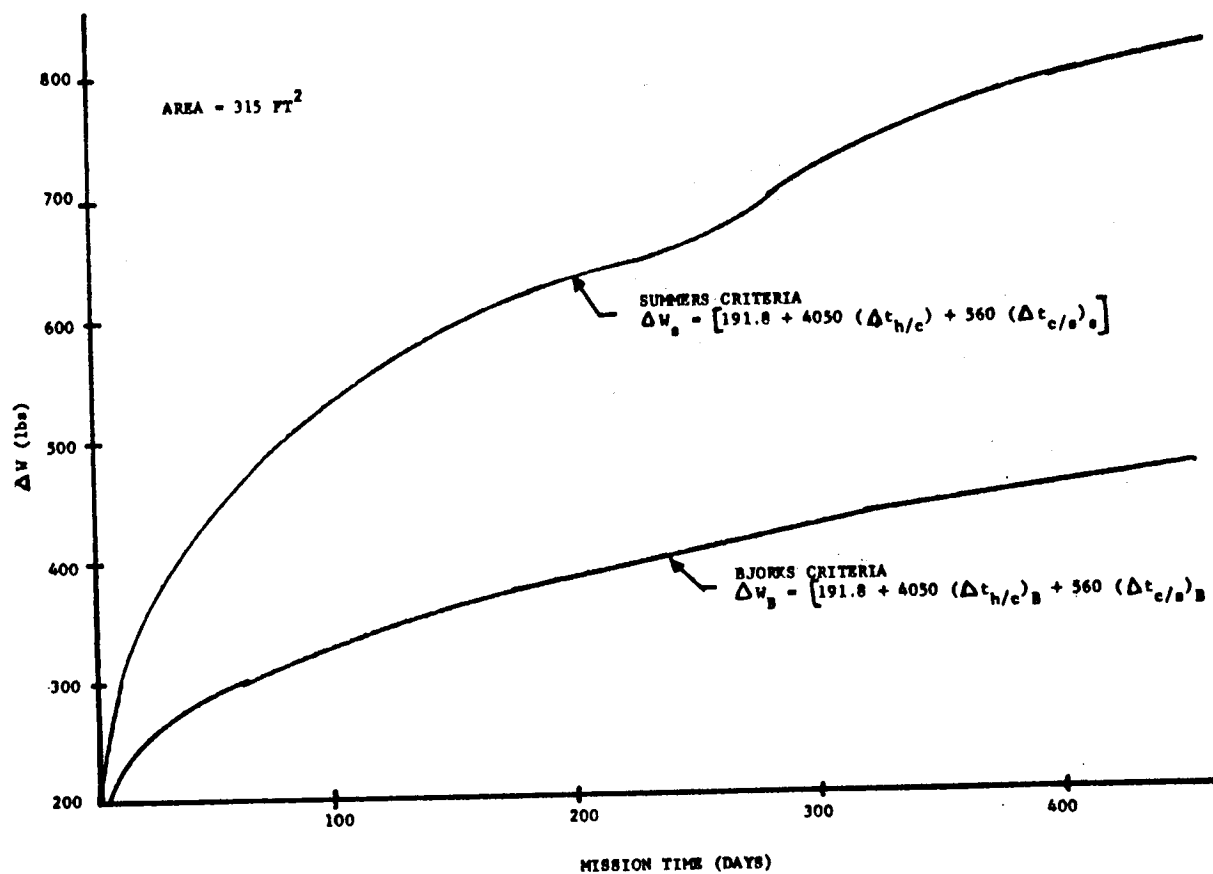


FIGURE 13-21. MICRO-METEOROID PROTECTION
 SHELTER WEIGHT INCREASE VERSUS MISSION TIME
 (Unidentified Source)

A real need is for easy leak detection from within the vehicle. IR image tubes may provide a means for finding cold spots around leaks. A simple adhesive tape patch would effect rapid repair. Chewing gum also might be practical, and would tend to harden at the exposed surfaces as it exuded through leaks.

13.3.6 Portals, Inflatable Structures

It is difficult to visualize an inflatable air lock structure. However, some telescoping and locking cylinder chambers with heavy fabric end covers conceivably could be designed. Very probably the air locks and cargo ports will have to be rigid members integrated with the inflatable structure.

13.3.7 Leakage

The general problem of pressurized vehicles. See paragraph 13.3.5.

13.3.8 Portal Reaction Thrust

Air locks in elaborate, future vehicles may be pumped-down to storage prior to exit. Early vehicles may merely vent the air overboard. In either case, the residual air escapes to space and transfers momentum as soon as the external door is opened. The air lock door becomes a rudimentary cold gas jet and can disturb the vehicle's attitude or trajectory. The exhaust air also will carry out with it any loose material inside the lock, such as a man with tools and equipment (for example, Astronaut White's glove).

13.3.9 Optical Window Degradation

Erosion, radiation coloring and thermal cycling can deteriorate the optical qualities. Pressurization will cause cold flow bulging of windows, which will only become serious after long flights and when optical flatness is critical to the window's function.

13.3.10 Waste Disposal

If waste is simply shoved out the air lock, it may follow the vehicle for a long time and obstruct the view.

If it is forcibly discharged into some other trajectory, the reaction thrust must be either accounted for or precisely balanced by simultaneous discharge in the opposite direction with a double barrel garbage gun.

Storage of waste material for the entire flight can become a serious morale factor.

13.4 CLOSED SPACECRAFT STRUCTURE

Large space stations and interplanetary spaceships preferably would be designed to permit shirtsleeve access to nearly all functional compartments.

13.4.1 Large Vehicle Assembly Operations

The orbital launch facility and space station studies presently are concerned with operational techniques and design constraints involved in piece-by-piece in-orbit assembly of large, prefabricated structures. Many details remain to be worked out.

13.4.2 Docking Shock

See paragraph 13.2.4.

13.4.3 Meteoric Protection

See paragraph 13.0.1.

13.4.4 Portal Reaction Thrust

See paragraph 13.3.8.

13.4.5 Optical Window Degradation

See paragraph 13.3.9

13.4.6 Mechanical Reliability, Rotating Structure

The hub of rotating configurations preferably would be space stabilized to provide a nonrotating, zero-g observation compartment and docking post. This implies a rotary drive mechanism between the hub and the rotating structure.

13.4.7 Portals, EVA

See paragraphs 13.3.1, 13.3.6 and 13.3.8.

13.4.8 Leakage, Atmosphere

See paragraphs 13.3.5 and 13.3.7.

13.4.9 Structural, Integrity

The entire vehicle should hold together without serious distortions, whatever normal maneuvers or external forces are experienced.

A tethered unit presents a special problem in this regard, since any thrusting maneuver could result in either loss or disturbance to stability. Oversights could result in severe damage.

13.5 PIONEER TYPE SPACECRAFT

This is an enclosed can with one axis of symmetry. Internal instruments are thermally isolated from skin.

13.5.1 High Temperature Materials

A probe which is to spend a brief period within 0.1 AU (10 million miles) of the sun is likely to experience skin temperatures in the neighborhood of 2,000 to 5,000° F. Under such conditions the mission would require providing data readout as rapidly as allowable before thermal destruction of the spacecraft occurs. Refrigeration by evaporation of cryogenic liquids might permit slightly prolonging the useful life of the instrument package. However, by providing a high temperature material capable of acting as a shadow shield or solar umbrella, the instrument package might be kept at moderate temperatures during the entire perihelion passage. The shield material would preferably sublime slowly, but even with the best materials available the shield will be massive.

13.6 CYLINDER SPACECRAFT

13.6.1 Vibration Suppression

The precision solar research satellite AOSO will use vehicle attitude motions for a scanning raster to sweep a whole family of telescopes across the solar disk. Any vehicle vibration involved would appear as noise on the data.

Figure 13-22 shows trends in aircraft type dynamic structures.

13.7 APOLLO C&SM TYPE

Of the general configuration of the Apollo command module atop the cylindrical service module, conceived as a logistics vehicle for re-supply of space stations. This will require transfer of cargo, personnel, air, water, and cryogenic propellants to the space station.

13.7.1 Cryogenic Frosting

Some leakage of cryogenic fluids at the coupling interfaces is unavoidable, and indeed serves as venting to avoid overheating. However, the extreme cold near water and air transfer lines (also leaky) can cause frosting which may make uncoupling difficult.

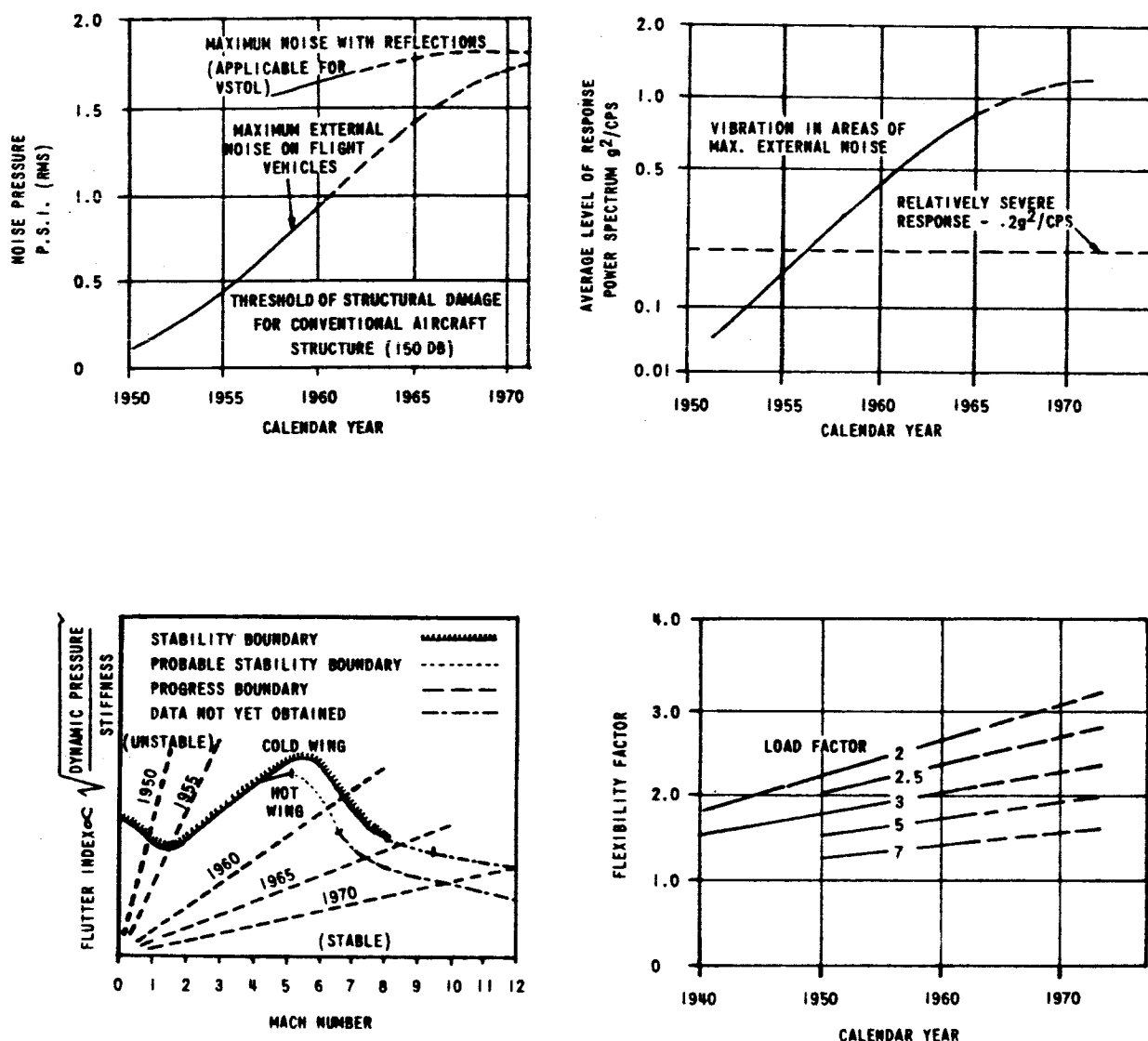


FIGURE 13-22. TRENDS IN STRUCTURAL DYNAMICS
(Unidentified Source)

Strong requirements for advanced technology in the areas of acoustics, vibration, aeroelasticity and dynamics loads are necessitated by requirements for ever expanding performance capabilities of contemporary and tomorrow's aerospace vehicles. The rapid increase in radiated and aero-acoustic noise generated by advanced propulsion systems and fluid flow phenomena is shown in Figure 1. New design concepts, with vertical thrust capabilities, pose even greater requirements for progress in noise control and fatigue resistant design. Significant increases in wide-band, random responses of structure associated with noise are shown in Figure 2. Trends in aerothermoelastic stability are shown in Figure 3. A typical flutter stability boundary is presented in terms of a nondimensional flutter parameter as a function of Mach number. Straight lines emanating from the origin correspond to lines of constant altitude and are related to knowledge and analytical capability for selected time periods. These radial lines terminate approximately at a Mach number corresponding to the maximum progress for the indicated year. The stability criterion is that the value of the flutter index lies below the stability boundary. The "hot wing" boundary represents known and predicted reductions in stability resulting from adverse thermal effects. Difficult technical problems have limited actual determination of the significance of thermal effects at higher Mach numbers. This has generated the new area called aerothermoelasticity. Figure 4 depicts the increasing susceptibility of the primary structure of advanced flight vehicles to dynamic loading conditions relative to flight vehicles of the past.

13.7.2 Complete Expulsion Cryogenics

The useful propellant cargo is that which actually is transferred to the space station. The residual in the tanks of the ferry vehicle is a payload tax. Complete expulsion may require tank lining bladders which remain flexible and completely collapsible at cryogenic temperature of hydrogen. Alternatively, a collapsible bellows diaphragm would exactly match the output end of the tank when collapsed (this is not reuseable).

Expulsion of 98 percent seems possible with present methods.

13.8 LIFTING RE-ENTRY BODY

The vehicle is similar in configuration to the X-20 design, but has the capability for subsonic, powered airplane flight after re-entry. It may be regarded as a jet airplane with re-entry capabilities.

13.8.1 Control Moment Generator

Airplane type control surfaces possibly could not be designed to survive re-entry. Conceivably reaction jets could be used for control, even during subsonic flight. Ablative, replaceable, spoiler tabs may be required for control during re-entry. The problem requires research and development.

13.8.2 Weight, Ablative

Nose, wing edges and other protuberances would require protection by ablative shields during re-entry. Very probably these would be jettisoned after re-entry and prior to subsonic flight. Hence they represent a payload tax.

13.8.3 Research and Development Required, Radiative Re-entry

The entire concept of a manned reuseable booster, rocket airplane, turbo/scramjets, etc. is under preliminary study. Out of this may emerge an acceptable one or two-stage-to-orbit-and-return vehicle which is capable of many such trips and operates from the usual jet airstrips. Development of such a vehicle is expected to lead to eventual phase out of brute force rockets such as the Saturn, Titan and Nova vehicles. But, aerospace technology probably is ten to twenty years from having the required capability.

13. 9 VARIABLE SWEEP WING

The first stage booster of the orbital airplane may be an airplane itself, which is capable of sea level subsonic to Mach 22 flight within the stratosphere. This wide variation in speed likely will necessitate a variable air frame geometry.

13. 9. 1 Hypersonic Release of Second Stage

Firing of second stage engines conceivably could disturb the flight stability of the booster severely. Research in launch from supersonic aircraft may shed light on the existence or nonexistence of serious problems.

13. 9. 2 High Temperature Structures

Refer to Figures 13-23 and 13-24. Speeds and altitudes of air-breathing vehicles are likely to continue increasing. Atmospheric friction will not go away (see paragraph 13.1 .6). Since the booster will be flying at near re-entry velocities, its problems will be similar to that for radiative re-entry vehicles. Structural and skin materials capable of high strength at high temperature are needed to permit weight reductions.

13. 9. 3 Research and Development Required Boundary Layer Cooling

Transpiration cooling (paragraph 13.1 .6) offers promise not only for cooling, but for damping of boundary layer eddy currents with consequent reduction of drag. The concept is being studied with transpiration damping of conventional aircraft wings and transpiration cooling of plasma jet nozzles.

13. 9. 4 Research and Development Required, Variable Sweep Wing

How does one jackknife a wing structure at Mach 20? The F-111 is a beginning and experience with it should lend insight into the faster, hypersonic aircraft.

13. 9. 5 High Strength Light Weight Structures

Stronger and lighter materials are needed in nearly all phases of aerospace engineering. Single crystal fibers imbedded in a carrier metal (or plastic or ceramic matrix) may begin to show promise during this decade.

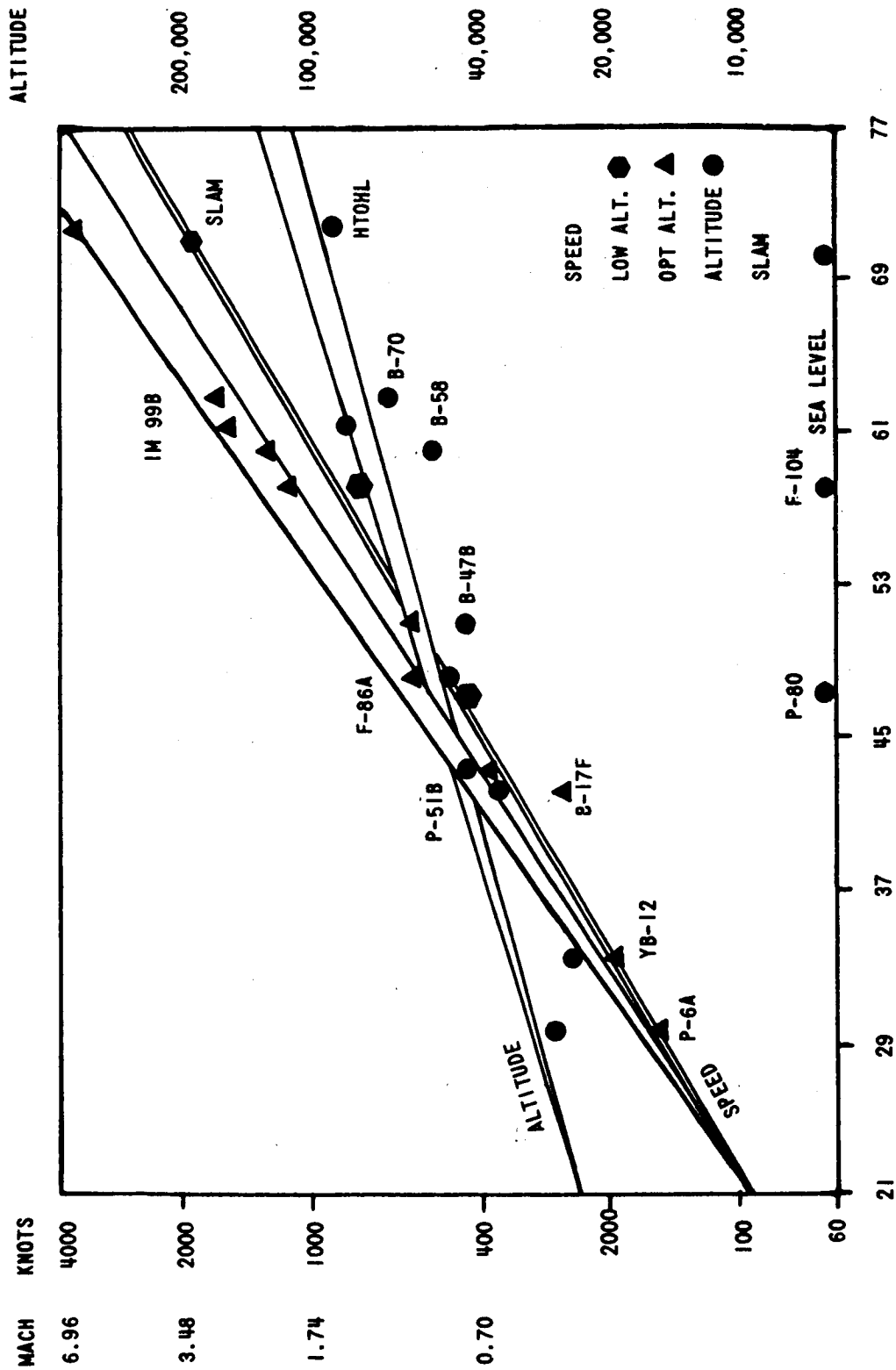


FIGURE 13-23. PERFORMANCE TRENDS
OF AIRBREATHING ATMOSPHERIC SYSTEMS
(Unidentified Source)

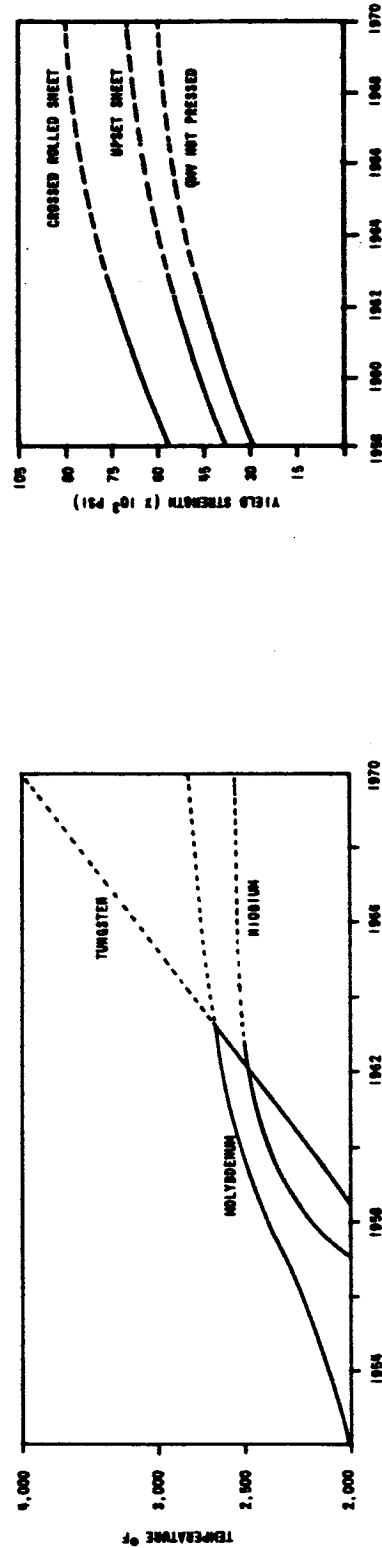
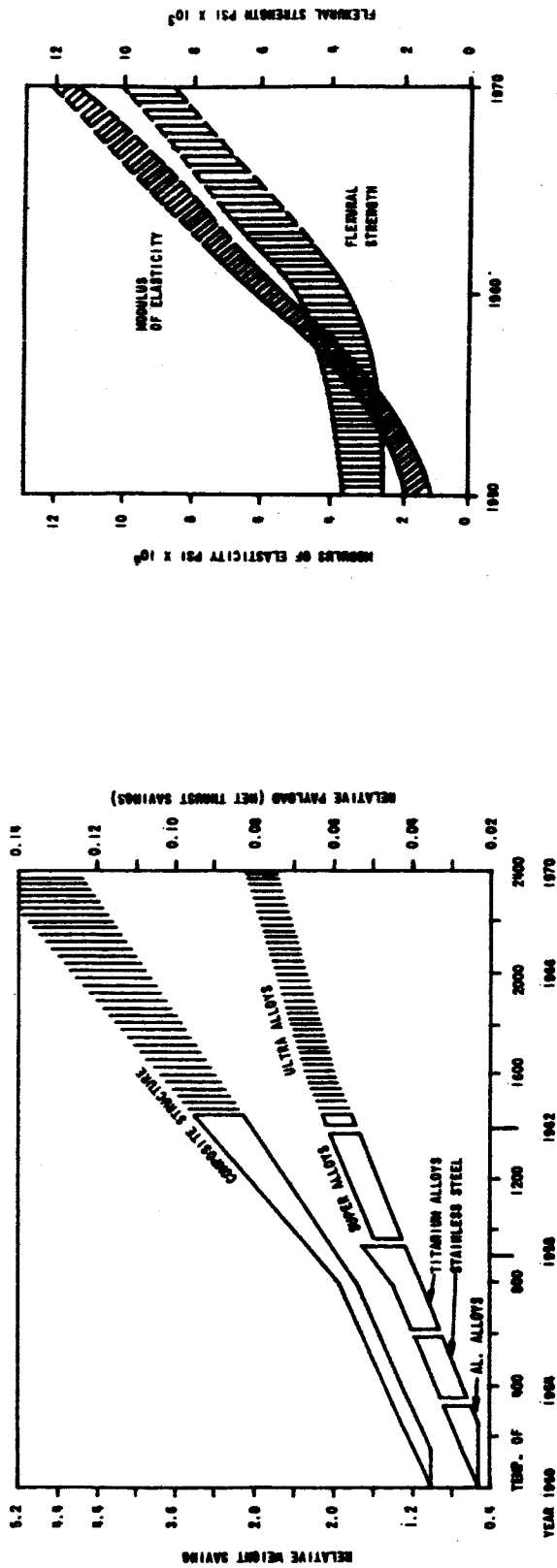


FIGURE 13-24. LOAD-BEARING MATERIALS TECHNICAL PROGRESS TRENDS
(Unidentified Source)

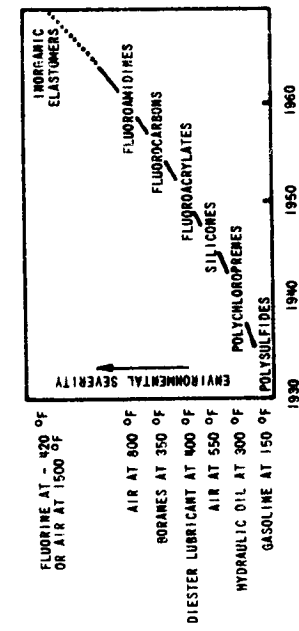
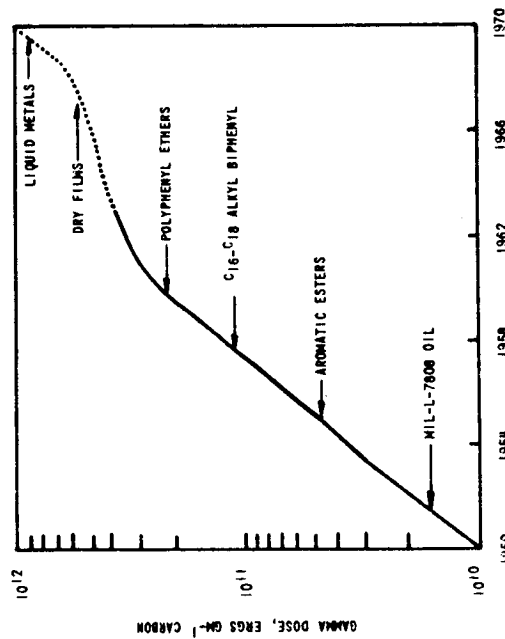
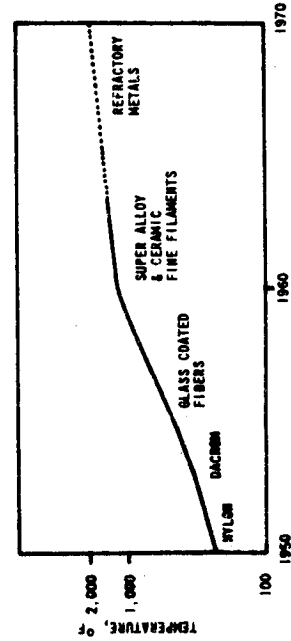
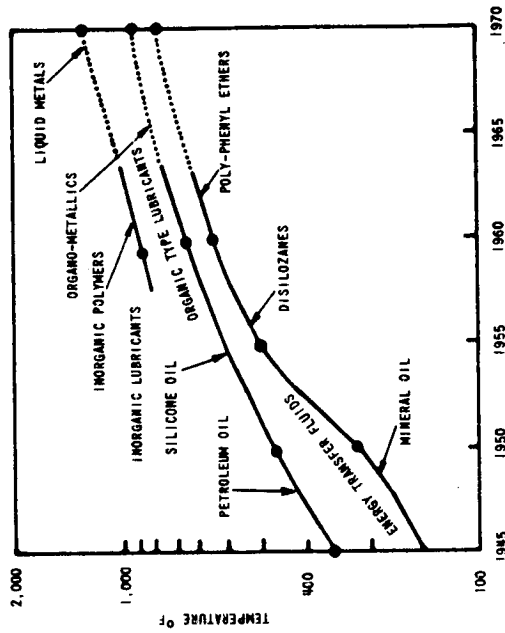


FIGURE 13-25. TRENDS IN FUNCTIONAL MATERIALS
(Unidentified Source)

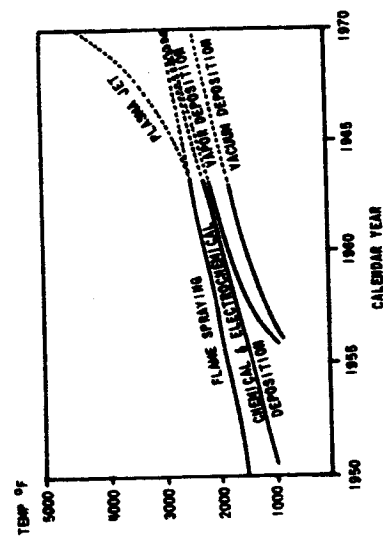
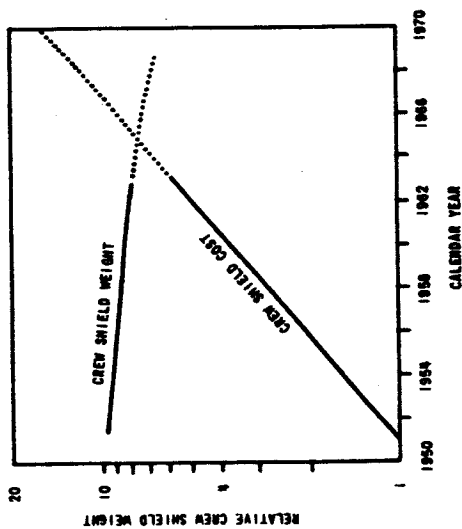
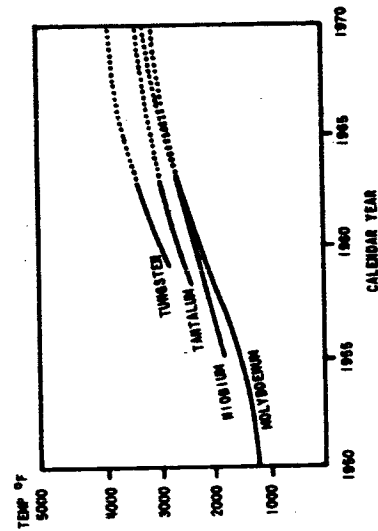
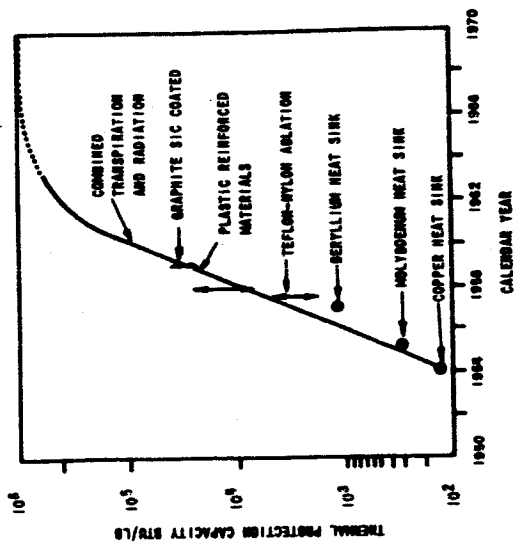
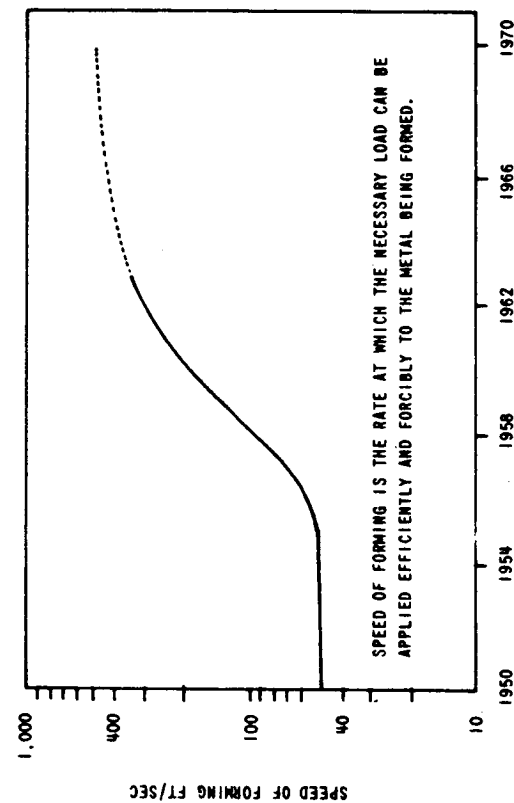
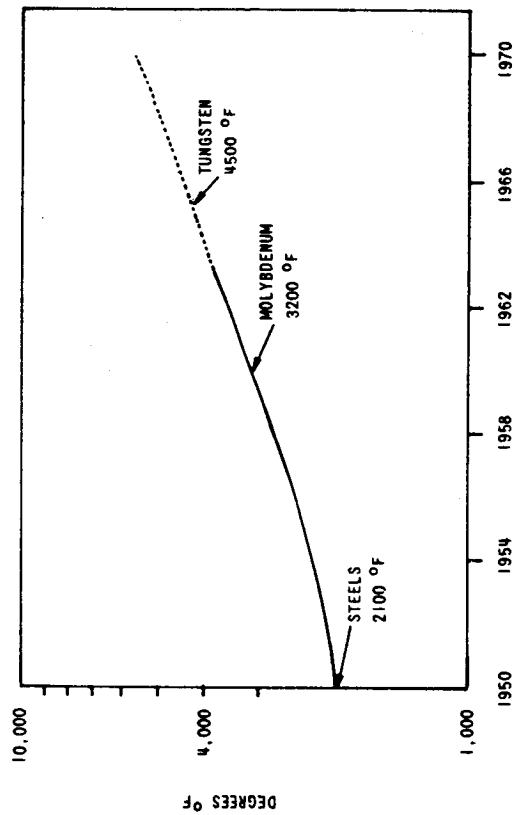


FIGURE 13-26. TRENDS IN PROTECTIVE MATERIALS AND PROCESSES
(Unidentified Sources)



The purpose of highlighting the interchange of materials application information is to reduce as much as possible the time lag between the development of a new material and its practical utilization in Air Force equipment. After a new material has been developed, considerable applied research effort still remains to be done before pay-off in terms of superior weapons systems is realized.

Test tube quantities of a new material may not be representative of production or even semi-production batches; statistically significant mechanical and physical property data must be obtained and provided to designers; forming and fabrication techniques must be worked out; and when extremely difficult requirements must be met it is sometimes possible to arrange or assemble two or more materials so that the better characteristics of one will make up for the deficiencies of another. Progress parameters in this area relate to the above factors.

FIGURE 13-27. TRENDS IN MATERIALS PROCESSING
(Unidentified Sources)

13.10 LOGISTICS VEHICLE STRUCTURE

Basically, this is a cylindrical metal balloon inflated with cargo and propelled by rockets, for system EO08 Z. Its structure probably is similar to that of the Atlas booster.

13.10.1 Cryogenic Transfer

The refueling in space problem. See paragraphs 13.7.1 (cryogenic frosting) and 13.7.2 (complete expulsion).

13.10.2 Cargo Transfer Techniques

Ideally, cargo would be transferred without requiring EVA. This would imply a pressurizable hold and pressure-seal dock-mating with a hatch on the space station.

Otherwise, EVA propellant expenditure may become a problem during unloading.

13.10.3 Cryogenic Storage

See discussion in Section 9. The perfect thermal insulator is needed for wrapping the cryogenic tanks. Venting is the present refrigeration method. Propellant loss due to venting depends upon insulation heat transfer.

13.11 NUCLEAR POWER UNIT STRUCTURE

Several designs for the lunar base power plant have been proposed. Probably it will be a cylinder atop a LEM truck type LLV with a deployable heat radiator as the upper section.

13.11.1 Protection from Landing Shock

This is very dependent upon progress in throttled, radar controlled touch-down. Reactor design should be fail-safe against a 100-foot drop to the Moon. Crushable strut shock absorbers are likely to be used on the LLV, if they prove satisfactory on the LEM.

13.11.2 Structural Integrity

Must withstand all space maneuvers and shock of lunar landing without falling apart.

13.11.3 Ease of Emplacement

Preferably no manual work would be required near the reactor other than deployment of output cables. A minimum of digging and filling should be required for shielding before start up.

13.11.4 Shielding

Lunar soil should provide adequate shielding. This might either be banked around the reactor with a powered back hoe on the LRV or used to fill a double-walled cylindrical caisson provided as part of the vehicle.

13.12 FUEL REGENERATION UNIT STRUCTURE

This unit accepts waste water from vehicles and electrical energy from the power station. By electrolysis the water is separated into hydrogen and oxygen which are purified, liquified and stored for use as fuel cell reactants in the LRV, breathing air or rocket propellant.

If sources of water are discovered on the moon, the FRU becomes the lunar fuel factory.

13.12.1 Research and Development Required, Fuel Regeneration Technique

It's all possible in principle, but many difficulties could beset us while sweating the details of hydrogen liquefaction and cryogenic storage, waste heat rejection, etc.

13.12.2 Waste Heat Rejection

The electrolytic process is inefficient and produces heat along with the H_2 and O_2 , but worse yet, we must remove most of the heat from the H_2 in order to liquify it. All this heat has to be thrown away. Part of it can serve as bath water heat, etc., but the problem requires research and development.

13.12.3 Reliability, Pumps

Several stages of pumping are essential during the entire cycle. These should have a useful life in excess of one year (preferably five) under the severe thermal cycling conditions of the lunar environment. Failure of any pump could mean failure of the FRU.

13.12.4 Leakage

Depressurization could cause freezing of the electrolytic bath as well as loss of H_2 and O_2 . This would put the FRU out of business. Leak detection may be a severe problem in the meteoric environment of the lunar surface.

13.12.5 Cryogenic Storage

See paragraph 13.10.3.

13.13 TRAILER MOUNT

Fuel cells and cryogenic tanks on two-wheeled carts to be towed in trains behind LRVs to extend range of operation from base by providing reserve propulsive power.

13.13.1 Waste Heat Rejection

Fuel cells are only about 50 percent efficient; the rest of the energy from the H_2/O_2 reaction appears as heat which must either be radiated away or convected away by evaporation of cryogenic materials. Possibly radiation at night and radiation plus evaporation in the daytime will be used. Solar heat will be developed in the radiator panels during the day.

13.13.2 Fuel Handling

LH_2 and LOX are transferred to the trailer from the FRU. Water is transferred to the FRU in exchange. See paragraphs 13.7.1 and 13.7.2.

13.13.3 Reliability, Structure

It may be a rough ride for fuel cells and tankage over the terrain on the lunar train. Trailers must function for about 500 miles of such jostling.

13.13.4 Meteoric Protection

The question here is whether meteoric bombardment of the moon is more severe than for space in general (as a result of gravity focusing of particles that fly by the earth toward the moon).

Tanks should be self-sealing. Possibly a layer of moist paper would permit icing at any minor puncture. A major puncture could send the trailer whizzing all over the moonscape by cold gas jet reaction.

13.14 WORKING SPACE

The (LD04)E maintenance shelter is a sort of lunar garage and machine shop which could permit shirtsleeve repair work to be performed on LRV, FRU, trailers, etc.

It may be pressurized with either normal air or with pure helium (during welding operations).

The need for such a working space can be visualized by imagining what would happen to a lunar "shade tree mechanic" whose wrench slipped and let his sleeve get ripped on a sharp corner. Death probably would follow immediately, as a result of a mechanic's usual everyday experience with wrenches.

13.14.1 Portal Air Loss.

See paragraphs 13.3.5, 13.3.7, and 13.3.8.

13.14.2 Environmental Control

Considerable heat may be generated by welding, cutting, brazing, soldering and general machine work. Similarly, noxious gases, smoke and odors must be removed in larger quantity than for living quarters.

13.14.3 Atmospheric Reclamation

As part of environmental control, it would be desirable to reclaim oxygen from CO₂ and CO removed from the air. In the working space considerably more of these may be generated than in living space.

13.14.4 Frosting At Leak Areas

This may become the principal means for leak location. IR image tubes may make leaks visible as cold (black) spots. R&D is needed to provide adequate leak detection instruments.

13.15 OPEN TWO-MAN VEHICLE

A small lunar roving vehicle (or Mars rover) with no closed cabin (space suits required).

13.15.1 Steerable Running Gear

Several methods have been proposed such as: braking one wheel to steer, rotation of two front wheels or two rear wheels or both by torquing at the coupling between sections, automobile type steering, drag plow steering, etc.

13.15.2 Crevice Avoidance

The problem really is one of crevice detection. Flails and even "lead dog" robot vehicles have been proposed, but no simple detector has been published (to the writer's knowledge).

13.15.3 Weight

The small LRV will be limited to about 1,500 pounds at most.

13.15.4 Uncertainty In Lunar Terrain (Or Lurain)

Self-explanatory. Can it support wheels and provide adequate traction?

13.15.5 Sterilization

Preferably the vehicle would remain sterile until manned, so as not to interfere with detection of native life forms. Sterilization after manning probably is vain. Even the cleanest of humans may carry several billion stowaways in or on his body. Chemical antiseptics applied to the exterior of the space unit might help, but it is doubtful that astronauts would bother with it after a day or so.

During unmanned, remote control operation, the vehicle probably could be kept sterile.

13.16 ENCLOSED TWO-MAN VEHICLE

The large LRV or MOLAB with living quarters. For use in extended lunar exploration.

13.16.1 Steerable Running Gear

See paragraph 13.15.1.

13.16.2 Crevice Avoidance

See paragraph 13.15.2.

13.16.3 Weight

Weight will be limited to that of a partial payload for the LLV of about 5,000 pounds. This places limitations on power, comfort, etc.

13.16.4 Uncertainty In Lunar Terrain (Lurain)

See paragraph 13.15.4.

13.17 OPEN ONE-MAN SPACE VEHICLE

Actually, an EVA maneuvering unit which provides adequate propulsion for the pushing, pulling, and stopping required during orbital assembly and cargo transfer.

13.17.1 Grappling Large Masses

Pulling a tank full of LH_2 into place may be easy-- once you get hold of it. Some sort of magnetic induction latch may be needed to permit coupling to flat or curved non-ferrous metal surfaces.

13.18 OPEN TWO-MAN LUNAR FLYING VEHICLE

A two-man rocket supported flying bicycle for rapid transit over rough lurain or to permit reaching heights which otherwise would be inaccessible.

13.18.1 Research and Development Required LFV

All we need is a rocket with a T/W ratio of about 10 (on the Moon) and a specific impulse of 10,000 or so. Otherwise we are severely range limited by available propellant. The lunar "helicopter" is undoubtedly needed; the question of how is not satisfactorily resolved.

A ballistic manned vehicle must have at least four times the velocity change capability of a ballistic missile for the same range, to permit a round trip (two boosts and two complete braking pulses). Propulsion failure would be fatal.

13.18.2 Weight LFV

Neutralization of weight is the core of the problem. With no air to float in or flail against, rocket thrust appears to be the only way. Specific impulse is the core of the rocket problem. Weight of propellant required exceeds that of all other factors.

13.19 EXTENDABLE RUNNING GEAR

Running gear for the Mars or Venus lander is to be retracted during entry and landing. It deploys to provide mobility on the surface.

13.19.1 Mechanical Reliability Of Running Gear

After being "fried" during entry, it may be frozen by convection during atmospheric descent. After having been stored in a retracted condition, in vacuum, for several months it must operate reliably as soon as it is deployed. However, failure would represent only a partial loss of data.

13.19.2 Weight Of Running Gear

Probably in excess of 500 pounds is paid as the price of surface mobility versus a simple lander/setter. In view of the reliability requirements, it is doubtful that significant weight reduction is possible.

REFERENCE LIST FOR SECTION 13

- 13.1 "Aerospace Technical Forecasts 1960-1970 and 1962-1972," Aerospace Industries Association, 1725 DeSales Street N. W., Washington 6, D. C.
- 13.2 Fred L. Whipple, "On Meteoroids and Penetration," Journal of the Astronautical Sciences, Volume X, No. 3, Fall 1963, page 92.
- 13.3 William R. Corliss, Propulsion Systems for Space Flight, McGraw-Hill, 1960.
- 13.4 R. Skags, "RCM Skin and Structural Alloys," RLM Project file, SAS, 14 January 1963 (Honeywell).

Section 14
DISPLAY

14.0 CLASSES OF DISPLAYS

The following arbitrary categories were selected.

14.1 MISSION MANAGEMENT DISPLAYS

14.1.1 Timing Displays

These present time information used in scheduling mission operations. Included are displays of earth clock time, time-to-go until start or completion of critical activities, and significant time-based events such as periods of light and darkness.

14.1.2 Scheduling Displays

These show various activities to be accomplished, time constraints, priorities, and manpower and material resources relevant to real-time scheduling of system operations.

14.2 VEHICLE CONTROL AND GUIDANCE DISPLAYS

14.2.1 Vehicle Orientation Display

This shows present and commanded pitch, roll, and yaw angles and rates for the vehicle.

14.2.2 Vehicle Orbital Parameters Display

This shows quantitative orbital elements such as inclination, right ascension, apogee, perigee, and period.

14.2.3 Vehicle Orbital Position Display

This shows the present position and predicted path of the vehicle in relation to the body being orbited. Geographic and cultural points of interest will also be shown on a map-type display.

14.2.4 Vehicle Navigation Display

This shows the present position, rate of change of position, and predicted path in relation to a specified reference path or points. The specific reference path or points will vary with mission phase (i.e., ascent, orbit, transfer, descent, and landing).

14.2.5 Target Orbital Parameters Display

This shows orbital elements of information of another orbiting object for avoiding it or intercepting it.

14.2.6 Rendezvous Display

This provides information necessary for solving the orbit-transfer problem to intercept a target object.

14.2.7 Docking Display

This shows information, such as radar range and range rate, required to close and dock an intercepted object.

14.2.8 Descent and Landing Display

This shows information, such as present and commanded re-entry angle, re-entry angle limits, altitude, and rate of descent, for controlling vehicle descent to a specified landing point.

14.3 COMMUNICATION DISPLAYS

14.3.1 Internal Communication Displays

These provide for audio and visual communication, including general alarm signals, within the spacecraft.

14.3.2 External Communication Displays

These provide for audio and visual communication between the spacecraft and ground station and other spacecraft.

14.4 SUBSYSTEMS STATUS AND MONITORING DISPLAYS

These are displays for monitoring and controlling the operation of the spacecraft subsystems. The following subsystems may be included:

- a. Environmental control, e.g., spacecraft atmosphere, heat, and pressure.
- b. Propulsion.
- c. Stabilization and control.
- d. Power.
- e. Communication.
- f. Navigation.
- g. Structure.
- h. Sensors, e.g., radar, visual, TV, I-R.
- i. Water.
- j. Waste.
- k. Radiation detection.
- l. Meteoroid detection.
- m. Biomedical monitoring.

14.5 MAINTENANCE DISPLAYS

14.5.1 Malfunction Diagnosis

These are displays for isolating faults.

14.5.2 Malfunction Correction

These are displays for use in correcting or replacing faulty components.

14.5.3 Preventive Maintenance

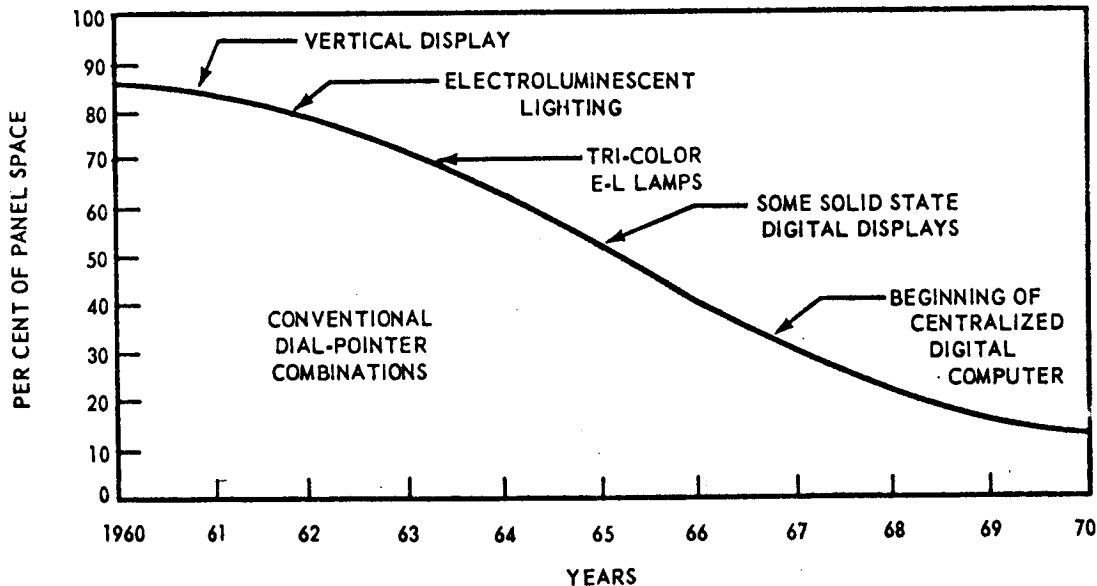
These are displays for accomplishing preventive maintenance of spacecraft subsystems.

14.6 SCIENTIFIC EXPERIMENTATION DISPLAYS

These are special-purpose displays used for the performance of experiments and collection of scientific data.

14.7 SCIENTIFIC DATA REDUCTION AND ANALYSIS DISPLAYS

These are displays to aid data reduction and analysis for guiding further experimentation and for transmission of observations and results to ground stations. Figure 14-1 shows trends in displays.



(SOURCE: Aerospace Industries Association)

FIGURE 14-1. DATA, STORAGE, PROCESSING AND DISPLAY

Section 15

GROUND SUPPORT FACILITY

15.0 GENERAL COMMENT

The general area of ground support includes:

- Vehicle transport and assembly.
- Checkout, testing, and maintenance of all systems prior to launch (see Section 7).
- Mission control through:
 - Schedule and coordination.
 - Launch control.
 - Tracking (see Section 8).
 - Ground computation and data processing.
 - Normal recovery (see Section 3).
- Post-mission review and data reduction.
- Simulation and training (see Section 12).

While all of these areas are important to space flight per se, it is tentatively considered as unlikely that they will lead to significant technology experiments of their own during the Apollo period. For this reason, the whole area of ground support has received minimal research for the purposes of this study.

(The following is quoted from Aerospace Technical Forecast of AIA, Inc.):

"The need for ground support equipment is high today and will increase substantially during the next decade. As the performance of all types of systems is boosted, operating environments are broadened and reaction time is shortened, more support equipment will be necessary to insure the proper functioning of aerospace vehicles. Some of the ground support

equipment must be easily transportable to move with mobile military systems. Some will be stationary at airfields and fixed launch sites. All of it will be automated to a greater degree than today. The rapidly accelerating shift toward automation is the most important trend in ground support equipment today. (Automatic checkout is planned for Apollo. - Ed.)

15.1 SPACE VEHICLE TRANSPORT AND ASSEMBLY

The general trend of boosters to grow in height, in diameter, and in weight influences ground systems in several major ways. In turn, vehicle size is affected by ground considerations-- primarily, transportation of stages from the factory to the launch site and by propellant production and delivery. Until we experience a large increase in propellant specific impulse, we may expect a gradual increase in booster height, leveling off at a height governed by practical crane construction and payload modular weight. The height trend may level off at approximately 350 feet.

Size and weight of certain individual stages already in existence or in design preclude the use of public roads for extended transportation. Certain local and special exceptions may be expected. Tanks larger than 10 feet in diameter will not be accommodated on all public highways or railroads. Other methods of transportation, such as by water and specialized types of aircraft, will predominate. Water will be the primary mode, allowing almost unlimited sizes and weights for individual stages.

The next decade will see serious consideration of the use of airships for special inland flights to deliver tanks or stages to remote testing grounds or to inland launch sites. Airships would find wider acceptance, if they were currently available. Helicopters for stage recovery from the ocean may be developed. In the early 1970's, stage recovery as a winged vehicle will require a recovery runway at or near the launch site. Stage renovation facilities and equipment at that location will be needed.

Water transportation will be the predominant mode for stages powered by large solid rockets, whether the rocket pouring and curing is done at the factory or at the launch site. Rail transportation of solid rocket segments in diameters of 10 feet or less will find limited acceptability.

15.2 LAUNCH PADS

As the diameter of the booster increases to accommodate higher thrust propulsion systems, the height will have to increase proportionally to preserve the required fineness ratio. These larger vehicles will not require a gantry enclosure for support, but inherently will have the capability to withstand the environment-- the wind, rain, lightning, etc. Relatively inexpensive vertical cranes or ginpole rigs will be used to assemble the complete vehicle. Doors at various levels will afford access to engines, equipment compartments, the payload cabin, etc.

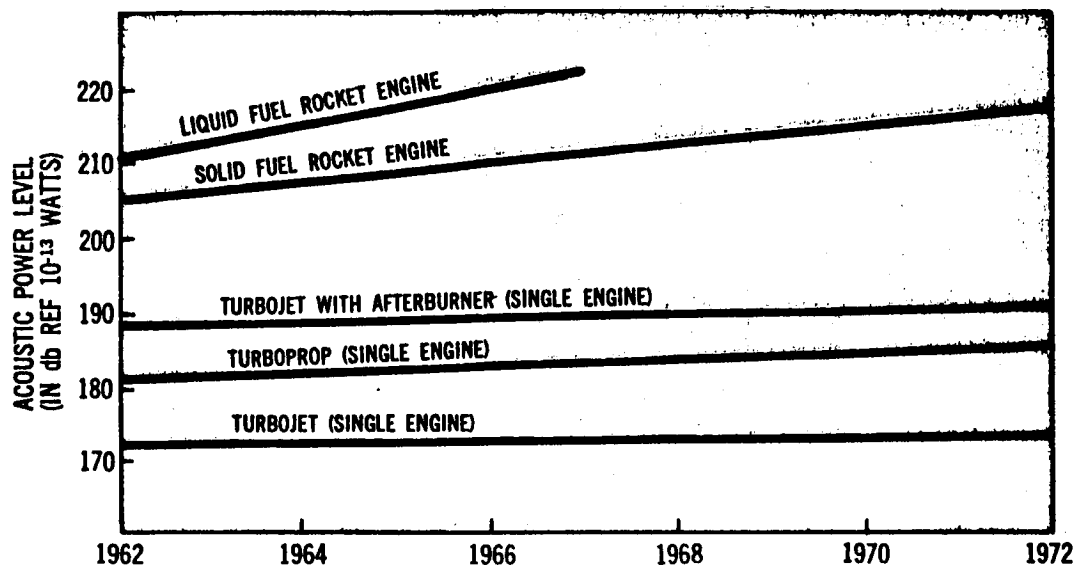
As boosters grow larger, use cryogenic propellants such as liquid oxygen and hydrogen, and have more and larger first stage engines, the launch site hazards will increase.

Real estate requirements at each launch site due to explosion hazards will be maximum when cryogenic propellants are used. Protection from overpressure created by an explosion or by normal engine operation noise during liftoff will dictate launch site separation distances and safety radii for exposed personnel. We may assume that unprotected personnel can safely bear a 120 db sound level and a 2.5 psi overpressure. See Figures 15-1 and 15-2.

Use of a nuclear rocket as a first stage engine will not likely occur until the end of the next decade. Radiation hazards will not be a factor at the launch site until that time, but when a nuclear first stage is used, considerable more land area will be needed to afford personnel protection. Offshore launch facilities and other sea launch techniques will probably be more feasible and economical for this purpose and will be investigated. Nuclear rocket upper stages, energized only after launch, will not be a factor in determining acreage at the launch site.

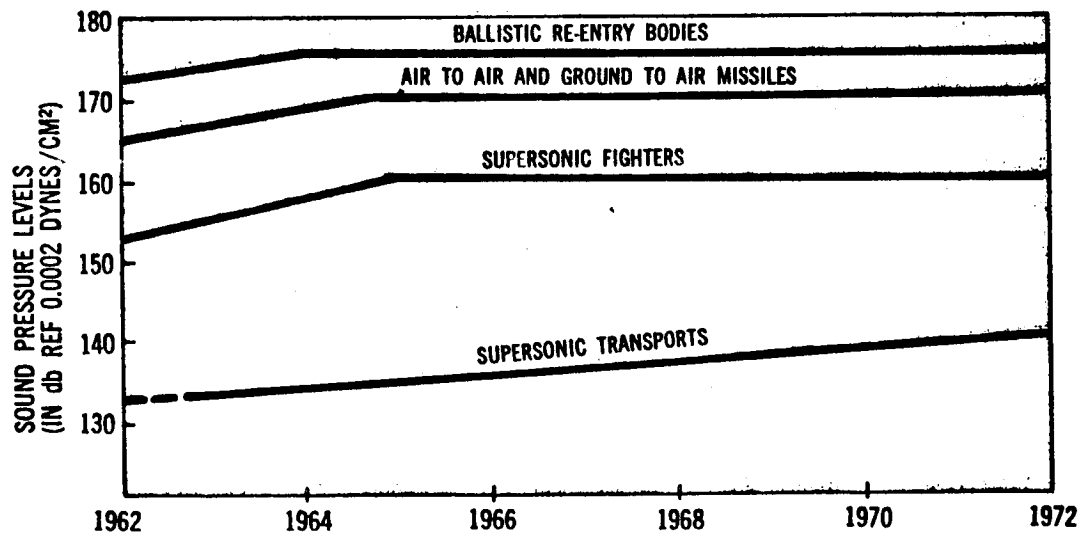
Launch sites today are characterized by large prominent umbilical towers used to support and route, service and control lines to the booster and sustainer stages and to the payload. As boosters grow larger, these lines will not grow in proportion.

There will be a trend to have flyaway connectors, disconnecting at the bottom of the first stage with lines attached to or running inside the structure as an integral part of each stage. The one exception may be lines running to the payload. Payloads will tend to be custom designed, each with different requirements. The booster and sustainer stages will tend to be standardized. The stick, or boom, from the vertical assembly crane will find wide use as a support for payload umbilical lines, as well as an auxiliary path of access from the payload to the ground.



(Source: Ref. 1)

FIGURE 15-1. MAXIMUM ANTICIPATED ACOUSTIC POWER FOR FOUR TYPES OF POWER PLANTS



(Source: Ref. 1)

FIGURE 15-2. ANTICIPATED OVERALL SOUND PRESSURE LEVELS OF TURBULENT BOUNDARY LAYERS

Launch stands in the future will be large and rugged with dry flame defectors and liberal use of reinforced concrete for support columns, foundations, etc. Rapid renovation for repeated launches will be achieved by simplified construction, protection of service and control lines, quick disconnect lines were exposed, and by easy access to maintenance and service areas. When nuclear powered first stages are operated, aluminum superstructures will replace steel. Aluminum retains residual radiation for a shorter period and allows personnel access to a site in a short time after nuclear radiation exposure.

15.3 COST

Figure 15-3 indicates a trend in cost of ground systems as part of total systems cost. In the next 5 or 6 years it is expected that large sums will be spent perfecting and custom-building payloads and boosters. Higher launch rates from each launcher will also be a factor. As a result, existing or presently planned ground systems will have increased use. However, in the later part of the next decade we may expect a whole family of larger boosters requiring new and expensive ground systems, perhaps located at remote sites. Thus the ground system cost will rise slightly.

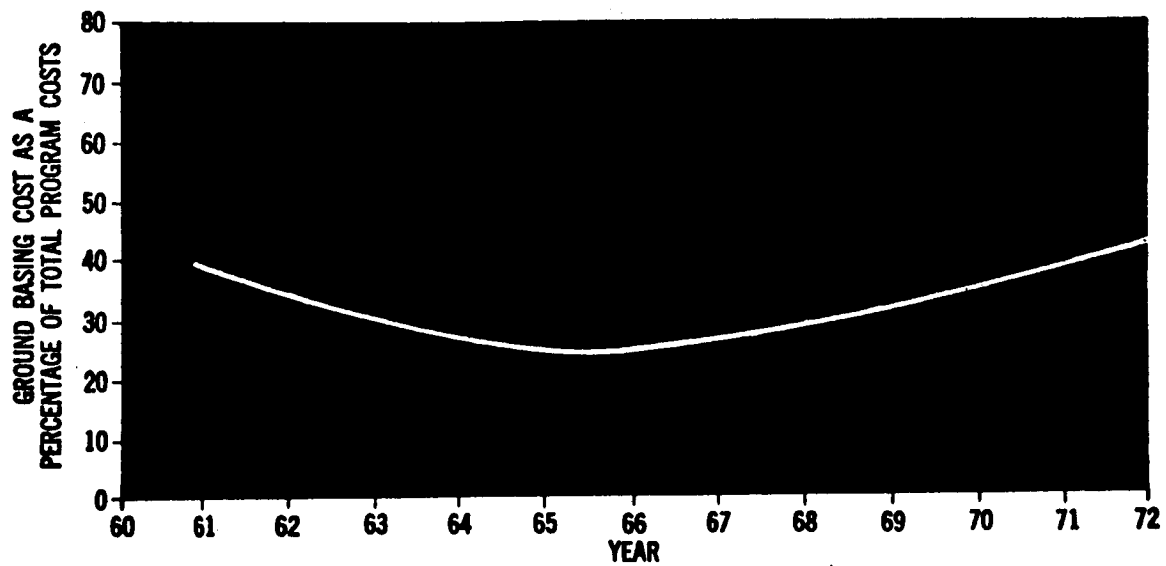
15.4 OTHER SYSTEMS

Such projects as orbiting reconnaissance satellites, and satellite interceptors are now in the research or planning stages. As this decade advances, the delineation of the military mission in space will be clarified and the projects will progress toward production reality. The necessary backup equipment on the ground to support such systems will be similar to the type of equipment necessary for the NASA space mission systems but the additional requirement of fast reaction time is added. Requirements for the equipment will be modified somewhat at the beginning by the experimental aspects of the job but they should mature rapidly if past history is any indication.

(End quotes from AIA forecast.)

15.5 GROUND COMPUTATION

No severe problems are anticipated in ground computation that would lead to space flight experiments to upgrade the technology.



(Source: Aerospace Industries Association)

FIGURE 15-3. ESTIMATED GROUND BASING COSTS
AS A PERCENTAGE OF TOTAL PROGRAM COSTS

15.6 NORMAL RECOVERY

Normal recovery at sea has been adequately demonstrated during Mercury and Gemini flights. Recovery to a land mass has been demonstrated by the Vostok and Voskhod flights.

Development of winged recoverable boosters and two-stage orbital transports will require runway recovery, but again no severe technology problems are anticipated.

15.7 POST-MISSION REVIEW

Post-mission review is primarily a software problem. This technological area will not be prominent until an important space mission fails catastrophically. Whereupon the problem becomes one of reconstructing the entire mission by means of analytical models and extrapolating past the last known data points to permit establishing the probable cause of failure.

In normal operation, this portion of mission control compares the data obtained with the expected results and with the stated mission goals to discover what new and/or unexpected knowledge was obtained and what recommendations should be made for subsequent missions. Data is reduced and prepared for study and reporting by experiment sponsors.