AD-750 554

ADVANCED PROPULSION CONCEPTS - PROJECT OUTGROWTH

Franklin B. Mead, Jr., et al

Air Force Rocket Propulsion Laboratory
Edwards AFB, California

June 1972

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AFRPL-TR-72-31

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ADVANCED PROPULSION CONCEPTS – PROJECT OUTGROWTH

F.B. MEAD, JR., EDITOR

TECHNICAL REPORT AFRPL-TR-72-31

JUNE 1972

AIR FORCE ROCKET PROPULSION LABORATORY
DIRECTOR OF LABORATORIES
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE
EDWARDS, CALIFORNIA

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UNCLASSIFIED
Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) Air Force Rocket Propulsion Laboratory Edwards, California	2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED 2b. GROUP

3. REPORT TITLE

Advanced Propulsion Concepts - Project Outgrowth

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)
Survey Report (1 January 1970 through 31 December 1970)

5. AUTHOR(S) (First name, middle initial, last name)

Franklin B. Mead, Jr., Editor

6. REPORT DATE June 1972	7a. TOTAL NO. OF PAGES 256	7b. NO. OF REFS 119
8a. CONTRACT OR GRANT NO. b. PROJECT NO. c. d.	9a. ORIGINATOR'S REPORT NUMBER(S) AFRPL - TR - 72 - 31 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)	

10. DISTRIBUTION STATEMENT
This document has been approved for public release and sale; its distribution is unlimited.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY Air Force Rocket Propulsion Laboratory Air Force Systems Command, USAF Edwards, CA 93523

13. ABSTRACT

A study was conducted by an ad hoc group within the Air Force Rocket Propulsion Laboratory during the calendar year of 1970 in an attempt to predict the major propulsion developments that may occur in the next 30 to 40 years. This report evaluates the future of conventional chemical rocketry based on thermodynamic principles and revolutionary conceptual approaches to system applications. Advanced concepts falling under the general headings of Thermal, Field and Photon Propulsion are evaluated to a degree necessary to define their potential. ~~This~~ report does not define a long list of very near-term technology program subjects, but is designed to encourage and motivate talented and interested scientists and engineers to once again strive for "Advanced Propulsion Concepts."

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DD FORM 1 NOV 65 1473

UNCLASSIFIED
Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Rocket Propulsion Chemical Propulsion ElectricalPropulsion Nuclear Propulsion						

AFRPL-TR-72-31

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"All men dream; but not equally. Those who dream by night in the dusty recesses of their minds wake in the day to find that it was vanity; but the dreamers of the day are dangerous men for they may act their dream with open eyes, to make it possible."

T. E. LAWRENCE
"The Seven Pillars of Wisdom"

PREFACE

The concepts and analyses presented in this report are the result of considerable effort by numerous individuals. The study of advanced concepts was initiated and directed by Mr. Donald M. Ross, then Deputy Director of the Air Force Rocket Propulsion Laboratory. Listed below are the names of the major individuals whose contributions have made this report possible.

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FOREWORD

This report includes a series of advanced propulsion concepts ated during the calendar year of 1970 under an in-house program « "Project Outgrowth."

This report has been reviewed and is approved.

THOMAS A. McCREERY, Colonel, USAF
Commander, Air Rocket Propulsion Laboratory

ABSTRACT

A study was conducted by an ad hoc group within the Air Force Rocket Propulsion Laboratory during the calendar year of 1970 in an attempt to predict the major propulsion developments that may occur in the next 30 to 40 years. This report evaluates the future of conventional chemical rocketry based on thermodynamic principles and revolutionary conceptual approaches to system applications. Advanced concepts falling under the general headings of Thermal Propulsion, Field Propulsion and Photon Propulsion are evaluated to a degree necessary to define their potential. This report does not define a long list of very near-term technology program subjects, but is designed to encourage and motivate talented and interested scientists and engineers to once again strive for "Advanced Propulsion Concepts."

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INTRODUCTION

The near-term gains to be realized in chemical rocket propulsion will come from evolutionary improvements in propellants, packaging, efficiency, ruggedness and system flexibility. These individual improvements may be combined for specific applications to produce significant increases in range and/or payload capability. In addition, increases in service life with attendant cost reductions are expected to be achieved in the next generation of missile systems. Much of the current Air Force rocket propulsion technology program is structured to make these potential improvements a reality within the next five to ten years. The purpose of the study contained in this document, however, was to identify and stimulate transitions to concepts beyond conventional chemical rocket propulsion which, if pursued, would bring about substantial step improvements in propulsion performance.

During the calendar year of 1970, an ad hoc group within the Air Force Rocket Propulsion Laboratory (AFRPL) conducted a study of "Advanced Propulsion Concepts" in an attempt to predict the propulsion developments and breakthroughs that may occur in the next 30 to 40 years. Various Government agencies, educational institutions, industries, and individuals were contacted and encouraged to submit inventions or suggestions in the area of propulsion and related sciences. Propulsion was broadly defined as any technique for transmitting a mass from one point to another in an aerospace environment. Contributors were encouraged to apply unrestricted thinking in approaching the problem of propulsion. The basic idea was to re-establish the type of free thinking and creativity that existed during the late 1950s and early 1960s, an inventiveness which to a large extent appears to have been lost during more recent times.

The ad hoc committee operated under a self-imposed rule that restricted its efforts to evolutionary chemical rocket concepts offering at least a 25 to 30 percent improvement in propulsion capability. Ideas not meeting this requirement, but offering immediate potential to conventional propulsion were promptly passed on to Staff and Division Offices within the AFRPL for consideration as routine business. Most of the ideas qualifying for committee action were sent to various individuals within the AFRPL for study and analysis. Several ideas that were substantially outside the technical charter or province of the AFRPL were sent to supporting research laboratories such as the Air Force Cambridge Research Laboratory, the Air Force Aerospace Research Laboratories, and the Air Force Office of Scientific Research, for analysis. Ideas that had a good possibility of materially advancing the current state of the art in propulsion were evaluated with a greater degree of thoroughness. To this end, a number of studies involving analysis and experimentation were either made under contract or conducted within the AFRPL.

A dual approach was taken by the ad hoc group. One portion of the effort was devoted to a review of the inherent limitations of chemical energy and its use in rockets. Comparison between the theoretical limits and state of the art established the magnitude of improvements that can be expected from chemical rockets in the future. This work also set a baseline against which advanced concepts could be compared. Ideas concerned with chemical rocketry usually fell into one of two general categories. Rocket propellants were examined to reassure ourselves that no significant improvements had been overlooked in this area. Various novel ideas involving rocket systems and components were examined in an attempt to alleviate one or more of the inherent limitations of chemical rockets.

Advanced concepts, the second portion of the study consisting of propulsion techniques involving energies other than chemical, fell into two general categories. Included under what might be called "Unknown Sciences" fell the concept of psychic forces of which psychokinesis illustrates the extent to which our unrestricted thinking was applied. This category of propulsion is discussed briefly in Appendix I. The second category entitled "Advanced Concepts" covers all those ideas which are governed by the known laws of physics and science. This category is presented in Part II of this report and is further subdivided into groups reflecting certain operational characteristics. "Thermal Propulsion" is characterized by the heating of a working fluid to high temperature, and expanding it through a nozzle. In addition to chemical rockets, a number of nuclear, laser, and electrothermal rocket concepts generally fall in this group. "Field Propulsion" utilizes electric, magnetic, or gravitational fields to achieve a thrust. These fields may produce a force by the acceleration of propellant mass or interaction with available environmental fields. "Photon Propulsion" consists of concepts utilizing light or electromagnetic energy. Under "Unique Concepts" were placed ideas that were not easily categorized by previous definitions. In Appendix I are a number of ideas excluded from in-depth study for the various reasons stated.

The ideas presented in this report do not all warrant the label, "Advanced Propulsion Concepts," and by no means should they be regarded as encompassing all possibilities. They do, however, in our opinion, represent some of the most interesting. The depth of analysis was limited to that required to define a concept's potential. The AFRPL's objective was not to prescribe a long list of very near-term technology program subjects, but to encourage and motivate talented and interested scientists and engineers to once again strive for "Advanced Propulsion Concepts" that are envisioned for use 30 to 40 years into the future.

PART I

CHEMICAL PROFULSION

CHAPTER I-1

INHERENT LIMITATIONS OF CHEMICAL ROCKETRY

In the search for new and advanced propulsion concepts, it seems appropriate to pause and consider in some detail the natural and physical limitations of conventional chemical rockets. By a review and understanding of inherent limitations, it may be possible to identify and develop new chemical propulsion concepts that otherwise would not have been conceived.

THERMODYNAMIC LIMITS

The chemical rocket engine is a heat engine subject to the laws of thermodynamics. The "ideal" thermodynamic process for a chemical rocket converts all the heat of combustion into kinetic energy of the exhaust gas. Unfortunately the laws of thermodynamics are not concerned with the physical method employed to accomplish the acceleration of the gas to its ultimate velocity. In addition, thermodynamics may be used to define the optimum situations. The First Law of Thermodynamics relates the rocket exhaust velocity or gas kinetic energy to the specific impulse which has traditionally been used as a performance yardstick in comparing the propulsion potential of chemical propellants. The ideal conversion of heat to kinetic energy is defined by:

$$Isp = \frac{V_e}{g_c} = \sqrt{\frac{2J}{g_c} \quad (h_c - h_e)}$$

where V_e is the exhaust velocity of the gas or products of combustion, g_c a proportionality constant, J the mechanical equivalent of work, and the term $(h_c - h_e)$ is the enthalpy or energy change occurring between the combustion chamber and the exhaust conditions. This energy term is equivalent to the energy available from the combustion of propellants. Letting the maximum available energy (Δh_r) of a propellant combination be the energy available from an isothermal reaction at $298^{\circ}K$, and assuming all this energy can be converted into kinetic energy (the temperature of the exhaust products is 298°K), a theoretical limit of Isp can be computed from standard thermodynamic tables. The relation between specific impulse and chemical energy in metric units may then be reduced to:

$$Isp = 9.33 \sqrt{\Delta h_r}$$

where Δh_r is the gravimetric heat of reaction in calories per gram at 298°K, and the specific impulse is, as is common practice, based on weight rather than mass. This equation represents the ideal situation. It expresses the theoretical equivalence of energy and specific impulse. Accordingly, this relationship indicates that chemical combinations should be optimized for maximum energy to achieve the highest theoretical specific impulse. This equation thus establishes the maximum theoretical performance limitation for chemical rockets and provides a baseline for predicting possible performance gains.

Since the gravimetric energy release precisely defines the theoretical limit of specific impulse, it is of interest to review the available energies from both common and exotic chemicals. The review should determine where among the elements of the periodic chart high-performance fuels and oxidizers may be sought, and approximately what the upper limit of specific impulse should be. It is emphasized that the heat of reaction is the difference in heats of formation between products and reactants, with the greatest effect coming from the product. Table I-1 shows the gravimetric heats of formation of Group I, II, and III elements listed in increasing order of molecular weight. The steady drop in energy per unit mass indicates that little of use exists below the first two periods. A similar trend exists for fluorides, chlorides and nitrides, the only other elements sufficiently electronegative to consider as oxidizers. Admittedly, approximations have been made to show the usefulness of this table; however, the strength of the trends it reveals must impress even the most open mind.

TABLE I-1. GRAVIMETRIC HEAT OF FORMATION

Group I:

Oxide	Atomic No.	Heat of Formation (cal/gm)
Li_20	3	-4790
Na_20	11	-1610
K_20	19	-1250
Rb_20	37	-420
Cs_20	55	-270

Group II:

Oxide	Atomic No.	Heat of Formation (cal/gm)
Be0	4	-5720
Mg0	12	-3570
Ca0	20	-2710
Sr0	38	-1370
Ba0	56	-870

Group III:

Oxide	Atomic No.	Heat of Formation (cal/gm)
B_20	5	-4340
Al_20_3	13	-3920
Ga_20_3	31	-1380
In_20_3	49	-800

This table does not indicate second order effects which can occasionally change the relative standing of particular elements. For example, a high heat capacity or heat of fusion, or a tendency to vaporize can limit the combustion temperature. The requirement of high energy per unit mass limits the usable elements for propellants to approximately 10 percent of the periodic table.

A comparison between the "theoretical" rocket performance calculated for gases expanded through a nozzle with the ideal performance of conventional propellants gives some indication of the possible performance improvements available. A number of typical rocket propellants are listed in Table I-2 with their corresponding gravimetric energy, calculated theoretical specific impulse, the ideal specific impulse and a comparison between theoretical and ideal impulse (percent efficiency).

It is clear from this table that within the atmosphere, these propellants are inherently capable of much higher specific impulse. However, in space or at high altitude the performance gap is not so great. Delivered performance has not been considered. The performance numbers in this table are based on calculations. A delivered specific impulse which is lower than that calculated will obviously show a greater performance gap than illustrated in Table I-2. The more exotic propellant combinations, such as ozone/beryllium or fluorine/lithium, are examples that illustrate the most energetic chemicals known. This, of course, does not include metastable chemicals which are covered separately in this report. If a technique could be devised for efficient conversion of the chemical energy of these propellants to kinetic energy and thrust, improvements up to 24 percent over oxygen/hydrogen might be realized. It is important to recognize the effect of the square root function involving the propellant energy. At greater propellant energies, the rate of increase in specific impulse is less, and may discourage the costly development of new propellants while offering small increases in chemical energy.

TABLE I-2. CONVENTIONAL AND IDEAL PERFORMANCE COMPARISONS

Propellants	Gravimetric Heat Release (cal/gm)	Calculated Isp 1000 → 14.7/0.2 psia	Calculated Ideal Isp	(Sea Level/Alt) Percent Efficiency
Monopropellants				
H_2O_2	690	165/192	245	67.3/78.4
N_2H_4	833*	216/264	269	80.3/98.1
Bipropellants				
ClF_5/N_2H_4	1713	313/372	386	81.1/96.4
N_2O_4/N_2H_4	1875	291/354	404	72.0/87.6
O_2/RP-1	2445	300/380	461	65.1/82.4
F_2/H_2	3122	410/489	521	78.7/93.9
O_2/H_2	3600	391/470	560	69.8/83.9
F_2/Li	5618	382/472	699	54.7/67.5
O_3/Be	6217	272/358	736	37.0/48.6
Solid Propellants				
$9CH_2/91NH_4ClO_4$	1428	249/304	353	70.5/86.1
$7CH_2/64NH_4ClO_4/29Al$	2017	264/317	419	63.0/75.7

*Maximum heat release ($3N_2H_4 \rightarrow 4NH_3 + N_2$)

In an ideal sense, high specific impulse is associated only with the requirement of high energy per unit weight while the conventional use of adiabatic expansion of combustion products through nozzles imposes a further requirement on the nature of the exhaust products (in terms of low molecular weight, proper specific heat and chemical equilibrium considerations). The requirement for specific properties in the exhaust combustion products is self-imposed by the nature of the machine and is not a theoretical requirement relating impulse and energy. If alternate conversion schemes and equipment could be devised to make use of chemical energy, substantially higher specific impulse could be achieved.

SYSTEM LIMITATIONS

The critical process in the operation of a chemical rocket engine occurs during the combustion process. This event initiates the thermodynamic processes which make a chemical rocket function. During the combustion process, energy is effectively liberated from the propellant reactants and transferred as heat to the combustion products which are expanded through a nozzle. Theoretically, it is desired that the pressure ratio approach infinity. Over the last decade, significant performance gains have been achieved by using higher pressure ratios obtained from higher chamber pressures and improved nozzle expansion ratios. The sensitivity of the theoretical specific impulse to the pressure ratio is significant. As the pressure ratio increases for a given chemical composition, the "theoretical" rocket specific impulse approaches asymptotically the "ideal" thermodynamic limit. This effect can be seen in the data presented in Table I-2. To illustrate this point further, the theoretical specific impulse of a common explosive material "PETN" (pentaerythritol tetranitrate) was evaluated as a rocket propellant under the following conditions: (1) The chamber pressure was fixed at 1,000,000 psi and the exhaust conditions were varied to lower and lower values until a pressure ratio of 1,000,000 to 1 was reached, and (2) the chamber pressure was allowed to increase incrementally with a fixed exhaust pressure of sea level until the pressure ratio reached 1,000,000 to 1.

The two cases showed minor differences in specific impulse values at the lower pressure ratios, but essentially the same values of specific impulse were calculated at the higher pressure ratios. The theoretical specific impulse values from Case 2 are listed and plotted in Figure I-1 to show the relationship between specific impulse and pressure ratio. It is apparent that at pressure ratios above 100,000 to 1, the expansion characteristics and molecular weight of the exhaust species have little influence on the specific impulse for this propellant. This trend implies that if rockets could be operated at these very high pressure ratios the propellant combinations could be optimized for maximum energy and substantially greater impulse.

The fact that combustion and expansion of a hot gas takes place in the presence of solid structures ultimately limits the maximum obtainable pressures and temperatures in a chemical rocket. In liquid rockets, the principal limitations occur in the areas of heat transfer and turbomachinery. With increasing chamber pressure, the heat flux to a constant temperature wall increases due to a larger gas-side film coefficient. At very high chamber pressures, the heat flux to the wall may exceed that which can be conducted through the wall and carried away by one of a number of conventional cooling techniques. Turbomachinery in liquid systems is used to pump the propellants to the high injection pressures required. By one of several techniques, power for the pumping function is obtained from a small percentage of the total propellant flow. As chamber pressures are increased, the power required to drive the pumps increases rapidly. This parasitic effect at high chamber pressure results in an attendant performance loss. Liquid rocket nozzles for specific applications have area ratios selected as a result of various trade-off considerations, including nozzle performance loss below design altitude, nozzle weight and vehicle base drag. By using altitude compensating nozzles, performance losses from the design point compromise may be reduced.

In solid rockets, where high chamber pressure may be sought solely in order to achieve higher specific impulse, additional motor case weight

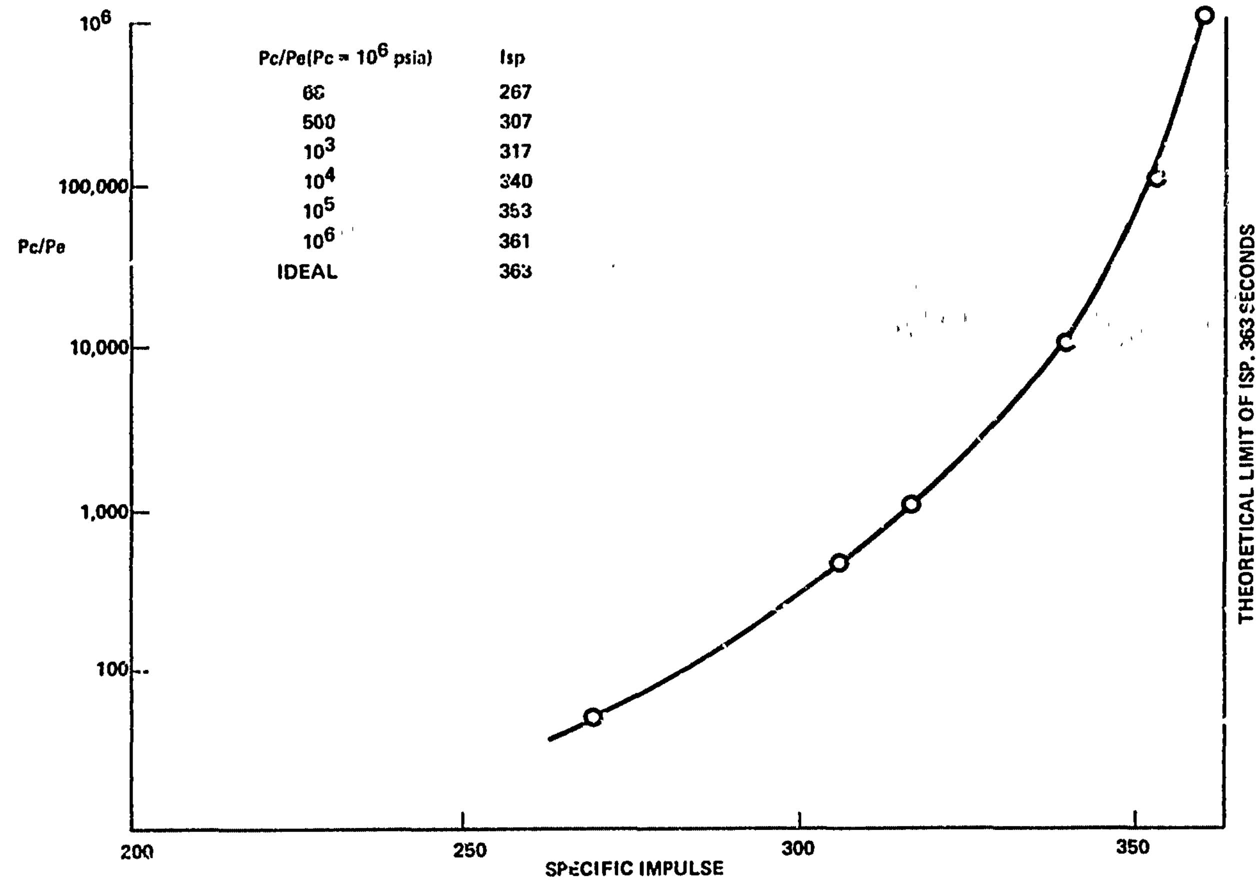

Figure I-1. Effect of Pressure Ratio on Specific Impulse (PETN) is required to contain this pressure. The added weight may result in a net vehicle performance loss as measured by terminal velocity. For solid propellant systems, chamber pressure is also a design variable which permits adjustment of propellant burning rate independent of some other system variables. Hence, the grain shape and, ultimately, the motor configuration are affected by chamber pressure. The nozzles used for solid rockets generally possess no provisions for active cooling or altitude compensation. For most propellant combinations, refractory materials and ceramic or plastic liners are used to handle the associated heat fluxes and thermal stresses. The practical consideration involved with storage of an oxidizer/fuel combination in intimate contact further dictates limits to material selection. Thus, for extremely long burning durations, utilization of high-energy propellants, or operation at high pressures, chamber and nozzle cooling remain a limiting factor.

ENERGY CONVERSION EFFICIENCY OF A ROCKET

Many different methods can be used to evaluate the energy conversion efficiency of a rocket machine. However, it seems of most interest to compare the proportion of energy represented by the mass of chemical propellant that is transmitted to increase the velocity of a payload mass. A chemical rocket propulsion system serves as a transportation system to move a useful package of given mass to a specific velocity or to a definite destination. The propellant weight and its gravimetric energy release defines the total on-board energy available for the mission. A significant portion of the energy is converted to heat and exhausted overboard with the combustion products. Only a portion of the energy is converted to work and an even smaller fraction is converted to useful work in transporting the desired payload package. Typically, 10 to 15 percent of the available chemical energy in a rocket is converted into kinetic energy of the payload mass regardless of whether a three-stage ICBM system or a small tactical rocket is considered. The energy conversion efficiency of a chemical rocket is compared to typical efficiencies obtained in other energy conversion processes in Table I-3.

TABLE I-3. TYPICAL ENERGY CONVERSION EFFICIENCIES

Device	Conversion Process	% Efficiency
Fuel Cell	Chemical to electrical	100
Electric motors	Electrical to mechanical	85 to 90
Guns	Chemical to projectile kinetic energy	30 to 35
Automobile	Chemical to crankshaft work	30
Rocket	Chemical to payload kinetic energy	10 to 15
Fluorescent light	Electrical to light	8
Incandescent light	Electrical to light	2

CHAPTER I-2. ROCKET PROPELLANT IDEAS

The following six concepts are representative of the continuous search for a quantum performance advance in the fields of conventional propellant chemistry and combustion technology. If successfully developed and demonstrated, some of these concepts would place us near the limit of chemical propulsion performance capability using traditional rocket propulsion, thermodynamic cycles and components. Previous attempts to tame these concepts have, however, met with a singularly negative response from Nature. We can only surmise that a barrier has been encountered or that our understanding of the underlying principles awaits further illumination.

Title: Liquid Ozone

Concept: Liquid ozone or mixtures of liquid ozone-liquid-oxygen would be used as the oxidizer in chemical rockets.

Attributes: As an oxidizer with any given fuel, ozone provides the maximum gravimetric energy release. With hydrogen, in particular, ozone theoretically provides the highest performance of any known chemical oxidizer. The inherent increase in specific impulse and density impulse obtained by the use of ozone in rocket systems is ultimately reflected by an increase in payload or ΔV capability. In addition, ozone should, at proper concentrations, provide hypergolic ignition with most fuels.

Analysis: The first serious consideration given to the use of ozone as a liquid rocket propellant was given in a German report (Ref. I-1) by Sänger and Bredt in 1944. They, in describing 10 years of research and analysis of liquid rocket propulsion, reported that with octane as a fuel, ozone gives a 10 percent increase in theoretical and effective exhaust velocity beyond that of liquid oxygen. In this discussion, they report the work of H. Schumacher (Ref. I-2 and I-3) in Frankfurt who had carried out a considerable experimental program to assess the applicability of ozone to rockets. Although Schumacher's work was done prior to 1944, it was not published until 1953. It was in Schumacher's work that the mutual non-miscibility of the ozone-oxygen mixture originally reported by Reisenfeld and Schwab (Ref. I-4) was confirmed and the limits of the two-phase region were determined. In the words of Schumacher, "Liquid mixtures containing up to 25 percent by weight of ozone are stable at -183°C and appear to be safe from explosion. Therefore, they are of importance for technical use. Mixtures between 25 and 55 percent ozone are not stable and split into a light phase, <25 percent ozone, and a heavy phase, >55 percent ozone. Since mixtures containing >55 percent ozone are in danger of exploding, they are without immediate technical value."

More recently, the study of the physical properties of liquid ozone and ozone-oxygen solutions has been the subject of a number of investigations. These works have been reviewed in detail by Hersh (Ref. I-5). Some of the more important properties of liquid ozone compared with liquid oxygen are given in Table I-4.

TABLE I-4. PROPERTIES OF OZONE AND OXYGEN

	Ozone	Oxygen
Formula	O_3	O_2
Molecular Weight	48	32
Boiling Point, $^{\circ}K$	161.3	90.24
Freezing Point, $^{\circ}K$	80.5	54.8
Density, g/cc at $90^{\circ}K$	1.571	1.142
Surface Tension, dynes/cm at $90^{\circ}K$	38.9	13.2
Viscosity, cp at $90^{\circ}K$	1.57	0.189

The liquid ozone-oxygen system is not homogeneous over the entire temperature-composition range. The consolute temperature is $93^{\circ}K$, above which they are completely miscible. However, below this temperature phase separation occurs. In 1955, it was discovered that a low concentration of Freon-13 and CF_3Cl_1, of the order of 3 to 5 percent, added to ozone-oxygen solutions completely eliminates the two-phase regions. Thus, it became possible to handle these solutions without the dangers which would attend the separation of an ozone-rich phase.

The only definitive work which has been reported on the experimental determination of the performance of liquid ozone systems was carried out in oxygen for various ozone concentrations up to 25 percent in an uncooled motor by Princeton University (Ref I-6). The expected improvement in effective exhaust velocity, 2 percent for the maximum loading of 25 percent ozone was actually obtained. In fact, an additional increment of 1 or 2 percent beyond the expected values was actually measured.

This effect was presumed to be due to the improved combustion efficiency seen in ozone systems. The use of ozone/hydrogen systems for space missions offers significant potential. Shown in Table I-5 is the performance potential of several combinations.

TABLE I-5. PERFORMANCE OF THE OZONE-OXYGEN SYSTEM WITH HYDROGEN (LIQUIDS)

Ozone, %	Mixture Ratio	Gravimetric Heat Release (cal/gm)	Theoretical Isp 1000→14.7/.2	Ideal Isp
None (O_2)	4	3220	391/470	527
25	3.87	3750	398/473	571
45	3.50	3880	404/479	581
100 (O_3)	3.50	4236	422/501	607

Studies conducted by NASA showed that low concentrations of ozone (5 weight percent) in oxygen are hypergolic with many fuels. Hydrocarbons were extensively tested; however, no real ignition tests with liquid hydrogen were carried out.

In 1959 (Ref. I-7), the investigation of the physical properties of the liquid ozone-fluorine system was started. Part of the motivation came from the calculations of Huff and Gordon (Ref. I-8). Their study indicated that with a JP-4 fuel, the performance with an ozone-fluorine system maximized at 30 percent ozone. The vapor pressure studies were in good agreement with theory. They indicated, as theory predicted, that mixtures of ozone-fluorine were homogeneous over the entire composition range and that stable solutions existed over the range of temperatures studied. All of the physical property data for the ozone-fluorine system indicate no unusual deviation. While no high-order decomposition was noted, experiments at 30°C showed that fluorine was a catalyst for the thermal decomposition of ozone. Although the point of significant decomposition of ozone in the presence of fluorine was not determined, it was reported to be below room temperature (25°C). Thus, to maintain composition, the liquid mixtures must be kept refrigerated.

The Air Reduction Company (Ref. I-9) used autodecomposition temperature (ADT) tests and shock tube tests to study the sensitivity of ozone solutions. In addition, they made studies on the critical diameter for the detonation of liquid ozone solutions. Some interesting results of the ADT experiments are worthy of mention. The effect of the presence of oxides of nitrogen on the sensitivity of ozone solutions had been debated but never resolved. The ADT experiments showed that these impurities (up to 600 ppm) do not affect the sensitivity of ozone solutions. Similarly, the distillation of the ozone prior to a test does not affect the results. The ADT test did show a significant break at about 55 percent ozone in oxygen which agrees with the previous discussion. Air Research also reported that the addition of fluorine did not alter the results of their ADT tests at similar ozone concentrations. Thus, ozone-fluorine mixtures were equivalent to oxygen-fluorine mixtures. Most significant, however, was their discovery that the addition of 8 to 10 percent fluorine homogenizes the ozone-oxygen system. Thus, it is possible to ensure against phase separation with a high-energy additive instead of an inert freon. The critical diameter and U-tube tests indicated that the system, 10 percent F_2/40 percent O_3/50 percent O_2, could be considered as a candidate oxidizer to upgrade performance and reduce the problems attendant with pure fluorine systems. Some interpolated values for ozone-oxygen mixtures containing 10 percent fluorine are shown in Table I-6. It is interesting to note that the 10 percent F_2/ 50 percent O_3/40 percent O_2 system with hydrogen is "theoretically" only one second lower than the fluorine-hydrogen combination of 410 seconds and three seconds better than 45 percent ozone in oxygen.

TABLE I-6. PERFORMANCE OF OZONE-OXYGEN MIXTURES CONTAINING 10 PERCENT FLUORINE WITH HYDROGEN FUEL (LIQUIDS)

Ozone, %	Mixture Ratio	Gravimetric Heat Release (cal/gm)	Theoretical Isp 1000→14.7/.2	Ideal Isp
0	4.2/6.0	3893	393/471	582
30	4.2/5.8	4107	402/484	598
50	4.2/5.8	4250	408/492	608
60	4.0/5.8	4321	412/496	613

Other ozone systems have been investigated (Ref. I-10). In particular, the ozone-oxygen difluoride system has shown promise. The viscosities, densities and surface tensions showed no deviations from expected values. The vapor pressure indicated, as was the case with fluorine, that the ozone-oxygen difluoride system is completely homogeneous. No sensitivity measurements have been made, but no unusual hazards were noted. Therefore, it appeared, at the very least, that ozone could be used to upgrade the performance of either fluorine or oxygen difluoride.

The preparation of pure ozone and ozone-oxygen solutions in sizeable quantities for rocket applications has been studied in detail (Ref. I-11 and I-12). The report indicates that attention must be given to the complete removal of organic matter from the oxygen used. This allows the production of ozone solutions with very high stability. The production and handling systems must be scrupulously cleaned and passivated prior to use. Passivation is accomplished by passing ozone-oxygen gas mixtures through the system while slowly increasing the ozone concentration. In 1957, an in-depth study (Ref. I-13) of a complete plant layout together with operating costs and material balances resulted in a projected cost of 18 to 20 cents/pound of liquid ozone. Recent commercial ozonators of more efficient design could reduce this figure only slightly.

Shortcomings: The overriding consideration of liquid ozone and its potential use in rocket propellants is its sensitivity toward detonation. Much experimental work has been carried out on this subject. However, little has been resolved. It has been shown that the reduction of organic impurities in oxygen markedly improved stability. But there is really no fundamental understanding of the sensitivity of liquids or solids toward detonation. As a result, there is no sensitivity test to predict the behavior of ozone or ozone-type systems under field conditions. Much work has been carried out to find a "magic" additive which will desensitize ozone. Such work is highly questionable until an understanding of the sensitivity is obtained.

Conclusions: The use of ozone in low concentrations to provide hypergolicity to the liquid oxygen/liquid hydrogen combination can certainly be considered feasible in 5 years. The use of a higher concentration (25 to 35 percent) with some freon additive to provide performance increases will probably be feasible in 15 to 20 years. The use of ozone in low concentrations with other oxidizers like fluorine and oxygen difluoride should also be considered feasible in the near future. Higher concentrations of liquid ozone will need much more research and sensitivity testing before a fair amount of potential can be realized.

Recommendations: The technology is sufficiently advanced so that small-scale development efforts utilizing mixed oxidizers containing ozone could be initiated. Programs could be outlined to investigate ozone sensitivity, phase properties of ozone mixtures, ignition and combustion characterisitcs of ozone mixtures with hydrogen, and production and handling procedures for producing ozone and ozone mixtures. However, the difficulties and complexities of ozone and its development as a rocket propellant must be considered in light of the performance gains available, the time period required for development, and the funds available. Under present circumstances, it appears more expedient to rely on the well-proved propellants and systems within the inventory than to embark on a new, expensive, and moderately high-risk program to develop ozone-based oxidizers for rocket engines.

Title: Stultene

Concept: A stable organic compound, N_6, can be synthesized. This compound would decompose to give three molecules of nitrogen (N_2), and would be used as a monopropellant or gas generator.

Description: Stultene is envisioned as a symmetrical, six nitrogen atom structure similar to the six carbon atom structure of benzene.

Attributes: Stultene promises to be an extremely high-density monopropellant offering chemical energies and ultimately performance far in excess of other state-of-the-art monopropellants. In fact, Stultene could surpass the performance potential of most storable bipropellant combinations.

Analysis: The physical and chemical properties of stultene can be estimated based upon the properties of similar chemical compounds. Benzene, a symmetrical, six carbon structure, has a density of 0.88 gm/cc and melts at 5°C. By the addition of two nitrogen atoms, replacing two carbon atoms located symmetrically in the benzene structure, "1,4 diazine" is formed with a density of 1.08 gm/cc and a melting temperature of 55°C. Thus, the symmetrical addition of nitrogen atom pairs into a benzene-type structure tends to increase the compound's density and melting point. Extrapolation of this trend to the complete six nitrogen atom stultene compound gives a density of about 1.5 gm/cc and a melting point above 100°C. An estimate of the heat of formation can also be made. It is expected that the heat of formation will be large. Based on bond energies, the gravimetric heat of formation is estimated at 2975 cal/gm. Molecular orbital calculations by Shell Chemical (Ref. I-14) suggest that the gravimetric heat of formation is 1297 cal/gm.

Table I-7 illustrates the performance which could be expected from stultene as a monopropellant.

TABLE I-7. MONOPROPELLANT COMPARISONS OF STULTENE

Compound	Gravimetric Heat Release (cal/gm)	Calculated Isp (1000→14.7/.2)	Ideal Isp
H_2O_2	405	165/192	188
N_2H_4	833*	216/264	269
N_6	1297	278/326	336
	2975	396/478	509

$*3N_2H_4 \rightarrow 4NH_3 + N_2$

Shortcomings: Kinetic arguments suggest that stultene would be extremly unstable. In comparison, the decomposition of benzene to acetylene requires that the "trigonal" centers become linear. This linearization requires considerable "steric" strain as two hydrogen atoms compete for the same position. In stultene, however, the decomposition requires only a molecular vibration and movement of electrons. Molecular calculations (Ref. I-15) suggest that while there would be significant resonance energy in the ground state of stultene, the first vibrationally excited state has no barrier to decomposition at all. At room temperature, approximately one percent of the molecules would be in the first vibrationally excited state, and the exchange rate would be so rapid that within a few milliseconds all molecules would have been in the first vibrational state and would have decomposed. Only at temperatures at which no molecules are in excited states would stultene have any chance of being stable. Even the decomposition of one molecule would provide sufficient energy to cause the decomposition of several additional molecules, and thus, catastrophic decomposition.

Even if stultene stability could be controlled at low temperatures, it would have to be used in a solid form. This represents serious handling and storage problems, and probably prohibits its use as a monopropellant.

Conclusions: Although stultene is attractive from energy considerations, it appears to be a very unstable compound. If synthesis of stultene is possible, it will be at very low temperatures with enormous handling problems. At best, it could only be synthesized in very minute quantities.

Recommendations: Enormous amounts of time and money could be spent trying to prepare a sufficient quantity of stultene to provide fleeting spectral proof of its existence. There is no theoretical consideration which lends support to any proposal to develop stultene.

Title: Use of Nitrogen to Support Combustion

Concept: This concept conceives the tailoring of propellants in air-breathing systems to react with nitrogen. In any air-breathing propulsion divice driven by chemical heat release, a reaction zone between the air flow and the fuel can be identified. This is particularly evident for an air-augmented rocket, pictured schematically in Figure I-2. Considering the air-fuel reaction zone, the distinction between reactive and inert nitrogen becomes clear. If atmospheric nitrogen is regarded as inert, it enters the reaction zone as a diluent, reducing the temperature in the reaction zone and generally impeding sustained combustion. Reactive nitrogen, which would combine with the fuel to release heat, could drive reaction zone temperatures up and improve performance.

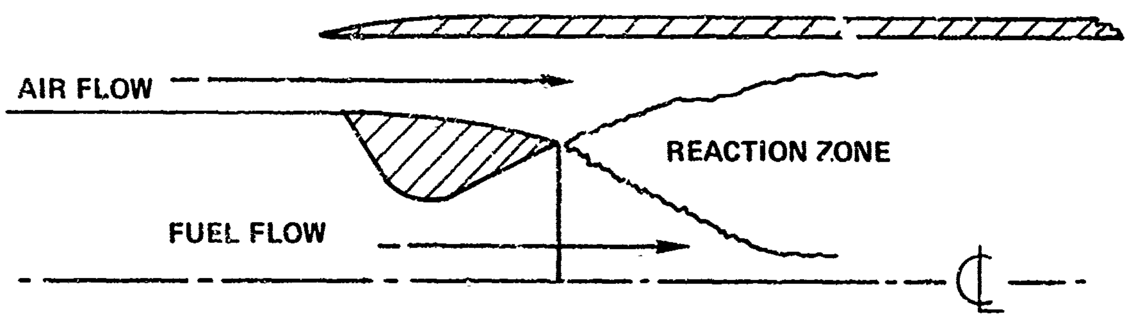

Figure I-2. Air-Augmented Rocket Schematic

Attributes: The potential benefits from reactive nitrogen are an increase in combustion efficiency, the option of operating at increased fuel-to-air ratios, and of course, the possibility of operation in predominantly nitrogen extraterrestrial atmospheres. The resulting system improvements include reduction of inlet size to reduce drag and the extension of the air-breathing flight regime to higher altitudes where the percentage of nitrogen increases.

Analysis: The prospect of increasing combustion efficiency by inducing reaction with atmospheric nitrogen is interesting. Candidate ingredients of a fuel for reaction with nitrogen are reported (Ref. I-16) to be boron, beryllium, hydrogen, magnesium, aluminum and lithium. Because of the favorable heat release with oxygen, boron is a major ingredient of many existing fuels for air-breathing systems. Recent attempts to improve the combustion efficiency of boron fuels at low combustion pressures have resulted in the addition of varying amounts of aluminum, magnesium, magnesium-aluminum alloy and lithium fluoride with good results. It is conceivable that an intermediate reaction with nitrogen participated in the improvement of combustion efficiency; however, additional data are needed to support this hypothesis.

Unfortunately, most of the work which has been attempted to identify nitride formation in air-breathing systems has emphasized detection of boron nitride (BN). It appears, however, that any BN formed in the combustor would have a very brief existence. A temperature of 1300°C is indicated (Ref. I-17) for BN formation form elemental boron. However, it is also reported (Ref. I-18) that BN is oxidized rapidly by dry oxygen and carbon dioxide (CO_2) at 700°C to form boron oxide (B_2O_3) and nitrogen (N_2). Rapid oxidation of BN in moist air is reported to occur at 800°C. These findings suggest that any BN formed during combustion with boron will be rapidly converted to B_2O_3. Another report (Ref. I-19) supports this supposition further in that the combustion efficiency of BN was found to be excellent at reduced pressures where the combustion efficiency of elemental boron is falling off markedly.

Any reduction in drag of air-breathing vehicles which might be achieved by operating at increased fuel-to-air ratios implied by smaller inlets is problematical. It was shown (Ref. I-20) that vehicle drag is relatively insensitive to inlet size because of compensating effects in the boattail region. The ability to reduce drag by reducing inlet size will depend upon the geometry of the related vehicle and the specific design point of the propulsion system.

Shortcomings: The likelihood that the fuel will react with oxygen in preference to nitrogen does not appear promising. The affinity of the fuel for oxygen creates little hope that reactive nitrogen will have a chance to participate in the combustion process unless the reaction zone is too fuel-rich.

Conclusions: The possibility of achieving nitrogen enhanced combustion cannot be completely ignored. Conclusive proof for or against this concept is still lacking To establish the possibility of a nitrogen reaction, it would be informative to conduct a limited number of tests with existing fuels to simulate an all-nitrogen atmosphere. If significant reaction is noted, propellant tailoring to enhance the amount of reaction could then be undertaken. Testing of the resultant formulation in air would then permit a qualitative evaluation of the combustion efficiency attainable by tailoring to a reactive nitrogen propellant formulation.

Recommendations: In the case of predominantly nitrogen atmospheres, such as that believed to surround the planet Mars (Ref. I-16), the propulsion requirements are likely to be identified well in advance of the needed operational capability. It seems reasonable to assume that there will be ample time for propulsion development after the definition of specific mission objectives; and that, consequently, no development effort need be expended prior to the definition of firm propulsion goals. It is concluded that the most meaningful potential benefit of reactive nitrogen in air-breathing propulsion systems is enhanced combustion efficiency of boron-based fuels. It is noted further that a small experimental program might determine the merit of tailoring fuels to be reactive with nitrogen, at least qualitatively.

Title: Tripropellants

Concept: The performance upgrading of certain high-heat-release bipropellants can be achieved by using elemental hydrogen as a working gas. The only important examples are beryllium-oxygen-hydrogen and lithium-fluorine-hydrogen. Lesser tripropellants based on aluminum and boron as fuels might be proposed, but these are not competitive with bipropellants such as fluorine-hydrogen on a theoretical specific impulse basis.

Attributes: The main attraction of the tripropellant concept is the high theoretical specific impulse afforded.

Description: There are many elaborations of the tripropellant concept, all of which solve in some measure the problem of how to combust a fuel which is normally a solid with a cryogenic oxidizer, and then inject a cryogenic fluid as working gas. These elaborations include fluidization of the solid beryllium in a stream of gassified hydrogen, gelling of the beryllium in an organic liquid, formulation of the beryllium metal into a solid propellant grain, and suspension of the beryllium in the liquid oxygen. More direct methods are available for the lithium-fluorine-hydrogen system, since lithium may be melted at reasonably low temperatures and injected as a normal liquid fuel.

Analysis: The tripropellant concept derives its high performance from the combination of the most energetic chemical reaction with the most efficient working gas. Beryllium oxide has a heat of formation of -5720 cal/gram, which makes it the most energetic combustion product known. Lithium fluoride follows closely with a heat of formation of -5680 cal/gram. Neither product has the ability to transform this energy into useful thrust since they are both nongaseous. At high temperatures, some vapor products are formed, but ordinarily the expansion function is served by gases produced from the rest of the propellant. The superiority of elemental hydrogen as a working gas is well known, and when the notion of combining the high heat release of BeO or LiF with the expansion efficiency of hydrogen was conceived, very high specific impulse systems were discovered. Standard sea level and altitude comparisons are given in Figures I-3 and I-4. The gravimetric energy release corresponding to the two systems is 5253 cal/gm and 5145 cal/gm.

Flame temperatures are comparatively low in tripropellant systems because of the large amount of hydrogen and this helps to minimize energy-sapping reactions. The beryllium system fares better in this regard since lithium fluoride melts and partially vaporizes at tripropellant chamber temperatures, and the heats of fusion and vaporization are substantial losses to the system. The main reason for the extra specific impulse of the beryllium system is connected with these losses in the LiF system. AFRPL efforts have concentrated on beryllium-oxygen tripropellants because of the higher specific impulse.

These efforts have been directed to two main problems. These are the physical problem of introducing a solid fuel and two liquid fuels into a combustion chamber, and the problem of low combustion efficiency generally observed with beryllium. The additional problems of beryllium toxicity and low system density characteristic of hydrogen are accepted as inherent and only bear on the appropriateness of this combination in a given application.

In summary, these efforts resulted in satisfactory progress on the first problem, but failed to solve the second. Beryllium was successfully fluidized in a stream of gaseous hydrogen and injected into a rocket chamber. Gels of up to 60 percent beryllium were made (Ref. 21)-- too low to offer attractive performance. A monopropellant slurry of beryllium in liquid oxygen was formulated and combusted (Ref. 22). The high burn

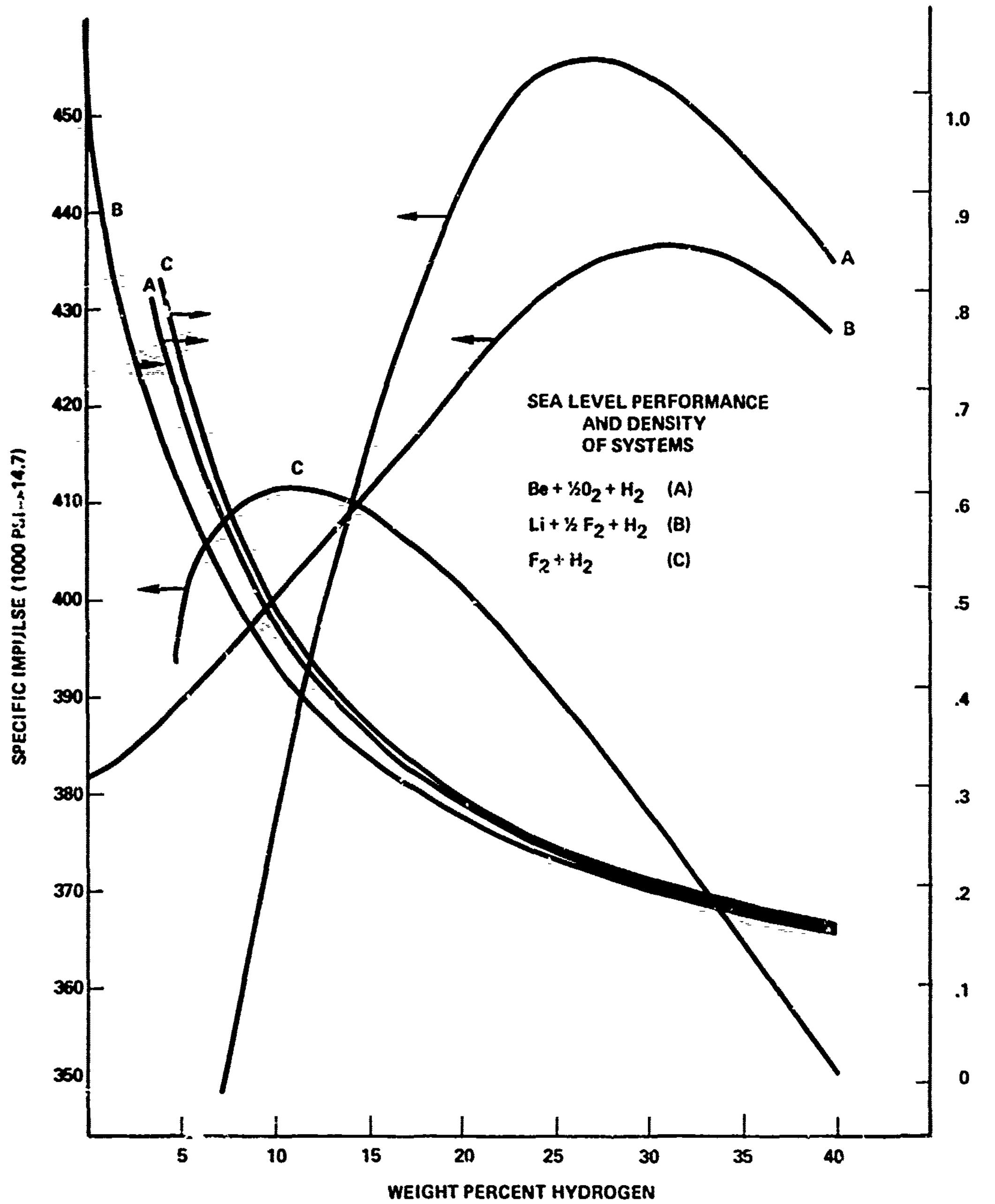

Figure I-3. Sea Level Performance and Density of Systems

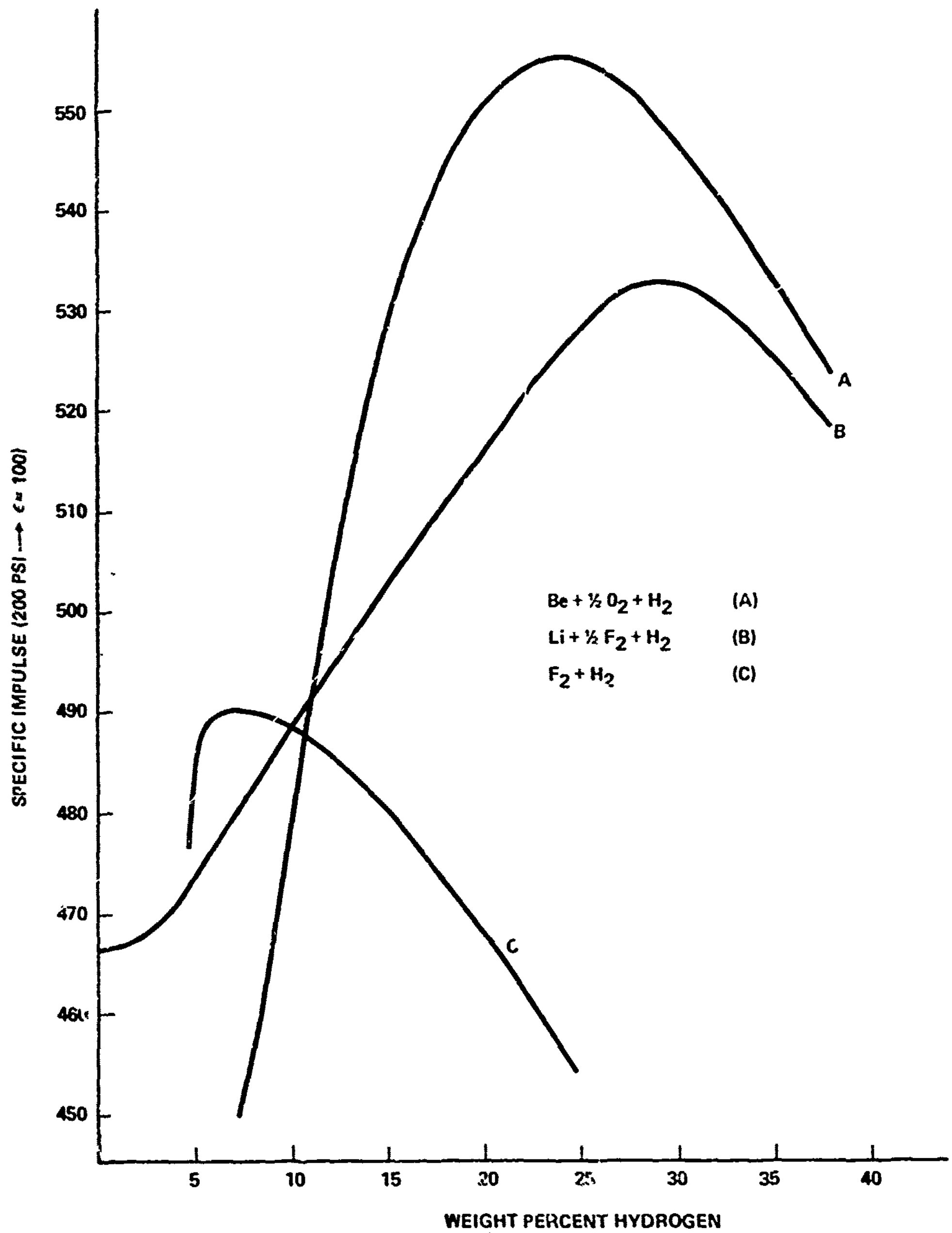

Figure I-4. Altitude Performance of Systems rate of this propellant was sufficient to propagate back up the injector orifices and on to the tanks unless injection velocities were kept high. A hybrid tripropellant known as the tribrid rocket was devised (Ref. 23) in which up to 90 percent beryllium in a hydrocarbon binder was combusted with liquid oxygen. None of these efforts demonstrated acceptable combustion efficiency.

The tribrid concept was selected for continued study at a larger scale. Out of this program came an upgraded solid propellant grain (Ref. 24) of 95 percent beryllium. However, firings at the 1000-pound-thrust level failed to achieve combustion efficiencies greater than about 85 percent.

These efforts were conducted in a period of high interest and concerted inquiry into the combustion characteristics of metals in general and beryllium in particular. A variety of solid and liquid propellant programs and fundamental laboratory studies pointed to similar conclusions: complete combustion of beryllium to its oxide in practical rocket chambers is not attainable. Higher efficiencies with beryllium have been obtained than the tripropellant program demonstrated; but these invariably were brought about by operating away from theoretically optimum mixtures, by overoxidizing to boost flame temperature (and reducing beryllium content in the process).

With this background a tradeoff study of the advantages and disadvantages of the beryllium-oxygen systems becomes very straightforward. If it is agreed that only realizable specific impulse is meaningful, then the $Be\text{-}O_2\text{-}H_2$ system has no advantage over the hydrogen-fluorine system. Its delivered specific impulse is at best equal to that of $H_2\text{-}F_2$, despite its high theoretical potential. On the other hand, its density is less than half that of the $H_2\text{-}F_2$ system, posing a severe disadvantage in mass fraction. The extent of this disadvantage is revealed in Figure I-5 which shows that in payload ranges typical of the Transtage application, the beryllium-oxygen-hydrogen system would require 95 percent combustion

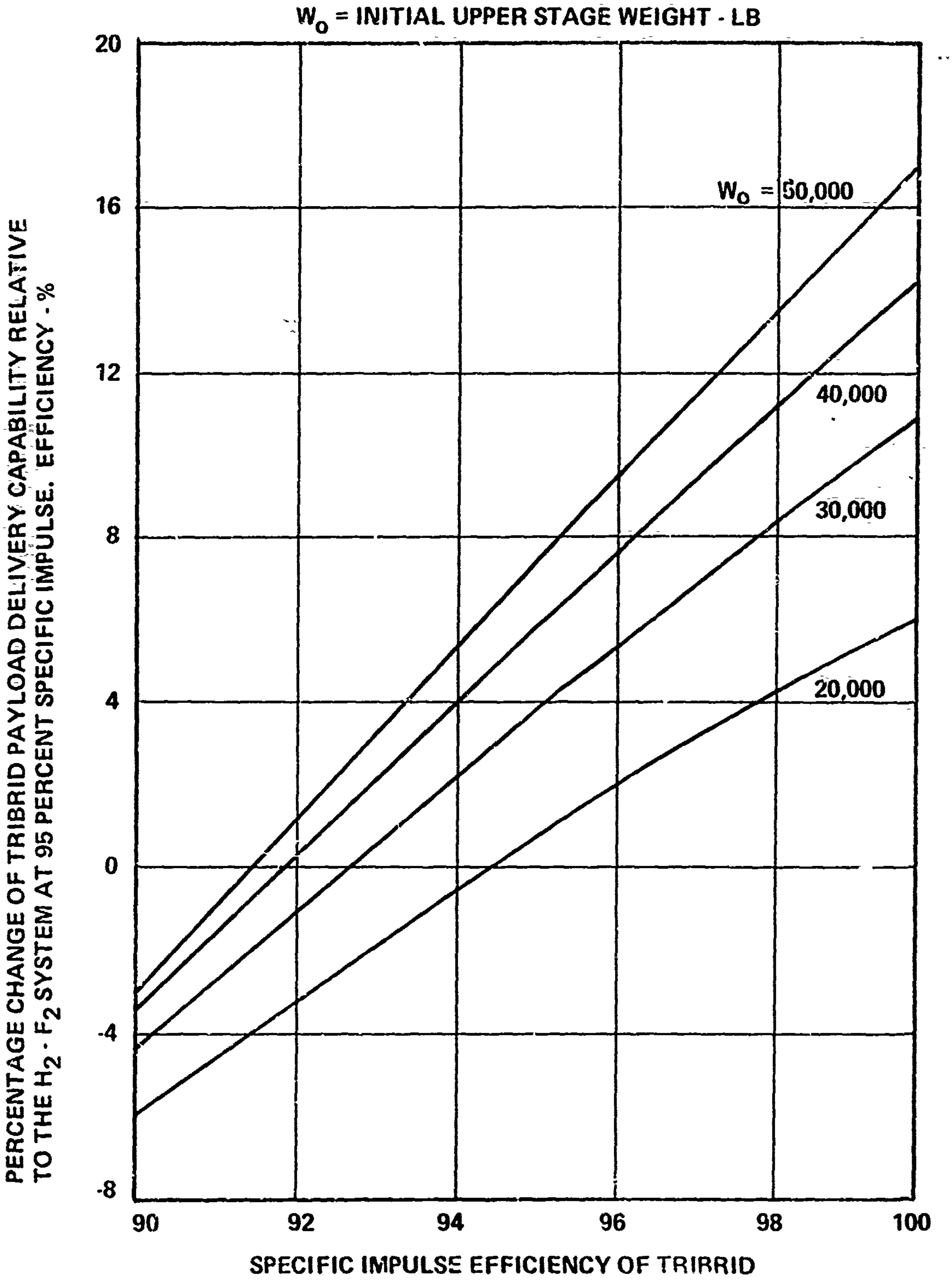

Figure I-5. Payload Advantage With Tribrid (Be-O_2-H_2) System Relative to Hydrogen - Fluorine System as a Function of Tribrid Impulse Efficiency

efficiency to equal the payload capability of hydrogen-fluorine. Even at 100 percent combustion efficiency only a 7 percent gain in payload may be obtained.

The lithium-fluorine-hydrogen tripropellant has been examined under NASA sponsorship. Engine studies have shown that this system is capable of much higher combustion efficiency than the beryllium system. Tradeoff studies have shown (Ref. I-25) that payload advantages of 5 to 10 percent are possible in direct-ascent, short-duration missions as compared with F_2-H_2. However, when long-duration space missions are considered, meteoroid shielding requirements for the oversized hydrogen tanks, plus loss of hydrogen by boil-off, reduce the payload capability below that of fluorine-hydrogen. The conclusion of that study was that the modest advantages of this tripropellant system did not warrant the cost of developing it.

Shortcomings:

A. Be-O_2-H_2 System

1. Low system density and resulting low mass fract...
2. Low combustion effi ciency
3. Complex feed systems
4. Beryllium toxicity

B. Li-F_2-H_2 System

1. Low system density
2. Complex feed system (need to melt lithium)

Conclusions: In the absence of new principles, it does not appear likely that the delivered pe. formance of the Be-O_2-H_2 tripropellant can be elevated to a useful le..l. Basic studies of beryllium particle combustion have indicated that burning is limited by the accumulation of an impervious "slag" of BeO on the fuel particles. The tripropellant system is more sensitive to this problem than usual, since the only source of energy is the oxidation of beryllium; there are no other products, such as CO_2 or H_2O to cover up the deficiency. The low mass fraction reduces the impetus for further concerted study to solve the problem.

A 5 to 10 percent advantage in payload is possible with the $Li-F_2-H_2$ system in short-duration missions.

Recommendations: No further funding in this area is warranted in the light of present knowledge. No missions are known which cannot be performed with more conventional propellants. The slight advantage which is possible with the $Li-F_2-H_2$ system in certain systems is not sufficient to compensate for the developmental costs and added systems complexity. The possibility that the beryllium combustion problem will be solved at some future date cannot be discounted, but active work to achieve this solution should be viewed as a fundamental effort.

Title: Solid Hydrogen Propellant System

Concept: The propellant hydrogen can be stored in its solid state form for long-duration space missions. The hydrogen could be stored in a tank where it is melted as required for use as either a liquid or gas. In the high-thrust case, hydrogen is stored as a solid and used as a liquid to feed a pump-fed engine. The liquid would be produced on demand by bleeding hydrogen gas from the cooling jacket of a regeneratively cooled engine and spraying this gas on the solid hydrogen to melt it.

Attributes: The use of solid hydrogen in rocket engine systems provides a storage density increase of 22 percent over liquid hydrogen. However, the most important feature is the improved storability characteristics because of the added latent heat of fusion. Propellant losses during long-duration space missions would be greatly reduced by using solid hydrogen as opposed to liquid hydrogen. For instance, preliminary calculations indicate that the loss rate of hydrogen can be reduced by as much as 43 percent (Ref. I-26).

Description: The specific application envisioned for storing solid hydrogen and supplying this propellant in sufficient quantities to the engine utilizes a helium-cooled storage tank for the solid hydrogen. Melting of the hydrogen is accomplished by impinging warm hydrogen gas on the solid material. Other melting techniques employing conduction and radiation might also be considered.

Analysis: While freezing of the hydrogen is a straightforward process, there is some question whether the solid's melting rate can be accomplished at the exact rate required for use by the engine. To answer this question, detailed knowledge of the heat and mass transfer at the surface of the solid hydrogen is required. Only a limited amount of thermal-physical property data is available (Ref. I-27). Preliminary feasibility studies based on these data indicate that melting of solid hydrogen may be accomplished quite rapidly. However, complete substantiation of this concept must be based on experimental measurements. In addition, a helium cooling system must be used to initially freeze the hydrogen. Studies (Ref. I-26, I-28 and I-29) conducted by the AFRPL using the Orbit-to-Orbit Shuttle (OOS) vehicle show that the effects of the additional weight to the vehicle may actually result in a performance loss (ΔV) of mission capability lasting for short durations (less than 50 days). Substantial increases in mission capability are still available for long-duration missions (greater than 200 days).

Shortcomings: Aside from the lack of demonstrated melting rates to verify that sufficient flowrates for large engines are obtainable, there is a significant weight penalty resulting from the heat exchangers necessary to accomplish the melting and freezing processes.

Conclusions: The use of hydrogen propellant stored in solid form appears to be an attractive concept even though there may be significant weight penalties to the vehicle system. Experimental data need to be obtained to lend credence to preliminary calculations.

Recommendations: A program should be established to measure the thermal physical properties and the melting process of solid hydrogen. Grumman Aerospace Corporation is currently conducting a program under Air Force contract to: (1) measure the thermal conductivity and thermal diffusivity of solid hydrogen over the range of temperatures from 10.8°R to 25°R, and (2) investigate the process of melting solid hydrogen from a gaseous hydrogen stream.

Title: Metastable Ingredients

Concept: Development of a propellant based on the energy contained in the metastable states of atoms and molecules.

Attributes: A significant increase in Isp over conventional propellants using chemical bond energy.

Description: Propellants in common use today derive their energy from the making and breaking of chemical bonds. However, greater energies may be obtained from free radical recombination, metastable species and other energy storage concepts.

Analysis: Among the most energetic chemicals available for propellants, approximately 3 Kcal/gm of energy is available from liquid propellant combinations such as H_2/O_2 or H_2/F_2. Solid and monopropellant systems are correspondingly lower. If it is assumed that the engineering and material problems can be solved, then energy densities up to 6.2 Kcal/gm can be obtained from the systems Li/F_2 and Be/O_3.

In the search for chemical propellants with greater energy densities, the Department of Defense studied unusual chemical species. Initial work concentrated upon free radicals (an atom or molecular fragment containing an unpaired electron) which if allowed to recombine can supply significant amounts of energy. The energy from these species is obtained when the fragments recombine on the unpaired electrons from a normal bond pair. The highest energy available is from H atoms which on recombining produce H_2 and 52 Kcal/gm. These studies were undertaken at the National Bureau of Standards between 1957 and 1959. The major result of this effort was that many free radicals were produced and stored in inert matrices at very low temperatures (usually a few degrees Kelvin) but never in concentrations high enough to be of practical use (concentrations never exceeded 0.1 percent).

In the early 1960's the Air Force Rocket Propulsion Laboratory reviewed the work of H. Damianovich in which he claimed to have made a variety of heavy metal helides. Since helium is normally an inert gas, it was hoped that these compounds might be relatively energetic. Attempts to reproduce Damianovich's work were unsuccessful. Many unusual experimental results were observed, but no compound formation was ever demonstrated.

During the research with helium, one contractor conceived the idea of producing a "plasmasol." This is a solid plasma which would consist of alternating layers of helium ions and electrons sandwiched between layers of an insulator. The insulator chosen was ceresin, a mixture of naturally occurring, high molecular weight, straight chain hydrocarbons. The normal heat of combustion of this material is 11.2 Kcal/gm. The contractor initially reported heat of combustion values larger than this by a factor of 5 and suggested large amounts of energy had been stored. Subsequent studies indicated, however, that these values were probably due to calorimetry errors, and only small energy density values (less than 0.7 Kcal/gm) were demonstrated.

During this same time period, other means of energy storage were sought. The only other concept studied in detail was metastable species. When an atom or molecule becomes excited, that is, becomes more energetic, it may do so in a variety of ways. If the species under consideration is an atom, it can move into higher (more energetic) translational (linear motion) or electronic (electron orbital) states. If the species is a two-atom (diatomic) molecule, it may also move into higher vibrational states (all atoms and molecules not at absolute zero contain some vibrational energy). Finally, if the molecule contains more than two atoms, rotational states are also possible. The study was limited to electronic metastable states. When an atom moves into these higher states, it may immediately relax (change to a lower energy with consequent energy release), or it may remain in this higher energy level for some time. If the latter occurs, the species is said to be metastable. Non-metastable states usually relax in very short periods of time, usually less than 10^{-6} second. Metastable states occur on a much larger time scale, usually 10^{-3} to 10^{-2} second. Finally, if the molecular weight of the species is low, large amounts of energy per gram can be obtained. Numbers between 50 and 100 Kcal/gm are possible.

All of the description given above relates to metastable molecules in the gaseous phase. If a propellant is to be developed, it presumably would need to be in a liquid or solid phase for density reasons. Metastable states have been reported for several diatomic molecules in the condensed phase. For example, Livensky (Ref. I-30) has reported a spectrum of ZrO in which one of the spectral features is believed due to a metastable state. Robinson and McCarty (Ref. I-31) have reported a spectrum of NH in which one of the spectral features is ascribed to a metastable state. To examine the possibility that storage of metastable states in the condensed phase is possible, General Electric attempted to repeat the earlier NH work of Robinson and McCarty. They also examined several other likely metastable areas, including N_2, BH, AlH and O_2. Each of these species was produced, trapped in an inert gas matrix and examined for spectral features that would indicate the existence of a metastable state. In some cases, the deposit was excited with an electron beam in an attempt to produce metastable states. No conclusive evidence for long-time stabilization of an electronically metastable molecule by matrix isolation techniques was obtained during the course of these investigations Because of the lack of positive results, the effort was concluded at this point.

Shortcomings: To date, there is no unequivocal evidence that conditions can be found which would allow metastable states to exist for realistic periods of time. Nor is there significant proof that metastable species can be retained for any significant period of time in a condensed phase.

Conclusions: This field of endeavor currently holds much potential but little promise. Any feasibility demonstration of this concept does not appear likely within the next 20 years.

CHAPTER I-3. ROCKET SYSTEM IDEAS

The previous chapter dealt with the production of chemical energy. The concepts in this chapter serve to illustrate the variety of possibilities that exist for utilizing the chemical energy. These concepts generally attempt to alleviate some limit imposed by conventional chemical propulsion. These limits include expansion ratio and mass fractions. Detonation characteristically achieves pressure ratios on the order of 1,000,000 to 1. By eliminating nozzles or cases, or infinitely staging a vehicle, improved mission performance may be obtainable.

Title: Infinitely Staged Rockets

Concept: The concept of infinite staging assumes that those portions of the rocket vehicle which have already performed their function and are no longer needed are ejected or consumed continuously throughout the burning time. At a point in time when a particular mass no longer serves a useful function, it becomes "dead weight" and detracts from the performance of the rocket vehicle by unproductively consuming energy. An infinite staging propulsion system would continuously burn or jettison unproductive mass and thereby improve performance.

Attributes: Infinite staging improves the flight performance of chemical rockets in terms of their achievable velocity increment (ΔV) and payload capability. It is essentially a more efficient method of utilizing the available energy from any given propellant combination. It represents the way to higher energy missions that could not be accomplished with an equivalent single stage no matter how large the vehicle.

Description: For a liquid rocket, the dead weight consists mainly of empty tankage. Therefore, an infinitely staged liquid rocket would eject or consume empty tankage. In a solid rocket, most of the dead weight is associated with the motor case. Therefore, an infinitely staged solid rocket would eject or consume that portion of the case that is no longer needed. The maximum possible performance is achieved if the inerts can be jettisoned from the vehicle without a performance penalty.

Analysis: The use of staged rocket vehicles is a concept that is almost as old as rocketry itself. The conventional method of performance improvement is to expend entire stages of multistage rocket vehicles at discrete points in the trajectory of the vehicle. Designers of solid propellant motors and liquid propulsion engines have continuously tried to improve vehicle performance by employing concepts that approach the optimum of infinite staging.

The performance of a single-stage vehicle and an infinitely staged vehicle can be compared in terms of the ideal velocity increment (ΔV). For single-stage vehicles, the ideal velocity increment is:

$$\Delta V = g_c \; Isp \; \ln \frac{1}{1 - M_p/M_v}$$

where:

ΔV = ideal velocity increment (ft/sec)

g_c = a proportionality constant

Isp = specific impulse (seconds)

M_p = initial propellant mass (lbm)

M_v = initial total vehicle mass (lbm)

The derivation of this equation may be found in almost any text on rocket propulsion.

The term, $1 - M_p/M_v$, is commonly called the mass fraction of the vehicle. For infinite staging, the ideal velocity equation must be modified. Starting from the basic momentum equation:

$$(1) \qquad M_v \frac{dV}{dt} = \frac{dm}{dt} \; v_e$$

$$(2) \qquad M_v(t) = M_{pl} + M_i(t) + M_p(t)$$

$$(3) \qquad -\frac{dm}{dt} = \frac{dM_v}{dt} - \frac{dM_i}{dt}$$

and

$$(4) \qquad Isp = \frac{v_e}{g_c}$$

where: $M_v(t)$ = vehicle total mass at any time

$M_p(t)$ = vehicle propellant mass at any time

$M_i(t)$ = vehicle inert mass at any time

M_{pl} = payload mass (constant

v_e = exhaust gas velocity

m = exhaust gas mass

K = inert to propellant mass ratio (constant)

Substituting for dm/dt and v_e, the equation becomes:

$$dV = -g_c \, Isp \frac{dM_p}{M_{pl} + (1+K)\, M_p}$$

Integration between the initial and final limits gives:

$$\Delta V = \frac{-g_c \, Isp}{1 + K} \left[\ln \left(M_{pl}\right) + (1+K) M_p \right]_i^f$$

$$= \frac{g_c \, Isp}{1 + K} \ln \left[\frac{M_v}{M_{pl}} \right]$$

$$= \frac{g_c \, Isp}{1 + K} \ln \frac{1}{1-(1+K)\, M_p/M_v}$$

where M_p and M_v equal initial conditions as originally defined. By defining a new parameter, λ, as:

$$\lambda = M_p/M_v$$

the equations for single-stage (1) and infinitely staged (2) vehicles become:

$$\Delta V = g_c \; Isp \; \ln \frac{1}{1 - \lambda} \tag{1}$$

and

$$\Delta V = \frac{g_c \; Isp}{1+K} \; \ln \frac{1}{1-(1+K)\lambda} \tag{2}$$

The equations are plotted parametrically in Figure I-6. Figure I-7 shows the percent of change in an infinitely staged system over an unstaged system. If the single-stage vehicle is represented by $K = 0$, then these figures illustrate that vehicles with small propellant ratios (λ) are relatively insensitive to staging; but vehicles with large propellant ratios, in the range of 0.7 to 0.9, are quite responsive to infinite staging.

Shortcomings: In order to improve the velocity increment of a vehicle significantly by infinite staging, the system concept must operate at either high propellant ratios which implies a small payload weight or high values of K, which automatically places the vehicle in a low ΔV performance region. In addition, most of the so-called "infinite staging" concepts that have been studied have potentially very little increased performance. Usually, they suffer from reduced Isp or large energy losses.

Conclusions: Infinite staging is an illusive concept that has not been accomplished by any technique that really delivers a significant payoff in flight performance. There are various schemes for eliminating system inerts, but in general they are not true infinite staging concepts.

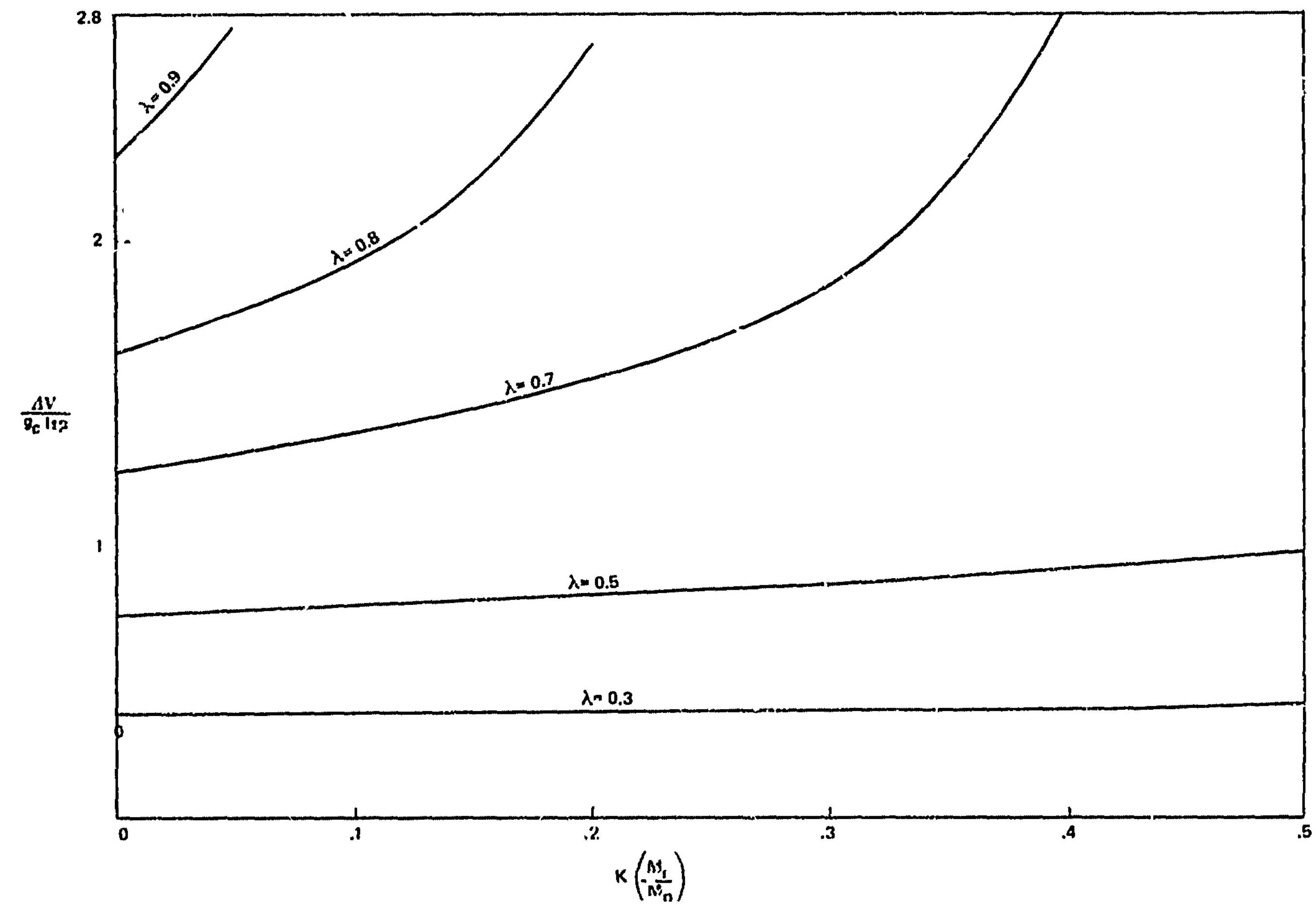

Figure 1-6. Infinite Staging Performance

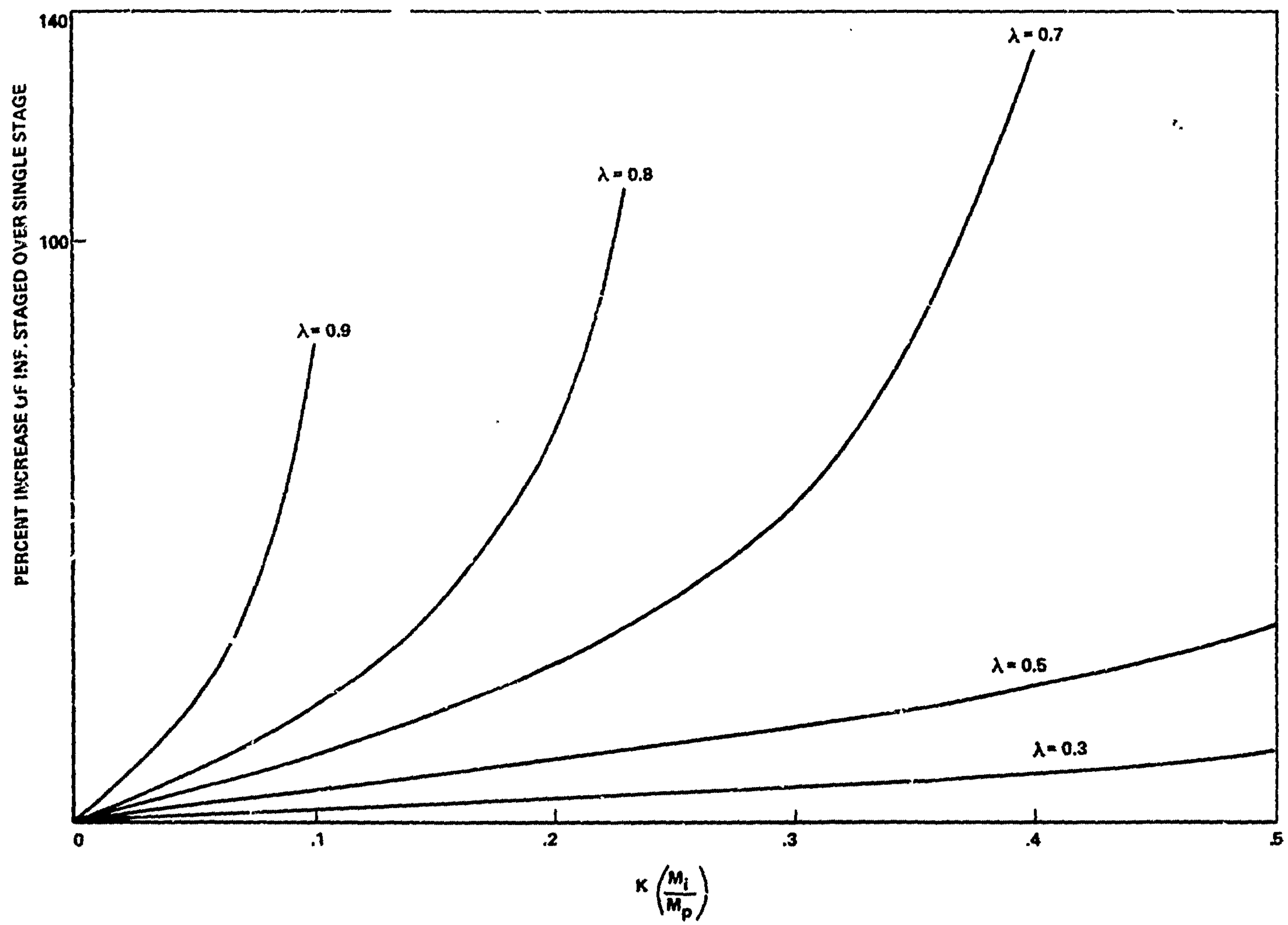

Figure I-7. Infinite Staging Performance Increase

Recommendations: The search for "advanced mass fraction" systems should be continued.

Title: Nozzleless Solid Rocket Motor

Concept: The nozzleless solid rocket motor utilizes a solid fuel grain geometrically contoured to function as a rocket engine nozzle during operation. Several nozzleless design variations are shown in Figure I-8.

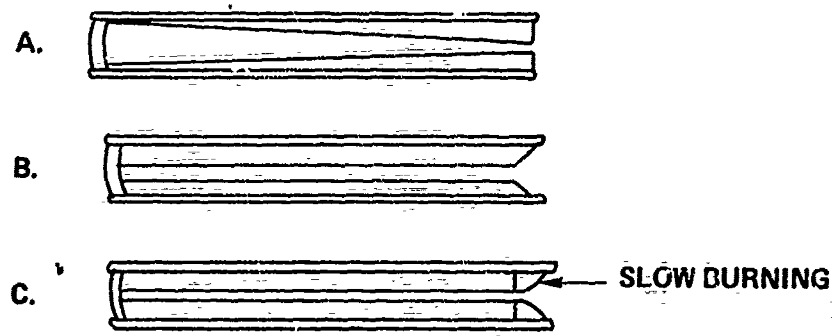

Figure I-8. Nozzleless Design Variations

Attributes: The nozzleless solid rocket motor is in reality an advanced mass fraction system. By avoiding the inherent weight penalty of a nozzle, the nozzleless motor potentially offers improved payload or terminal velocity capability. The nozzleless motor could also be simple, low cost and reliable; and, with the addition of a consumable case, it could approach the performance of an infinitely staged system.

Analysis: The nozzleless solid rocket motor concept dates back to the early 1960's when Grand Central Rocket Company (now Lockheed Propulsion Company) conducted a program with NASA support.* During this program, a ballistic model was developed for a nozzleless motor and four 7-inch motors were tested. Correlations between the firings and the model proved to be reasonably accurate. Lockheed concluded that nozzleless motors perform in a predictable manner, are capable of high performance in an appropriate configuration, and are

*Feasibility Study of a Nozzleless Rocket Motor, GCR Report F-0014-61.

cheaper than conventional solid motors because of reduced hardware and simplified construction.

A computer program, based on the Lockheed ballistic model, was used by the AFRPL to study nozzleless motor designs. This computer program accounts for the pressure drop down the grain length but is limited in its ability to simulate erosive burning. The studies indicated that nozzleless motors, by proper propellant selection and grain design, can achieve up to 90 percent of the total impulse delivered by conventional nozzled designs (on a fixed-volume basis). Cost savings were estimated at 20 to 30 percent, derived primarily from the elimination of the nozzle.

The attractiveness of nozzleless solid rocket motors is due not only to cost savings and simplification but their increased potential for flight missions. Using the terminal velocity or ΔV capability of a rocket as a measure of performance, nozzleless rockets may be compared to conventional rockets. The ideal ΔV equation is:

$$\Delta V = g_c \, Isp \ln \frac{1}{1 - \lambda(1 - M_p/M_v)}$$

where

ΔV = velocity increment

g_c = proportionality constant

Isp = specific impulse

λ = motor mass fraction (propellant to motor ratio)

M_p/M_v = payload to vehicle mass ratio

Assuming a typical air-to-ground missile with:

$$Isp = 250 \text{ seconds}, \lambda = 0.7, M_p/M_v = 0.3$$

as a baseline, the performance between a conventional system and a nozzleless system is illustrated in Table I-8.

TABLE I-8. RELATIVE PERFORMANCE CAPABILITY OF NOZZLELESS ROCKETS

Motor Design	Isp	λ	M_p/M_v	ΔV	% Change ΔV
Conventional	250	0.7	0.3	5400	0
Nozzleless	250	1.0	0.3	9675	79.2

These results assume that the total mass and payload mass are kept constant. In other words, the nozzle mass is replaced with an equal propellant mass and a negligible case weight.

The AFRPL studies were extended to the improved 2.75-inch rocket and the 40-millimeter rocket assisted projectile. Using constant-volume systems, the nozzleless versions were about 5 percent heavier when the nozzles were replaced with propellant. The total impulse and specific motor performance requirements for the nozzleless motors were 9 to 12 percent less than the conventional motors.

A total of 24 nozzleless firings have been made (18 within the past year) by the AFRPL and various companies. During the course of motor firings, a combustion instability problem occurred in seven firings. This problem has taken the form of severe and repeated characteristic chamber length (L*) extinguishment during motor operation. Most of these tests used 2.75-inch hardware and had simple cylindrical grains. Various conventional propellants have been used in these firings. By simply removing the nozzle from the conventional motor and firing it, about 70 percent of its standard total impulse is achieved. This technique does not provide a true comparison between motors with and without nozzles for several reasons. The most important reason is the propellant burn rate. Conventional motors are generally designed for operation in the pressure regime of 1000 to 2000 psi. Burn rates and nozzle sizes are matched appropriately. Port areas are usually much larger than the throat area in these motors to avoid erosive burning. When the nozzle is removed, the port becomes the throat, thus reducing the chamber pressure and propellant burn rate. The lower pressures result in lower total impulse. Decreasing the burn rate of the propellant without changing other motor parameters can thus have a significant effect upon the delivered total impulse. A motor designed for operation without a nozzle would use the volume originally taken up by the nozzle for additional propellant; would have a higher web fraction; would use a propellant having more desirable propellant properties (such as a low pressure exponent); and would probably have a lighter case. All these changes would greatly enhance the comparison between nozzleless and conventional motors in terms of performance.

Shortcomings: Considerable development is necessary to operate at low pressure and to avoid the instability problem. It is assumed that proper propellant tailoring and grain design could accomplish stable operation. In addition, the nozzleless motor designs thus far envisioned cannot accommodate the traditional fins necessary for flight stability.

Conclusions: Elimination of the nozzle from a conventional rocket motor will result in a 30 percent decrease in total impulse. By placing additional propellant in the volume originally occupied by the nozzle, utilizing improved grain design, and selecting proper propellants, the decrease in total impulse can be reduced to 10 percent or less. Preliminary studies indicate that nozzleless motors would cost 20 to 30 percent less than conventional motors, and the use of nozzleless rocket motors appears most attractive for small, unguided, air-launched solid rockets.

Recommendations: The analytical design capability for nozzleless rockets needs to be improved and coordinated with test data. A low level of effort is recommended to develop design capability and to study instability problems. If the results appear encouraging, the nozzleless concept should be demonstrated with several inventory-sized motors, and possible solutions to flight stability and guidance problems should be investigated.

Title: Caseless Rocket Motor (CRM)

Concept: The Caseless Rocket Motor is a vehicle/projectile propelled by the combustion of gases external to the vehicle/projectile.

Attributes:

1. Increased propellant mass fraction. This attribute could positively affect the following characteristics:

   a. Impulse-to-weight ratio
   b. Acceleration
   c. Time-to-target
   d. Warhead capability
   e. Range

2. Simplicity. This implies possible improvement in reliability, accuracy, and projectile cost.

3. Modestly increased specific impulse due to aerodynamic enhancement of thrust.

Description: Two basic afterbody configurations are being investigated: (1) an end-burner using a consumable case, and (2) a truncated cone burning on the side and end (Figures I-9 and I-10, respectively). Both configurations propel an ogive forebody which has nonreacting surfaces (nonburning).

Analysis: The conventional rocket motor provides propulsion through the momentum of the exhaust gases. The external burning rocket concept relies on an aerodynamic compression process as well as momentum to provide thrust. As the supersonic freestream approaches the afterbody,

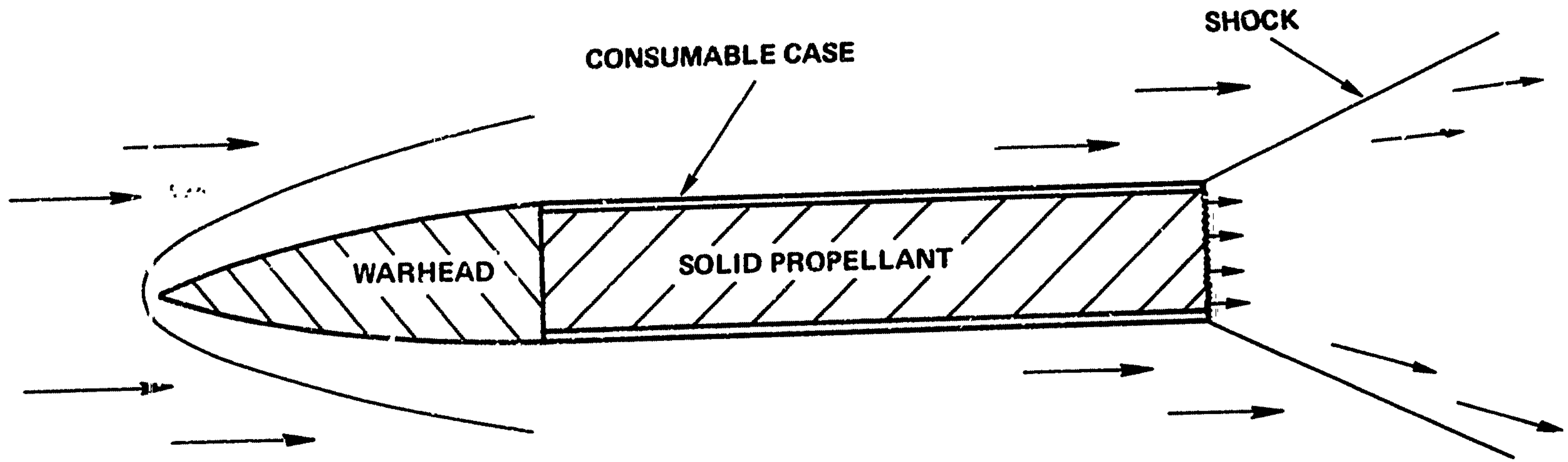

Figure I-9. Initial Performance Model (End-Burner)

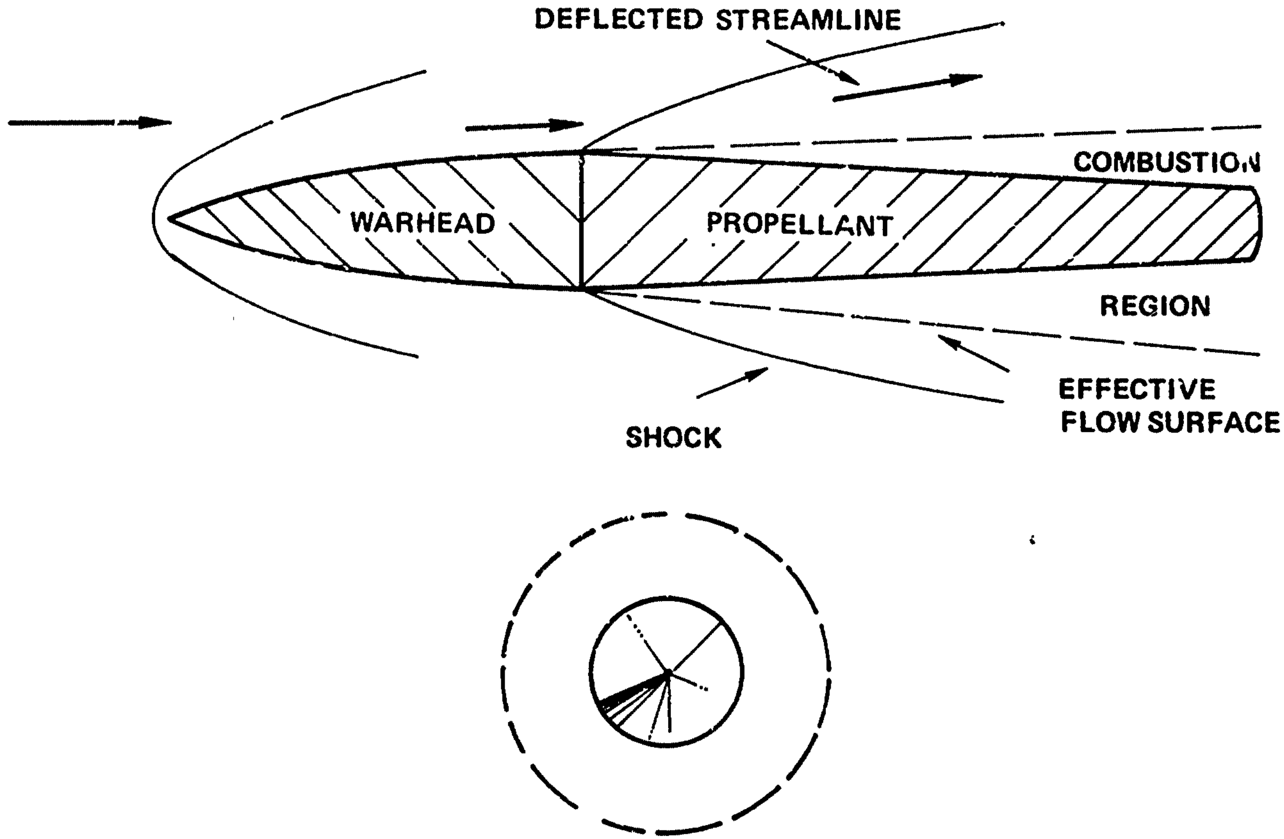

Figure I-10. Effect of Reacting Afterbody Surfaces the streamlines in close proximity to the projectile body see the combustion gases from the reacting surfaces. If these combustion gases blow off the reacting surface w . sufficient strength, the streamlines are then deflected as though they had encountered a geometric surface; thus, a compression region is generated. The flowfield aft of the compression shock structure is elevated above freestream static pressure. A force is generated normal to the surface by the momentum of the exiting propellant and by the integrated pressure-area term of the increased pressure field acting over the surface. This force is composed of two components. One component acts only as a compressive force on the propellant surface and is of no immediate aid in achieving thrust. The other component acts as a propelling force.

The optimum cone angle for the truncated cone afterbody is being examined at present. This implies an optimization on cone angle for maximum thrust component and on the integrated pressure-area term. The potential of fuel-rich propellants will also be examined.

It might appear obvious to maximize the propelling component and minimize the compressive component. This implies an aft cone angle of 90°, the case of the end-burner. However, for a given projectile diameter, a 90° cone angle minimizes the surface area over which the pressure may act. This limiting case was examined as the initial performance model and is illustrated in Figures I-9 and I-11. It is interesting to note that the truncated cone configuration will geometrically approach the end-burner configuration as the burning time progresses.

Shortcomings:

1. A large effort will be required to determine aerodynamic stability.

2. Combustion instability may be a problem.

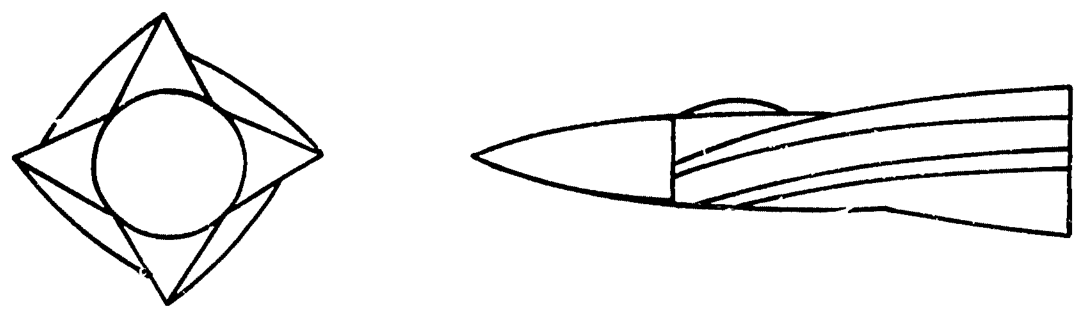

Figure I-11. Possible Application (Cannon Shell - End-Burner)

3. The mass-flow requirements of the end-burning configuration may be beyond the state of the art of propellant formulations.

4. Ignition and boost mechanisms could present problem areas.

5. Propellant structural integrity could be a problem.

Conclusions: Analysis of the pure end-burning caseless rocket motor indicates that acceptable specific impulse values and thrust levels are possible with solid propellants having sufficiently high burn rates. For typical propellants, a theoretical specific impulse of 180 seconds was calculated. This is somewhat lower than the specific impulse of a conventional solid propellant rocket motor; but for the same vehicle size and weight, more propellant can be utilized since the CRM functions without nozzle or combustion chamber. Analysis of the CRM boattail configuration indicated that the boattail has lower thrust and requires higher burning rates than the pure end-burning CRM. The only possible advantage of the boattail is better aerodynamic stability, but much experimental investigation would be necessary in that area. The principal disadvantage of the CRM configuration is that it requires very high propellant burn rates (on the order of 2 to 6 inches per second). Burning rates of this magnitude at the required low pressure are definitely far beyond the state of the art. The required burning rates coupled with the mediocre performance gains indicated by preliminary analysis make the CRM concept unattractive by present-day standards.

Recommendations: Considering the state of the art of the solid propellant field, the CRM appears to be unfeasible at the present time. In the event of a significant advance in mass evolution rates of solid propellants, it may be worthwhile to investigate the concept in more detail.

Related Concepts: Presently, various agencies are pursuing a related propulsion concept which is termed external burning (EB). Contrary to the CRM configuration, a combustion chamber is employed and a fuel-rich propellant is burned partially in this gas generator and then expelled through discrete nozzles to burn to completion in the inviscid supersonic flow adjacent to the wake of the vehicle. Rather than generating thrust principally by imparting momentum to the gases in the combustion chamber, the EB concept relies on entrapment and compression of the recirculation region of the wake by propellant gases burning outside but adjacent to this viscous region. Experimental investigations have proved that back pressure over free stream pressure ratios of 1.3 and Isp's of approximately 480 seconds are possible. Considering that normally (without combustion) P_B/P_∞ is approximately 0.5 and that a P_B/P_∞ of approximately 1.6 will allow sea level flight at Mach 2.3, it is obvious that the external burning concept holds considerable promise in the near future.

Title: Sustained Detonation

Concept: Rotating Detonation Wave Engine (RDWE)

Attributes:

1. Possible elimination of liquid rocket combustion instability problems.

2. Lower engine weight per unit thrust.

3. Compact engine component packaging capability.

Description: The RDWE consists of an annular combustion chamber in which propellants are injected through the injector at a constant and uniform rate. Combustion of the propellants is by a detonative mode rather than the deflagration mode used in conventional chemical rocket motors. Detonation waves completely fill the chamber cross section and propagate steadily in the same circumferential direction. Ideally, propellants can be injected in liquid or gaseous form; however, operation is simplified if gaseous propellants can be used.

Analysis: The occurrence of combustion instability has plagued liquid propellant rocket engine development programs since their very beginning. Because this phenomenon is nonlinear and analytically not well understood, it has not been possible to scale-up in thrust level from the successful operations of small-scale, laboratory combustors. Although not verified, several combustion investigators believe that detonation occurs in some form as a part of the instability process. The thought behind the RDWE is to force the occurrence of detonation deliberately in a controlled work manner; thus, so to speak, use the instability process to do useful work and eliminate its random, unexpected, destructive occurrence. In this way, it is believed that the scaling problem would be minimized.

Because of the nature of the detonation process, higher energy conversion rates per unit chamber volume can be expected with the RDWE concept than with conventional systems. Thus, it is conceivable that the overall chamber size and therefore, weight, could be reduced for a fixed-thrust application using the RDWE. This observation is only postulated and has yet to be verified. Actually, because of the high temperatures associated with the detonation waves, heat transfer rates in the chamber of the RDWE are about equivalent to those measured in the throat of the conventional rocket thrust chamber. For structural containment of the high pressure and thermal loads associated with the RDWE concept, a heavier structure may be required, thus possibly negating weight savings realized by the smaller chamber size. A complete system analysis would be required to determine if any real savings in weight can be obtained. Because of the current exploratory nature of the concept, such a study has not been conducted.

Some advanced engine concepts currently being pursued include an annular chamber configuration with compact packaging of conventional rocket engine components. Since the RDWE uses an annular chamber, it should exhibit the same packaging benefits.

During the period from June 1962 through January 1964, the University of Michigan conducted an AFRPL-sponsored program to investigate the feasibility of the RDWE (Ref. I-32). This program included several separate studies pertinent to the feasibility of a detonation device. These studies and reasons for them included:

1. Detonation in a two-phase medium. Steady propagation of a detonation wave in a gaseous-liquid droplet environment had never been attempted.

2. Detonation at low temperatures and high pressures for gaseous mixtures of hydrogen and oxygen. Some propellants of interest would be cryogenic, and hence, properties of detonation in the hydrogen-oxygen environment were important.

3. Heat transfer associated with the detonative process. High wall heat transfer rates were expected.

4. Detonation in curved partially confined channels utilizing premixed hydrogen and oxygen. Using this geometry a detonation wave traversed an annular combustion chamber attached to an annular exhaust nozzle.

5. Detonation in annular and linear motor configurations with separate gaseous fuel (hydrogen and methane) and gaseous oxidizer (oxygen) injection. Experimental results from an actual RDWE were essential.

6. A simplified analytical model of the idealized gas dynamics in the annular chamber of the RDWE. Approximate theoretical analysis of the gas dynamics of the RDWE was necessary to assess potential scaling factors and motor performance. From the University of Michigan studies, the following major conclusions were drawn:

a. The droplet shattering process is of extreme importance in stabilized two-phase detonations. It was indicated that detonation waves in a heterogeneous medium could be expected only if either the drops are very small (less than 10μ) or if convective velocities behind the shock can shatter the larger drops into very small drops in a very short period of time (on the order of 10μ sec).

b. Experimental detonation velocities at high pressures and low temperatures are more rapid than those predicted theoretically. This information is important in that it has a strong influence on the maximum pressures realizable in the engine.

c. Theoretical heat transfer to the wall of the RDWE is of the same order as it is at the throat of a small conventional rocket motor.

d. Detonation velocities in curved, partially confined channels suffer a degradation of about 7 percent compared to detonations in straight, confined tubes (using hydrogen and oxygen).

e. Although a sustained detonative process was not achieved in the experimental motors, it was concluded that nothing fundamental prevents this accomplishment. Great difficulty was experienced in getting the wave to move in only one direction, and success was never achieved in obtaining continuous rotation after the first cycle. The reasons for this were believed to be among the following:

(1) Severe attenuation due to the discrete injection pattern
(2) Insufficient mass flow rate into the motor
(3) Continuous burning after the passage of the first detonation wave

In support of this last reason, the investigators found some work by the Russian, Voitsekhovsky, in which he successfully obtained detonations in an annulus using premixed reactants.

f. As a result of the analytical studies, no significant change in the sea level Isp using the RDWE over that achieved with conventional rockets is expected.

No additior " · on the RDWE has been conducted since the expiration of t University of Michigan program.

Shortcomings: The RDWE concept is essentially unverified and still in the early exploratory stage. Critical technology needs involve programs to:

1. Define the feasibility and limits of propagating detonation waves in a liquid/gas and liquid/liquid media.

2. Demonstrate that the RDWE motor concept is capable of sustaining repeated detonation waves in a controllable and predictable manner in an annular combustion chamber containing the types of environmental media discussed previously.

Conclusions:

1. The RDWE concept is feasible but would probably require 10 years to develop.

2. The concept offers little benefit over currently available propulsion and should not be pursued. Although cases of combustion instability are still not well understood and prediction techniques need to be improved, there are ways to handle the problem which were not available in 1962-1964, during the era of the RDWE.

Title: Detonation Propulsion (Solid or Liquid)

Concept: Detonation propulsion uses the energy released from sequential explosions exterior to the vehicle for thrust.

Attributes: Some of the potential advantages for detonation propulsion are: a 50 percent increase over conventional propulsion systems in delivered specific impulse; the development of an infinite staging solid propulsion system; built-in altitude compensation; optimum performance capability in a high-pressure environment; and a significant cost reduction for the propulsion system.

Description: The detonation propulsion concept is based on using the detonation of a thin layer of explosive to propel a payload. A solid explosive design could utilize a system of alternating explosive wafers with wafers of attenuating material. A liquid explosive concept could utilize a system which ejects a thin film of explosive over the thrust plate or base material which supports the payload. Sequential detonations would then propel the payload.

Analysis: The use of controlled explosions to propel objects has been explored in many fields. Fragmentation bombs, pile drivers, mining core drills, and squib-actuated cable cutters are examples of current applications using the detonation propulsion principle to a limited extent. The next step is to extend the use of this technique to the transport of payload systems into space.

It has long been known that chemical rocket propulsion could be used to deliver substantially increased specific impulse if expansion ratios on the order of 100,000 to 1 or greater could be achieved. To date, no practical means of attaining these ratios have been devised. However, it has been postulated that the use of common explosive [illegible] propulsion might be a method of obtaining very high pressure differentials and, in turn, increasing the delivered specific impulse. The pressure differentials will pose no problem since pressures of 3 to 7.5 million psi are common to conventional explosives. The problem then is to obtain the performance increase and devise a practical means of isolating the payload from the impulsive loading of the detonation shock wave. Preliminary analytical and experimental studies have been conducted to test the feasibility of the performance increase. No firm conclusions have resulted from the preliminary work because there was a wide divergence in the data obtained.

In late 1969 and early 1970, Aerojet General Corporation measured the velocity of a steel projectile that had been propelled by the explosive EL-506-C (Detasheet). Although the data exhibited considerable scatter, specific impulses as high as 350 seconds were calculated. Additional computer analyses using a one-dimensional hydrodynamic code and low explosive-to-projectile weight ratios indicated obtainable specific impulses up to 360 seconds (Ref. I-33).

Physics International Corporation, under contract to the AFRPL, conducted an analytical study of the detonation propulsion principle using both conventional solid and liquid explosives. Using a one-dimensional computer code and low explosive-to-projectile weight ratios, Physics International predicted specific impulses in the low 200-second range. They also conducted an independent experiment in which steel blocks were bonded to one side of an aluminum sheet with Detasheet borded to the other side. The explosive was detonated and the resultant velocities of the blocks measured. The computer calculations and the experimental measurements agreed to within 5 percent (Ref. I-34).

The AFRPL conducted similar experiments using Detasheet to propel steel blocks horizontally. The data exhibited low specific impulses ranging from 150 to 200 seconds (Ref. I-35).

The reason for the wide divergence in the Aerojet and Physics International results is unknown; however, one major difference exists in the computer codes used to calculate the projectile velocities (which are directly proportional to specific impulse): Aerojet used kinetic energy and Physics International used momentum. Further uncertainty is added by considering experimental data generated by Aerojet in an AFRPL-sponsored study of the detonability of solid rocket propellants. One experiment was designed to determine the detonation initiation threshold as a function of impulse imparted to RDX-adulterated propellant samples. Aluminum plates of various weights were propelled toward the sample by detonation of Detasheet. The explosive was partially confined by a steel back plate. The velocities of the aluminum plates were measured and specific impulses from 150 to 4[illegible] seconds were calculated for explosive-to-plate weight ratios of 1.4 to 0.18, respectively.

Even if the performance of explosive propulsion is found to be less than that of conventional systems, situations exist where overall system performance improvement can be achieved or where such a propulsion system may be the only system that could perform a specific mission. One such example is the use of "infinite staging" where only the payload remains after the propellant is expended. In a system where the payload-to-gross weight fraction is 0.42, an infinitely staged system with a specific impulse of 130 seconds will match performance with a solid rocket motor with a specific impulse of 240 seconds.

A mission with a requirement for the propulsion system to exhaust into a high-pressure environment would be a potential application for explosive propulsion. For example, it is unlikely that conventional rocket propulsion can perform a retro-fire function for a Venus soft landing where atmospheric pressures may be very high. Possibly explosive propulsion could perform the task.

Assuming that the specific impulse delivered by detonation is greater than that which conventional rockets deliver at sea level, the built-in altitude compensation due to the high pressures of the detonation becomes important. This advantage would lead to performance throughout the entire atmosphere that would approach that of a vacuum.

Shortcomings: Some potential disadvantages or problem areas associated with detonation propulsion are: (1) no increase in delivered specific impulse has been observed, (2) unconventional chemical rocket techniques have not been developed for initiation of liquid or solid explosives on rockets, (3) for liquid detonation rockets, the injection system is extremly complex, (4) adequate means for attenuation of the detonation shock to the spacecraft structure and payload have not been developed, and (5) there are stringent design requirements on the guidance and control system for a detonation propulsion system.

Conclusions: The concept of propelling a payload by sequential detonations of an explosive has not been adequately investigated. The limited activity in this area has concentrated on establishing performance data and the results have not been conclusive.

Recommendations: Further effort is required to resolve the performance question. Parallel investigations should be undertaken to understand the mechanism of impulse transfer, the optimum geometry of the explosive/ base material interface (todate only a flat plate has been considered), the most efficient type of initiation, infinite staging techniques, and candidate base and attenuator materials. If demonstrated performance continues to be lower than conventional rockets, the effort should be discontinued.

Title: Water Launcher (Ref. I-37)

Concept: The water launcher system uses buoyancy forces as a means of imparting an initial velocity to a rocket vehicle.

Description: The concept envisions a long water-evacuated tube submerged vertically in a body of water. At the bottom of the tube in contact with the water is a piston platform on which is placed the rocket. At launch time, the piston is released, and the rocket and piston are accelerated upward by the pressure forces on the bottom of the piston. At the surface or point of maximum velocity, the rocket would be ignited and complete its mission.

Analysis: Two different variations of the water launcher concept will be considered. The first is a "constant length tube," into which water flows at the bottom only. Analysis of this approach indicates a limited payoff due to the accumulated large mass behind the piston of water which must be accelerated. As an example, if a 1000-foot-long tube, 90 inches in diameter were used with a 75,000-pound missile, the maximum velocity obtained would be slightly under 170 ft/sec. Also, this velocity is reached long before nearing the surface. The constant-length tube is illustrated in Figures I-12 and I-13.

The second variation is a "variable-length tube." This approach eliminates the necessity of accelerating a large mass of water behind the piston. Accordingly, some technique must be used to allow water to enter the tube at a point directly behind the piston to maximize acceleration. This might be accomplished by staging the tube (dropping off portions of the tube), allowing the piston to trip gates that would admit water after the piston passes, or consuming the tube in some other fashion. Using a solid propellant ICBM with an ideal ΔV of 28,500 ft/sec, the variable-length tube 90 inches in diameter accelerates the 75,000-pound vehicle

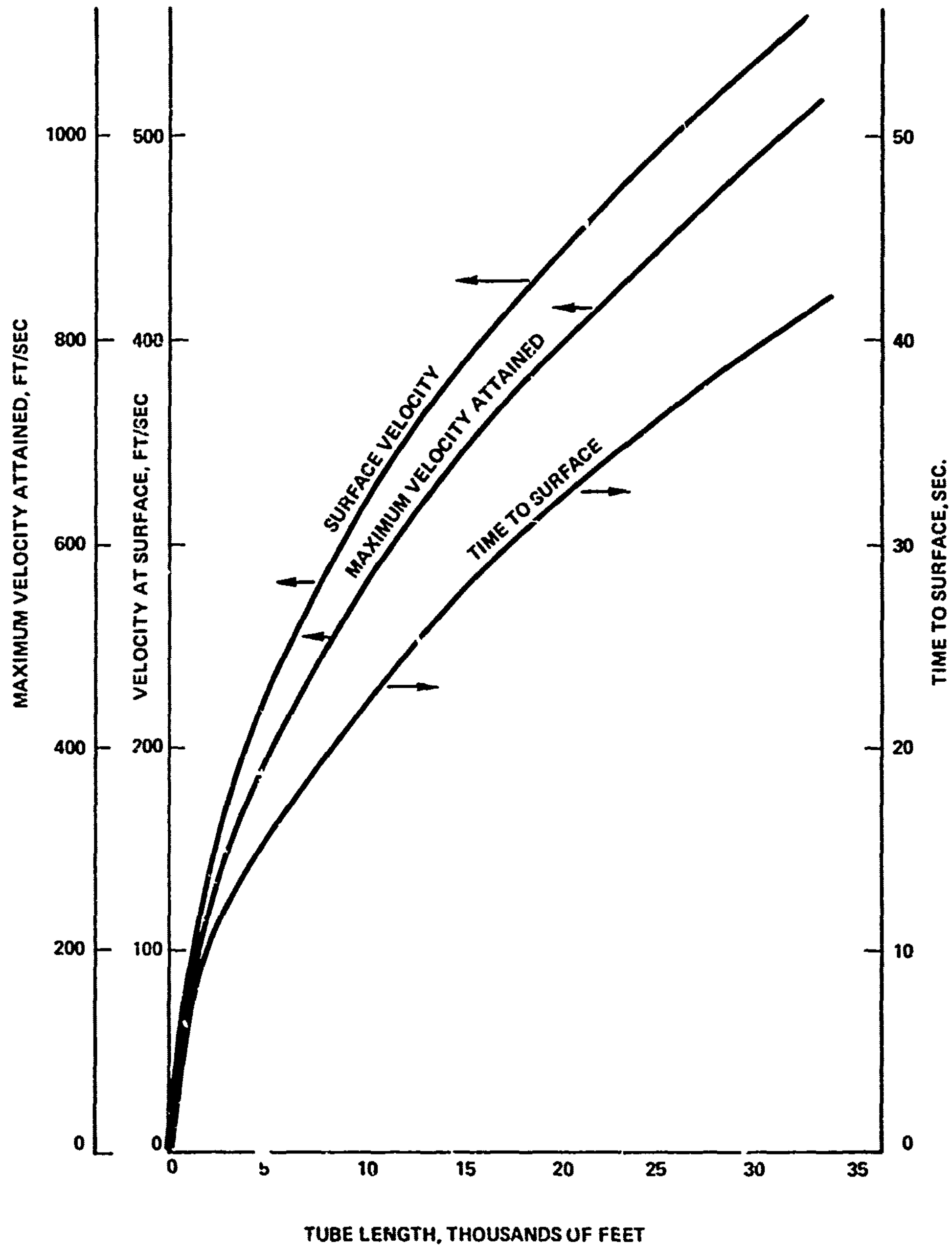

Figure I-12. Maximum Values For Water-Launched Rocket

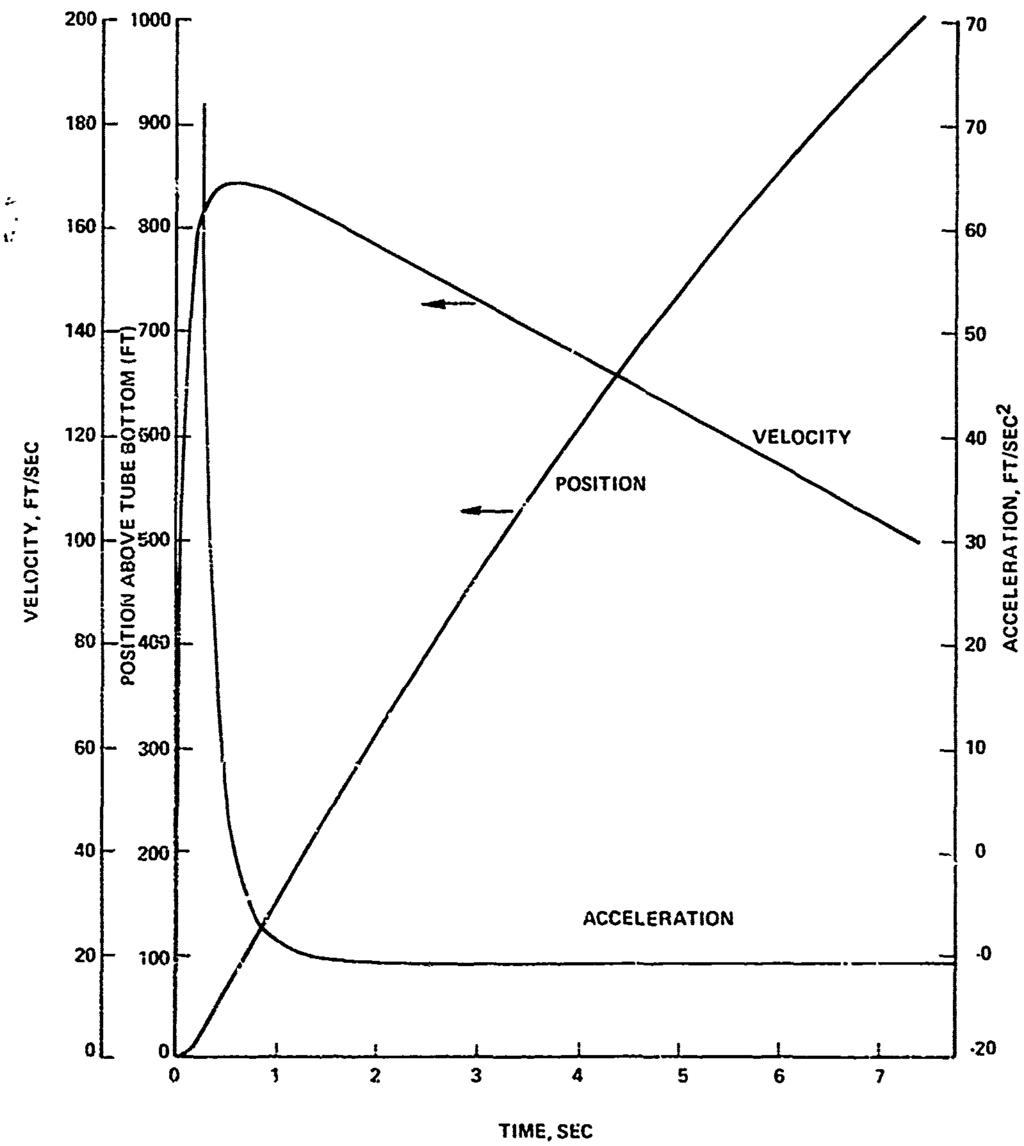

Figure I-13. Water-Launched Vehicle Dynamics from an initial depth of 1000 feet to a velocity of about 1101 ft/sec at the water surface. This initial velocity provides a gain in ΔV capability of 3.85 percent. This roughly translates into a 5.8 percent gain in payload or an 11.5 percent gain in range. The obtainable velocities for other initial launch depths are shown below in Table I-9.

TABLE I-9. OBTAINABLE VELOCITIES

H (feet)	V (ft/sec)
100	110
500	551
1000	1101
1500	1651

Note that in the above example the gain in ΔV capability is about 4 percent of the total ΔV assumed for the hypothetical missile. The ΔV capabilities of various stages on the three-stage ICBM may range from 25 to 45 percent of the total system ΔV. Thus, water launching does not appear as a means for elimination of a stage. Also, neither drag nor buoyancy forces on the tube were considered in the analysis.

Shortcomings: It has been shown that only a "variable length tube [illegible]ives ΔV gains of sufficient magnitude to warrant any development e[illegible] Tube construction to withstand the pressures at 1000-foot depths and the tendency of the tube to float as a result of buoyancy forces require some thought. Perhaps the worst design problem is developing the technique that allows the influx of water behind the piston to approach the ideal situation. In addition, because of high accelerations, the missile system may have to be structurally reinforced for use with the water launcher.

Conclusions: This concept appears feasible and could probably be developed within 5 to 10 years. Because no specific design was envisioned, development and construction costs are unknown at this time.

Recommendations: This idea should be given to some other laboratory or agency.

CHAPTER I-4. SUMMARY FOR PART I

The various chemicals and propellant combinations studied in the previous chapters serve to illustrate that there are indeed limitations imposed by the available chemical energy. It appears that an ideal specific impulse somewhere between 700 to 750 seconds may be the maximum limit of chemical energy. However, the conventional use of a nozzle to accelerate gases automatically imposes a 20 to 25 percent degradation in this specific impulse limit. To avoid this performance loss and approach the ideal specific impulse, chemical rockets must operate at pressure ratios above 10^5.

The development of ultra-energy compounds for propellants does not appear to be a rewarding endeavor within the current capability of chemical synthesis. Past experience has shown that storage of energy for release by methods other than the breaking and combining of chemical bonds will be very difficult. The development of metastable species does not appear promising. If ultra energy compounds can be synthesized in sizeable quantities, the work could take 20 or more years. Some work for more near-term use might be considered using ozone.

The study of new system approaches appears to be the most rewarding area in which advances to conventional chemical rockets may be achieved. Various studies indicate that small changes in propulsion system parameters such as mass fractions and Isp may provide significantly increased mission capabilities. In fact, even at the expense of Isp, increased mission performance may be obtained by radically modifying system concepts and configurations (i.e., infinite staging, nozzleless and caseless rockets, and detonation propulsion). Although the system concepts that were studied illustrated that significant advances might be obtained, none of the ideas reported here showed outstanding potential in their present form.

To summarize chemical rocketry, it is not anticipated that the desired quantum increases in performance or mission capability will be achieved by exploration of new high-energy chemicals or the design of new chemical rockets along the lines of conventional technology. Radically new approaches for the use of chemical energy are required to provide revolutionary advancements in rocketry.

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PART II

NONCHEMICAL PROPULSION

CHAPTER II-1. THERMAL PROPULSION

The concepts included under thermal propulsion generally heat a working fluid for thermodynamic expansion through a nozzle*. Although variations of several concepts such as the pulsed laser fusion engine do not fall within the strict definition of a thermal propulsion system, it was judged that these concepts had more in common with thermal systems than field, photon or unique systems. Thermal systems are in general, an extension of the principles applied to the utilization of chemical energy in conventional rocketry.

*Traditional chemical production of thermal energy was treated in Part I of this report. This chapter treats of other thermal energy sources.

Title: Electrothermal Propulsion

Concept: Electrothermal propulsion comprises those techniques whereby a propellant gas is heated electrically and then thermodynamically expanded through a nozzle to produce thrust.

Attributes: When compared to other thermal propulsion systems, electrothermal thrusters offer the advantages of high specific impulse; simple, reliable design; low cost through the use of inexpensive, non-strategic materials; life and restart capability; and selection from a large number of propellants.

In addition, the advantages of electrothermal thrusters over competitive electric propulsion systems include low power-to-thrust ratios and the capability to use a variety of propellants including residuals from other large thrust engines.

Description: Electrothermal propulsion encompasses all techniques whereby a propellant gas is heated eletrically and then thermodynamically expanded through a nozzle to produce thrust. Physically, there are many substantially different electrical means for heating the propellant flow. The techniques used most often include resistance elements, arc discharges, electrodeless discharges, and high-frequency excitation (Ref. II-1).

Only three systems warrant consideration as serious propulsion concepts: (1) the thermal decomposition resistojet, (2) the high temperature resistojet, and (3) the thermal arcjet. The first two systems are fundamentally very similar devices. They transfer the electric power to the propellant through a solid resistance element. The thermal decomposition resistojet (Figure II-1, page II-4) utilizes either an external or an internal resistance element to heat hydrazine fuel to the point of exothermic decomposition. The high-temperature resistojet (Figure II-2) by continued heat input creates temperature in the range of 3000° to $4000^{\circ}F$. To minimize the heat transferred to the chamber walls, the high temperature resistojet generally uses an internal resistance element.

The arcjet is an electrothermal device that produces a propellant flow whose interior gas temperature is higher than the melting temperature of the chamber walls. This is achieved by passing an electric arc through the propellant, leaving a cooler layer near the chamber wall (Ref. II-1). The highest performance has been demonstrated by the "coaxial flow arcjets" in which the propellant and arc both pass through a narrow constrictor passage (Figure II-3). This design produces the highest gas temperatures and the lowest heat losses (Ref. II-2).

Analysis: Historically, the first electric propulsion concepts to be developed were electrothermal thrusters. These devices received much emphasis during the early 1960's. These systems, including the resistojet and the arcjet, were directed toward kilowatt and megawatt space systems application. When it was realized that these large power supplies would not be available, electrothermal propulsion was deemphasized. The development of the Shell 405 catalyst in 1963, permitting the design of catalytic hydrazine thrusters for space propulsion, ended much of the electrothermal research.

The most serious departure of electrothermal thrusters from the ideal models used to describe them arises from the strong temperature dependence of propellant specific heat and the inability of these gases to maintain internal energy equilibrium during the rapid expansion through the nozzle ("frozen flow" losses). This problem not only influences thruster design but plays a major role in propellant choice. The major problem of electrothermal thruster development has been the reduction of such "frozen flow" losses as vibration, dissociation and ionization (Ref. II-1).

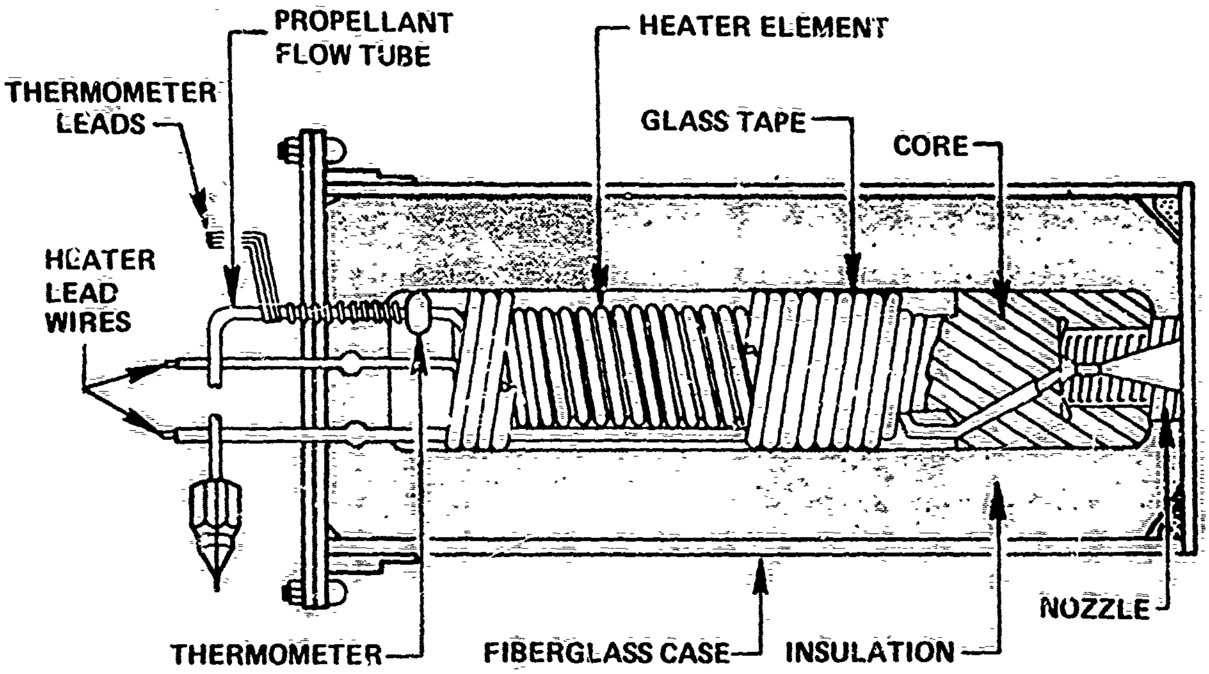

Figure II-1. Thermal Decomposition Resistojet (Ref. II-6)

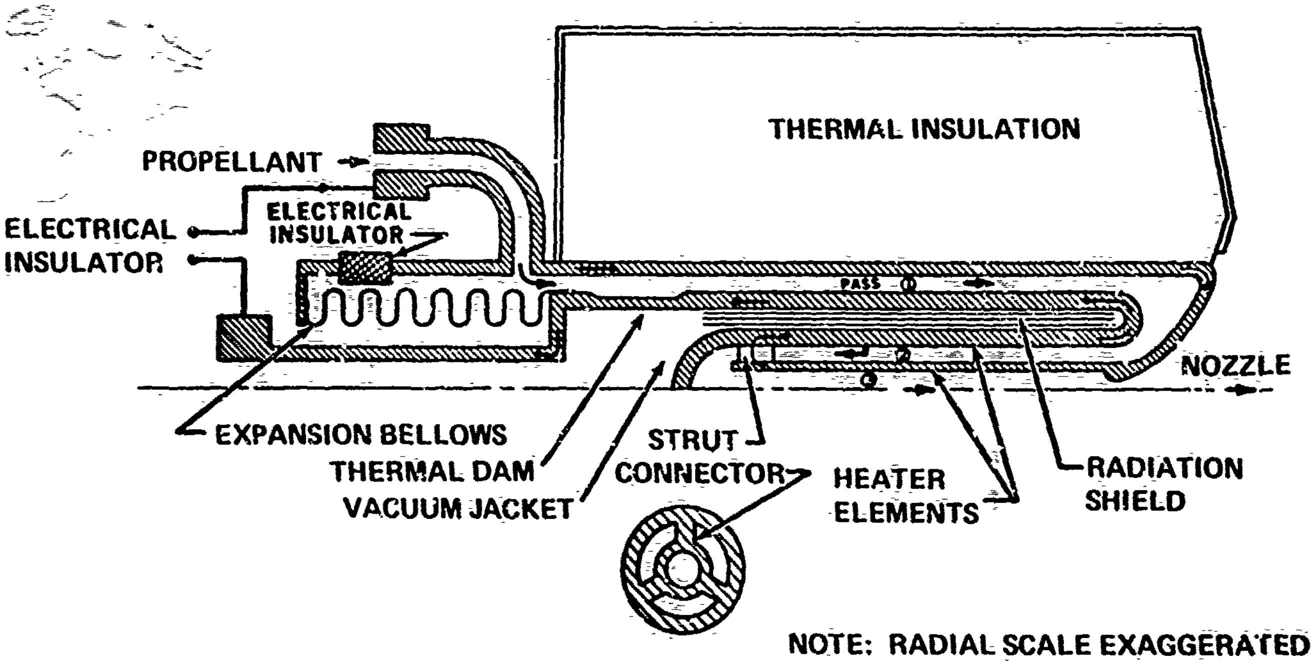

Figure II-2. Evacuated Concentric Tubes Resistojet Concept (Ref. II-7)

The final selection of an arcjet propellant is based primarily on frozen flow and specific impulse considerations. However, other considerations such as power requirement, propellant 'ensity and space storability may modify propellant selection, especially for resistojets.

Although hydrogen is attractive because of its high theoretical specific impulse and helium for its low frozen flow losses, these propellants present severe space storage problems and l--k high density (see Figure II-4). Ammonia is attractive from a space storability standpoint, and its theoretical specific impulse is not far below hydrogen. Although hydrazine (N_2H_4) has a theoretical specific impulse slightly lower than ammonia, its exothermic decomposition gives it a significant power advantage over other propellants. An exhaust temperature of 2000^oF (Isp of 245 seconds) is essentially "free" with hydrazine. Hydrazine's most important advantage is commonality with propellants used in other high-thrust propulsion systems aboard satellites. In addition to a tankage weight saving , any propellant not spent for orbital maneuvers can be used to extend the lifetime of the attitude control and stationkeeping electrothermal propulsion systems. Another a'vantage of hydrazine systems is that the clean exhaust mixture of ammonia, hydrogen, and nitrogen eliminates contamination of spacecraft surfaces (Ref. II-3).

Thermal decomposition resistojets in the one to one-hundred millipound thrust range require less than five watts of power to maintain wall temperatures of 1100^oF (Ref. II-4). This temperature is sufficient to initiate and maintain the exothermic decomposition of hydrazine in the combustion chamber at a power cost well within the budget generally available for satellite propulsion. The external resistance element design is the most reliable, and lifetimes of over one million pulses have been demonstrated (Ref. II-4). A quick pulse response time and low thrust level produce minimum impulse bits on the order of 10^{-3} lbf-seconds. This makes electrothermal thrusters ideal candidates for attitude control systems.

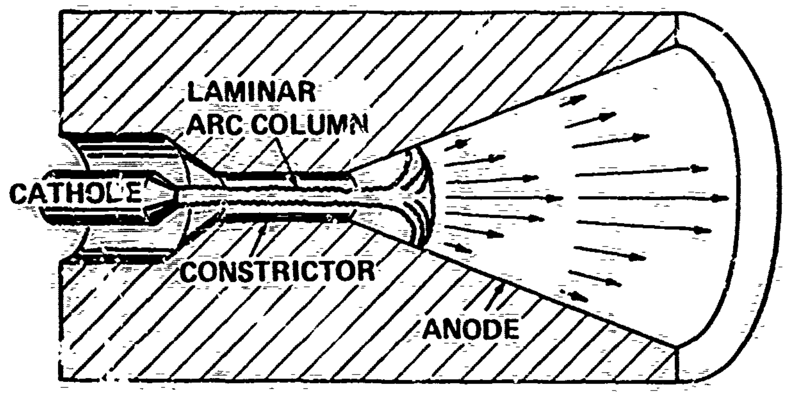

Figure II-3. Core-flow Pattern in a Constricted Arcjet. (See Ref II-1)

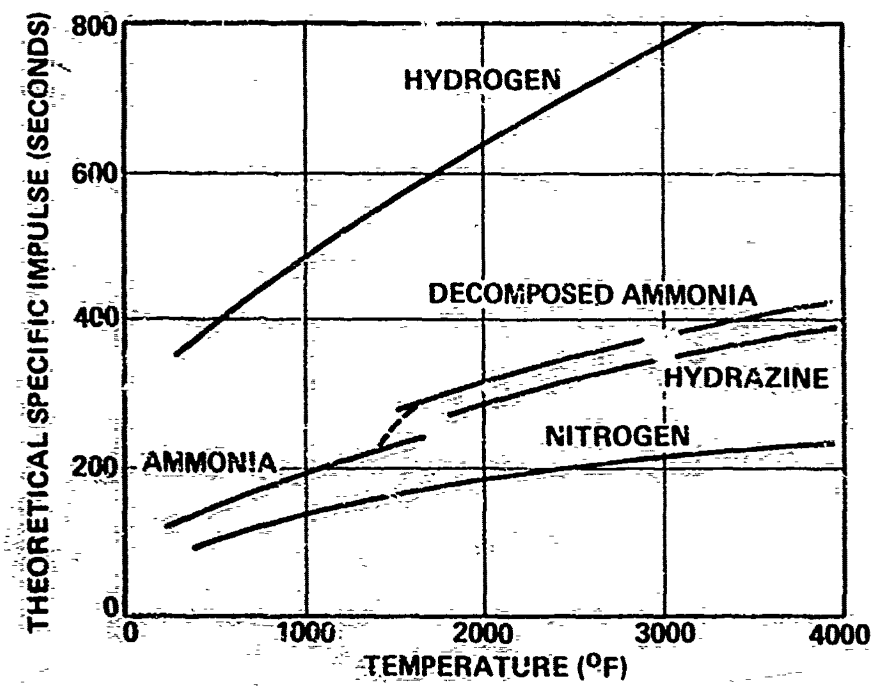

Figure II-4. Theoretical Specific Impulse Versus Temperature for Candidate Propellants (See Ref II-4)

To extend resistojet performance into the high-temperature range, the external resistance element must be abandoned in favor of an internal heater. This design attempts to minimize the problems associated with heat transfer from the resistance element to the gas stream, radiation losses from the thruster, and heat transfer to the chamber walls. However, the internal heater design subjects the resistance element to a severe aero-thermal environment which is a serious problem affecting thruster life. Hydrazine-fueled, high-temperature resistojets offer a significantly higher specific impulse (350 to 400 seconds) at moderate power levels. In the one to one-hundred millipound thrust range, power-to-thrust ratios below 5 watts/millipound are attainable and consistent with power budgets available for near-term satellite auxiliary propulsion. In this high-temperature (3000 to 4000°F) regime, hydrazine propellants offer the same advantages previously mentioned. As Table-II-1 demonstrates, ammonia resistojets with marginally higher specific impulse require two and one-half times as much power as comparable hydrazine thrusters.

The major life-limiting factors of hydrazine resistojets appear to be materials and propellant purity. The injector and combustion chamber must be constructed of materials which resist the severe nitriding environment and maintain structural integrity for the mission duration. Any non-volatile components in the propellant (primarily chlorides of sodium, calcium, potassium or iron) may be deposited on metal surfaces of the engine. Although the combustion chamber can withstand fairly large accumulations, non-volatile deposits can cause injector blockage and reduced performance. Carbon-containing compounds in the propellant cause similar results. Because of carbon contamination, high-purity hydrazine must be used. If carbon contamination problems can be eliminated, a range of higher specific impulse propellant blends containing carbon will be available for use in resistojet thrusters. Such propellants as unsymmetrical dimethylhydrazine (UDMH) and monomethylhydrazine (MMH) could be used in common with other high-thrust bipropellant engines.

TABLE II-1. ELECTROTHERMAL PROPULSION SYSTEM PERFORMANCE

System	Exhaust Temp (^{o}F)	Thrust (mlbf)	Power/Thrust (watt/mlbf)	Isp (seconds)
Thermal Decomposition Resistojet (Ref. II-4)	2000	1-100	0.25	245
Ammonia Resistojet (Ref. II-4)	2000	1-100	7.5	255
Ammonia Resistojet (Ref. II-4)	4000	1-100	12.5	375
High-Temperature (Ref. II-4) (N_2H_4) Resistojet	4000	1-100	5.0	400
Thermal Arcjet (H_2) (Ref. II-5)	10,000 to 15,000	100	10.0	1100
Thermal Arcjet (NH_3) (Ref. II-5)	10,000 to 15,000	531	56.5	1012

A traditional candidate for future high-thrust missions is the thermal arcjet. Its advantage over other electric systems is its relatively low power-to-weight ratio (Table II-1). Although specific impulse and frozen flow loss mechanisms play a significant role in arcjet propellant selection, exothermic decomposition fuels such as hydrazine still offer significant power advantages. The thermal arcjet converts available electrical energy to heat and then attempts to convert this random energy into directed kinetic energy. Thermodynamically, this is not a very attractive sequence due to the inability of many of the slower internal energy processes (such as dissociation, ionization and vibration) to reach their equilibrium levels during the rapid expansion in the nozzle. These frozen flow losses are especially severe in the thermal arcjet and result in significant performance limitations. For this reason, the most attractive specific impulse range of the thermal arcjet lies between 1000 and 2000 seconds using hydrogen as the propellant. The corresponding gas temperatures range from 10,000 to 15,000^{o}F above which dissociation and ionization losses become unbearable. Though the arcjet experiences electrode life problems, several thousand hours life and several thousand starts have been demonstrated. Electromagnetic noise emitted by the arc can be a major problem affecting on-board electronics and disturbing communications with the vehicle. Although many questions remain to be answered, the thermal arcjet does not appear to be a competitive propulsion system. With its low power requirement and wide range of usable propellants, it could be very attractive.

Shortcomings: Four basic shortcomings affect the applicability of resistojet thrusters to space missions:

1. Their maximum specific impulse (excluding hydrogen fuel) is not significantly higher than 500 seconds. This limits their application to low total impulse applications such as attitude control and stationkeeping.

2. Although they require only moderate power in the one to one-hundred millipound thrust range, scaling to thrust levels in excess of one pound required power levels far in excess of that normally allocated for propulsion.

3. Although material compatibility is not a current problem for hydrazine resistojets, it may affect the attainment of lifetimes in excess of one million pulses.

4. Carbon and nonvolatile propellant constituents may contaminate engine components and decrease lifetime. Unless a technical breakthrough in engine design occurs to alleviate this problem, significantly longer lifetimes may only be gained through the use of extremely pure hydrazine fuel.

The thermal arcjet suffers three basic shortcomings:

1. The arcjet's application is constrained by the severe frozen flow losses encountered at specific impulses above 1000 seconds using fuels other than hydrogen. These losses generally limit the arcjet to a narrow performance range.

2. Electrode degradation is a severe life-limiting factor.

3. Electromagnetic noise emitted by the arc may affect on-board electronics and communications.

Conclusions: Thermal decomposition resistojets offer significant life, reliability and pointing accuracy in the 1 to 100 millipound range. The high-temperature resistojet offers a higher specific impulse. Although both resistojets have moderate power requirements, they are of interest only for near-term attitude control and stationkeeping applications.

The thermal arcjet is of interest only in a narrow performance range. Only a major breakthrough in engine design or propellant technology that significantly reduces the frozen flow losses could extend this performance range. A more promising alternative is the development of the electromagnetic arcjet.

Recommendations: Both the thermal-decomposition resistojet and the high-temperature resistojet appear very attractive for near-term, low total impulse missions. Current development efforts in these areas should be monitored with periodic evaluation as propulsion candidates for appropriate missions. The most significant questions remaining unanswered pertain to lifetime limitations resulting from nitriding and engine contamination by nonvolatile propellant constituents. However, these problems do not need solutions to meet present requirements. Additional work should be done to reduce carbon contamination problems which now eliminate UDMH and MMH from consideration as propellants. Barring major technical breakthroughs, thermal arcjets do not appear to be promising area for future developments.

Title: Nuclear Propulsion (Fission)

Concept: The heat released in the chain reaction of a nuclear reactor is used to heat a working fluid, which is exhausted through a nozzle to produce thrust.

Attributes: The potential performance of nuclear propulsion concepts indicates that they may be applied to a wide range of missions. Furthermore, system and mission studies which have been conducted indicate that nuclear engines will outperform chemical engines over a wide range of operating conditions. They are especially attractive where high ΔV, large spacecraft and multiple use are required. As the developments of nuclear rocket engines progress, higher performance will be available to mission planners. This fact is due to potentially high specific impulse at thrust levels between 10,000 and 1,000,000 pounds, and flexibility in engine design. The attributes of each class of engine over those of lower performance are:

1. Solid Core

   High Isp
   5 to 10 year mission life
   High controllability
   Storability

2. Particle Bed

   High Isp
   High thrust-to-weight
   Simplicity
   Growth potential
   Core can be re-supplied
   Core can be dumped overboard

3. Gaseous Core

   High Isp
   Total fuel containment
   Core can be re-supplied

Description: This report will consider only those systems in which the thermal energy of the reactor is used to heat a working fluid which is exhausted through a nozzle to produce thrust. There are several ways of accomplishing this, each with its unique advantages and disadvantages. The evolution of nuclear rockets, in terms of specific impulse, suggests the following as the most promising concepts.

1. Solid core nuclear rocket engines: In engines of this class, the reactor is comprised of a solid bed of uranium-rich fuel material surrounded by a reflector for neutron conservation. The propellant is pumped around the engine for cooling, and through the reactor core which has flow passages designed to transfer the maximum amount of heat to the propellant. After exiting the core, the propellant exhausts through a nozzle. This type of engine has been extensively developed in the NERVA program, using gaseous hydrogen as a propellant, and in the PLUTO program, using air as a propellant (Figure II-5).

2. Particle bed nuclear rocket engines: In order to increase the performance of a nuclear engine over that of the solid core device, it has been suggested that the reactor core be constructed of small fuel pebbles which offer a greater heat transfer surface to the propellant and suffer less severe thermal stress loads. The pebble bed is fluidized by the flow of propellant through it. To counteract the force exerted on the pebbles by the flow and prevent them from leaving the core, the bed itself is rotated so that the centrifugal forces on the pebbles counterbalanace the drag forces. Such a pebble bed is said to be fluidized (see Figure II-6).

Two reactor concepts are currently under consideration. The Air Force Aerospace Research Laboratory is working on a rotating bed composed of colloid or micron-sized particles suspended by the vortex flow of the propellant. Brookhaven National Laboratory is examining a

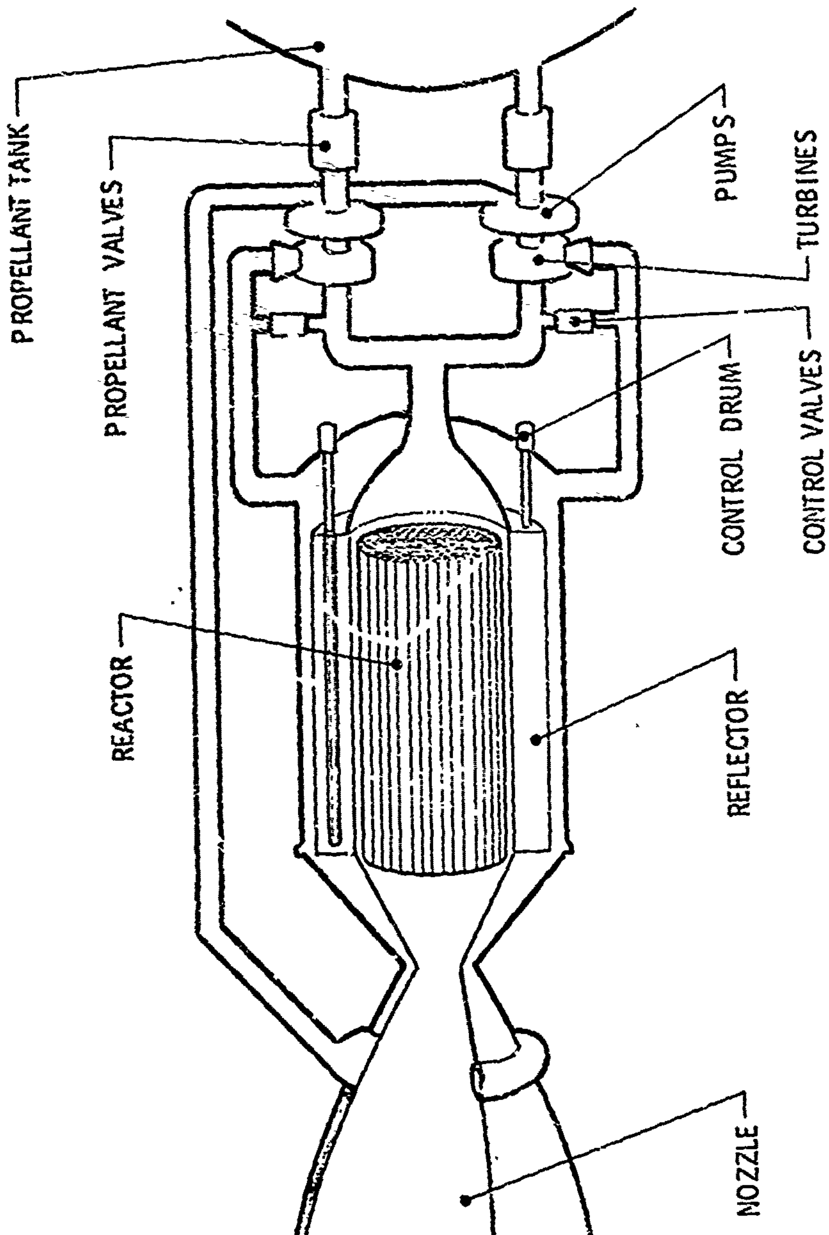

Figure II-5. Schematic of Solid Core Nuclear Rocket Engine

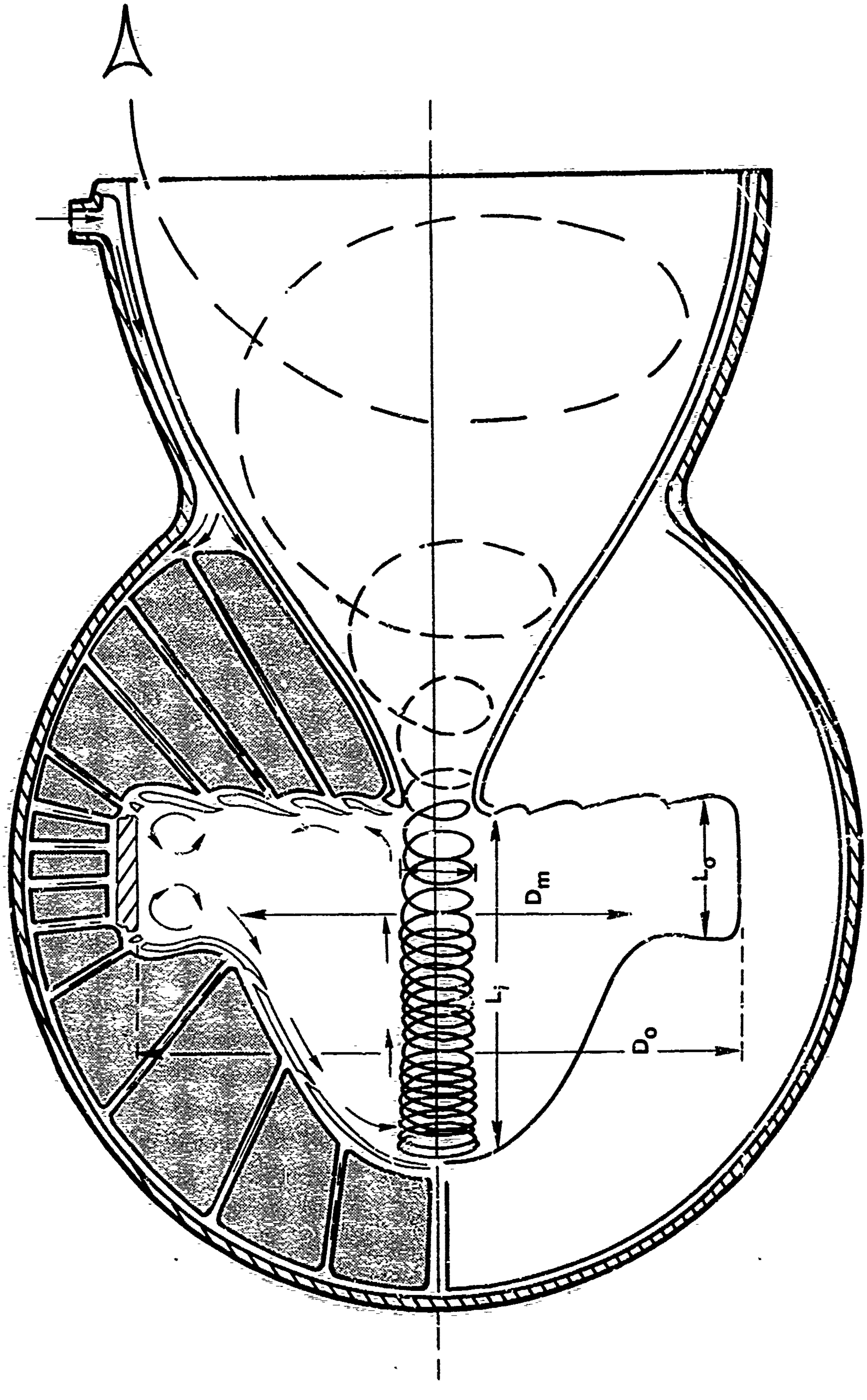

Figure II-6. Schematic View of Colloid Core Reactor.

reactor bed which is suspended in a metal drum rotated by a motor. In both cases, the heated propellant passes through the bed before exhausting out the nozzle.

3. Liquid core nuclear rocket engine: The solid and particle bed engines are temperature limited by the melting point of the reactor core and the pebbles. It is possible that even higher performance can be obtained by bubbling hydrogen gas through a core of molten uranium fuel.

4. Gaseous core nuclear rocket engine: The gaseous core engine concept has the highest performance potential. In this engine, fuel would exist in a gaseous state at extremely high temperatures. In order to prevent loss of the gaseous fuel from the engine, it was deemed necessary to keep the fuel and propellant separate. Thus energy is transferred to the propellant primarily by radiation.

Two gaseous-core concepts are currently being explored. United Aircraft is the prime investigator for the "nuclear light bulb", in which the gaseous uranium is separated from the propellant by a solid wall of fused silicon which is internally cooled. The Lewis Research Center is exploring the coaxial flow reactor in which the propellant flows around, but not through, a central region of gaseous uranium before exhausting out the nozzle.

Analysis: The concept of nuclear-powered rocket propulsion systems appeared first in science fiction stories during the early 1920s, shortly after the realization that nuclear transmutation processes were possible and might yield energy. A typical fission reaction of uranium 235 releases 200 Mev of energy; 167 Mev in heavy fission fragments, about 15 Mev in neutrons, plus gamma and beta radiation. The kinetic energy of these fission fragments is converted to thermal energy by collision with the surrounding medium. When a fission chain reaction is sustained by using enough uranium (critical mass) such that the neutrons emitted in a reaction cause more than one additional uranium nucleus to undergo fission, the surrounding medium is heated. This medium then acts as a heat source for the circulating propellant which is, in turn, raised to a high temperature. Since the energy of the engine is not dependent on the working fluid, any propellant such as air, methane, ammonia or hydrogen which is compatible with the reactor may be used. For example, if high specific impulse is desired, hydrogen may be used as the propellant. This adds a great degree of flexibility to system design. Although nuclear rockets are not a new concept, their development has proceeded to a point at which their possible application should be examined in light of present and future mission requirements.

Nuclear energy may be converted to propulsive thrust by many mechanisms. Direct utilization of pulsed nuclear energy was considered under the code name "Orion" and, although technically attractive, proved to have significant political liabilities. Conversion of nuclear energy to electrical power for use in conjunction with electrostatic, electrothermal or electromagnetic propulsion devices has also been suggested.

In order to deal with each proposed nuclear propulsion device, each will be considered separately. Table II-2 indicates the potential performance of each system considered.

1. The solid core nuclear engine has evolved from a broad base of theoretical and development effort. Work on the NERVA engine was begun in 1956 and in 1958 was taken over by the newly formed National Aeronautics and Space Administration. Most of the work was performed by the Los Alamos Scientific Laboratory, the Aerojet-General Corporation, and Westinghouse Astronuclear. The Nuclear Engine Technology program was completed in September 1969 with the testing of an experimental engine which demonstrated stable performance over a wide operating envelope, restart capability, startup at low tank pressures and engine

controllability without nuclear instrumentation. The principal design goals of a flight configuration have since been proven feasible by an extensive testing program which has shown the NERVA engine to be flexible, stable, safe and reliable (Ref. II-8). If desired, an engine could be prepared for flight testing by 1978.

Work is currently under way at Los Alamos to develop fuel materials which will significantly improve the performance of the solid core nuclear rocket. The development of high-temperature carbides, based on UZrC technology may produce fuels capable of giving a specific impulse near 1000 seconds.

2. Particle bed reactors offering significant increases in performance currently appear to be the logical follow-up to solid core engines. Two concepts are currently being pursued.

Flow studies have been conducted at Brookhaven National Laboratory on a dust bed reactor simulating the suspension of small-diameter uranium fuel particles supported within a porous cylindrical container. The reactor is rotated to hold the particles in place by balancing the drag forces with the centrifugal force. Flow of hydrogen would be radially inward, implying that the rotating machinery, bearings, shafts, etc. can be kept at a reasonable temperature. This fact, coupled with an expected rotational speed of less than 3000 rpm, indicates the concept should be mechanically feasible. Work on this engine has indicated that stable flow conditions can be achieved in a rotating bed. Thermal and nuclear problems seem to be amenable to solution, and moderate engine size and weights appear feasible (Ref. II-9).

TABLE II-2. TYPICAL NUCLEAR ROCKET PERFORMANCE POTENTIAL

	Isp (seconds)	Thrust (pounds)	Engine Weight (pounds)	Thrust/ Weight
Solid Core (NERVA)	825	75,000	23,000	3.26
Advanced Solid Core	970	75,000	30,000	2.50
Dust Bed (BNL)	970	58,000 200,000	23,000 26,500	2.52 7.55
Colloidal Core (ARL)	1100 to 1600	100,000 10,000	20,000 2,000	5.00 5.00
Gaseous Core light blub (UA)	1800	92,000	70,000	1.31
Coaxial Gaseous Core	1800	100,000 1,000,000	200,000 400,000	0.50 2.50

Work done at the Air Force Aerospace Research Laboratory (AFARL) indicates that a bed of colloid size (1 micron or less in diameter) particles may be retained at critical mass in the reactor by appropriate vortex flow of the propellant. Preliminary studies indicates no inherent roadblocks to the development of such an engine (Ref II-10), and a follow-on study is being funded by AFARL and the Air Force Rocket Propulsion Laboratory (AFRPL) to determine more precisely the tradeoffs involved in reactor criticality and thrust-to-weight. The development of colloid-sized particles for solid or particle bed reactors will require a more complete understanding of the fuel loss mechnisms from the high-temperature carbides under consideration.

3. Liquid core nuclear engine research has been, for all pratical purposes, terminated due to the engineering problems involved in bubbling a hydrogen gas through a stable bed of molten uranium. The high potential performance of gaseous core reactors makes research in that area more desirable.

4. United Aircraft Research Laboratories is currently conducting the most extensive work on the "nuclear light bulb" gaseous core engine concept. Current work is aimed at understanding the mechanisms of interaction between the uranium gas and the fused silica wall which contain it, as well as understanding the heat transfer mechanism involved. The results obtained to date have been encouraging. Based on the analyses and experiments conducted to date, this concept appears feasible (Ref. II-11).

A "coaxial-flow" gas core engine concept is being studied in a program at Lewis Research Center (LeRC). Work by LeRC, Douglas Research Labs, Georgia Institute of Technology, and United Aircraft Research Laboratories has shown that seeded transparent gases can be heated by thermal radiation. Current work is aimed at determining a proper flow field for maximum separation of fuel and propellant. Recent developments with curved cavities and gas injection through a porous material have yielded promising results in terms of separation (Ref. II-12).

Shortcomings: The principal shortcomings of nuclear fission engines are their high development costs and long lead times due to the fact that nuclear reactor technology has not yet been operationally proven for use in a vehicle. Experience gained in the NERVA and PLUTO engine development programs, as well as the extensive analyses which are being performed by NASA on the operational requirements of a Reusable Nuclear Shuttle should lessen these difficulties.

In addition, nuclear radiation which poses a potential hazard to equipment and personnel must be taken into consideration in system design. Escaping fission products may pose a sufficient danger to the biosphere to preclude atmospheric use for all concepts but the nuclear light bulb which emits no contaminants. Reactor cores may be designed so as to minimize nuclear contamination, such as was done in the PLUTO program. Shielding of on-board systems adds weight to the engines, which in general makes nuclear engines heavier than chemical engines.

Finally, the necessity of using some propellant to cool the engine after shutdown may lead to a degradation of Isp from the high values expected. Nevertheless, specific impulse would still be attractively high.

Conclusions: Many of these shortcomings associated with nuclear propulsion are amenable to engineering solutions, and none sufficiently impair performance to rule out the use of nuclear fission engines. A need exists to show a sufficient economic benefit to warrant the development costs and effort required to overcome the hazards associated with nuclear devices.

Nuclear fission rockets offer large performance gains over chemical rockets, especially where high ΔV, large spacecraft and multiple missions are required. Because of their unique features (high Isp, choice of propellant, and long life) nuclear engines offer a high degree of flexibility to mission planners. For example, missions requiring a ΔV greater than 30,000 feet per second and fast response time may not be achievable without nuclear engines.

The development of technology for nuclear rocket engines is well under way. A solid core engine could be operational by 1978. Advanced designs could follow as mission requirements become more severe. The most pressing problem is political. Potential users seem hesitant to turn to nuclear engines, even with their greater performance, because of the uncertainties involved in going to a new energy source.

Recommendations: Because of the long lead times necessarily required with any new technological development, such as nuclear fission rocket engines, effort should be made to stimulate thinking in the Air Force to define space propulsion requirements for the 1980 to 2000 time period. Imaginative uses of nuclear engines should be investigated. However, certain factors indicate that nuclear rocket development should be continued without specific applications in mind. There are presently several alternative design options being explored, well-defined technological milestones to be met, and a trained manpower supply available.

When related to present and near-term Air Force missions, nuclear rockets offer significant advantages. These include the ability to deliver large payloads to high or synchronous orbit and return, make large plane changes, operate for long times, stay at a station for years before use, and operate with air as a propellant. If a requirement materializes for operation in near-earth space on a continuous basis for many years the development of nuclear engines could be the most efficient method of operation.

Title: Thermonuclear Propulsion (Fusion)

Concept: This concept utilizes the energy released from thermonuclear reactions for propulsion.

Attributes: In principle, the energy and performance of thermonuclear propulsion is second only to the hypothetical photon rocket which uses mass annihilation. The performance can be tailored to fit mission requirements; thermonuclear propulsion systems are inherently failsafe; and fusion uses inexpensive propellants readily available in almost unlimited quantities that produce no residual radioactive waste.

Description: There are a number of ways in which thermonuclear energy may be applied to propulsion. Confined in a closed reactor, the energy of fusion may be converted directly to electricity and used to power an advanced electric thruster. By allowing a portion of the hot plasma from a closed reactor to be expanded through a magnetic nozzle, thrust may be obtained directly with very high specific impulse. This idea is sometimes called the "leaky bottle." Similar to a fission solid core rocket, fusion energy may be used to heat a working fluid such as hydrogen for expansion through a conventional nozzle. Recent studies have indicated that fusion energy may be effectively utilized in a pulsed mode initiated by the intense light of lasers. Fusion pulsed propulsion systems may be internal or external. If the reaction takes place in a chamber with the hot plasma expanded through a "nozzle" the concept is internal. The external concept employs an explosion behind the vehicle with the thrust obtained through a momentum exchange between the rapidly expanding plasma and a mechanical shock absorber. This external concept is like the Orion concept for nuclear fission. The internal technique is very similar to chemical combustion only the energies involved are much greater.

Analysis: Recent advances in controlled thermonuclear fusion research have given rise to renewed optimism regarding the practical feasibility of obtaining energy from confined fusion reactions. A major breakthrough in plasma containment time was recently obtained in the USSR by the Tokamak-3 magnetically closed toroidal reactor. The Tokamak concept offers the potential of obtaining a net power producing reactor. Another promising approach to obtaining energy from fusion uses lasers. Here, powerful pulsed laser light, focused on frozen fuel pellets, initiates energy-releasing fusion reactions. Also, experiments indicate that useful electrical power may be obtained without a confining magnetic field (Ref. II-13). Accordingly, it has been stated that there are now no known reasons to prevent controlled thermonuclear fusion.

Conceptually, any two nuclei can undergo fusion. Fusion is a process of "building of elements" from lighter elements. To fuse or stick together, atoms must collide violently. The energy release is proportional to the binding energy of stable elements. However, the lighter nuclei are preferred since a greater energy yield per fusion results and radiation losses are minimized. The best candidate for a fusion fuel is deuterium. Because deuterium and hydrogen differ so much in atomic weight, deuterium is inexpensively separated and the technology for producing heavy-water far advanced. Each gallon of sea water contains 1/6 teaspoon of heavy-water, and the cost of extracting that quantity of deuterium is less than ten cents. Lithium-6, another possible propellant component, is relatively abundant and inexpensive (Ref. II-14). The four fusion reactions that use deuterium are listed below.

$$ {}_1D^2 + {}_1D^2 \rightarrow ({}_2He^3 + 0.8 \text{ mev}) + ({}_0n^1 + 2.45 \text{ mev}) \quad (1)$$

$$ {}_1D^2 + {}_1D^2 \rightarrow ({}_1T^3 + 1.0 \text{ mev}) + (H^1 + 3.0 \text{ mev}) \quad (2)$$

$$ {}_1D^2 + {}_1T^3 \rightarrow ({}_2H^4 + 3.6 \text{ mev}) + ({}_0n^1 + 14.1 \text{ mev}) \quad (3)$$

$$ {}_1D^2 + {}_2He^3 \rightarrow ({}_2He^4 + 3.7 \text{ mev}) + ({}_1H^1 + 14.6 \text{ mev}) \quad (4)$$

In these equations D stands for deuterium, T for tritium, H for hydrogen, He for helium, and n for a neutron. The subscripts refer to the number of protons and the superscripts refer to the nucleon's weight (protons plus neutrons). The resulting distribution of energy among the particles is given in millions of electron volts. Indications are that the first two reactions take place with about equal probability. However, there is some temperature dependence. Reaction (3) is about 100 times more probable than reactions (1) and (2). The last reaction is the least likely. All four reactions are required to complete the fusion of deuterium to helium.

Because the third reaction using tritium is so much more probable and leads to the lowest ignition temperatures, it is tempting to use it. Tritium can be generated through another nuclear reaction using Lithium-6:

$$ {}_{0}n^{1} + {}_{3}Li^{6} \rightarrow {}_{1}T^{3} + {}_{2}He^{4} + (4.6 \text{ mev}) \qquad (5) $$

The 14.1 mev neutron from Reaction (3) might be used in this "breeder" reaction to obtain a closed cycle balance. Thus lithium could act as a neutron moderator of energetic neutrons and a source of tritium. Unfortunately, such a process introduces complications and disadvantages (Ref. II-14).

The ignition of a fusion reaction is initiated by raising a quantity of deuterium gas to a temperature approaching 10^8 °K. At such temperatures, the gas is fully ionized and has become a plasma (free electrons and ions with equal charge density). Furthermore, in order for the particles to react, the plasma must be confined in a relatively fixed volume so that the reactions can occur. The probabilities or cross sections of the fusion reactions as a function of particle energy are plotted for the fusion reactions in Figure II-7. An average energy of 10 kev corresponds to a temperature of about 10^8 °K. Once the ignition temperature is attained and the fusion reactions initiated, a self-sustaining net energy-producing process is still not guaranteed. Clearly, an answer to this problem depends upon how much energy will be lost by radiation and other mechanisms. Major energy loss mechanisms affecting thermonuclear reactions are:

1. Radiation Losses: These are unavoidable losses primarily in the form of X-rays that occur when electrons collide with nuclei. These losses are known as bremsstrahlung radiation. The energy loss rate or power loss is proportionally given by:

$$P_L \propto \Sigma_i(\eta_i Z_i) \times \Sigma_j(\eta_j Z_j^2)\, T_e^{1/2} \qquad (6)$$

where P_L is the possible loss, η the ion number density, Z the ion atomic number, and T_e the electron temperature. The subscripts denote the ion species. It is important to note that the bremsstrahlung losses increase dramatically with plasma impurity content. For example, a D-D reaction with a 1 percent oxygen impurity entrained in the plasma suffers bremsstrahlung loss 77 percent greater than a pure D-D reaction. Cyclotron or magnetic bremsstrahlung losses occur for magnetically confined geometries only and arise due to the gyration of electrons about magnetic field lines.

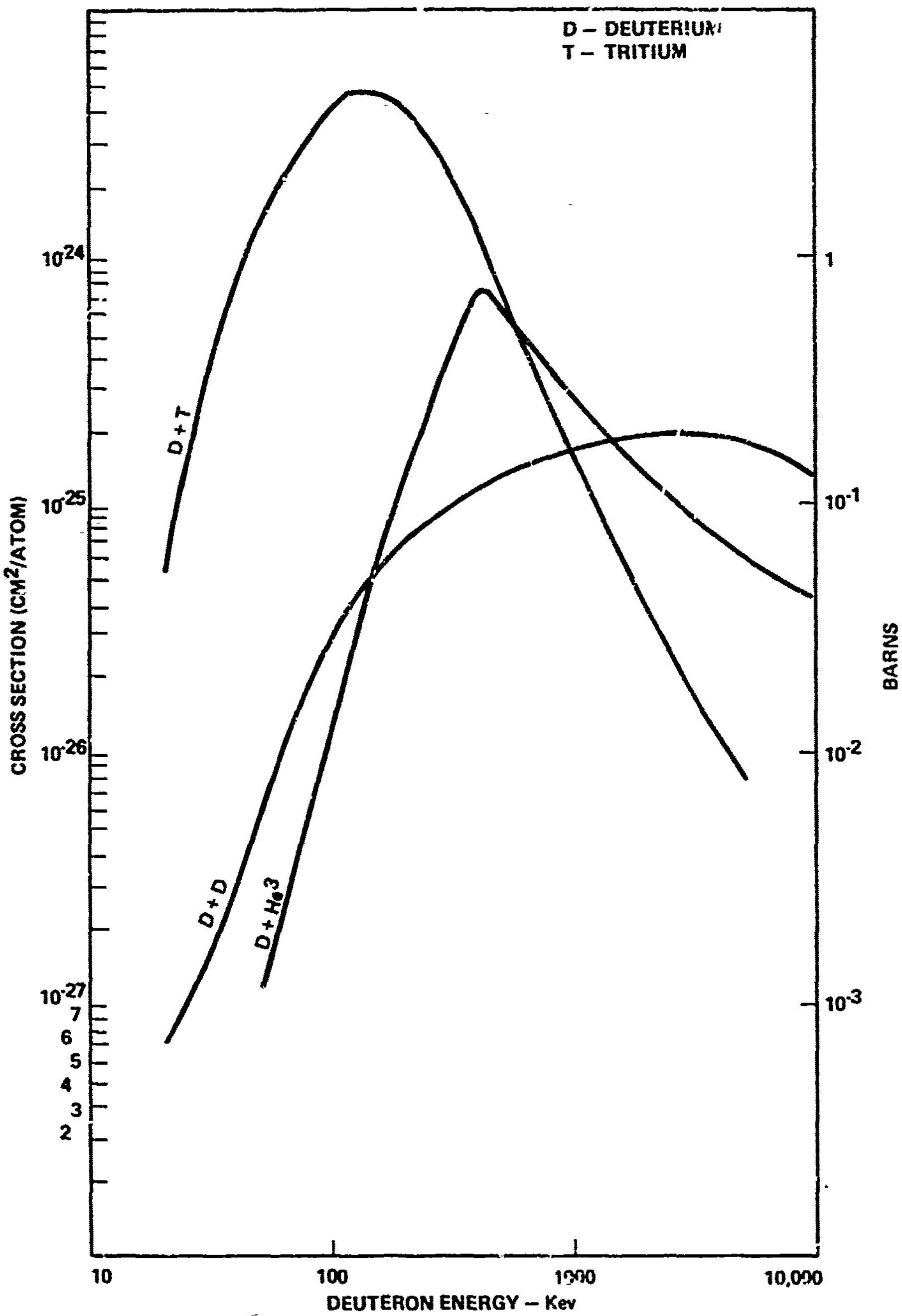

Figure II-7. Cross Sections for D-T, D-D (Total) & D-He^3 Reactions (Compiled from ORNL3113 Revised Aug-64)

This power loss apparently occurs in the infrared region and is proportionally given by:

$$P_L \propto \Sigma_i(\eta_i Z_i^2)\, T_e^2 \tag{7}$$

When the plasma temperature $T = T_e$, this loss allows only the D-T reaction to occur unless the radiation can be partially reabsorbed within the plasma. However, if the radiation is not reabsorbed, the presence of this loss mechanism casts grave doubt on the practicality of magnetically confined reactor geometries. A simple power balance accounting for only bremsstrahlung losses in the D-D and D-T reactions are shown in Figure II-8. The ideal ignition temperatures for these reactions are seen to be approximately 5 kev for D-T and 36 kev for D-D.

2. Temperature Losses: Charge exchange losses occur when heated ions accept electrons from cold neutrals which in turn become cold ions that need to be heated to maintain the fusion reaction. Such losses appear to preclude fusion ignition below T = 100 kev when excessive neutrals are present. Contact with the walls of a container can also cause significant energy losses. Contrary to popular opinion, the reason for avoiding contact with the wall is not that the plasma will vaporize the walls, but that contact with the walls will instantly quench the reaction by cooling the plasma (Ref. II-14).

The practical problems involved in creating the conditions for ignition of fusion reactions are great. Not only must the deuterium gas be heated to extreme temperatures, but it must be contained for a sufficient period of time at that temperature in such a way that the energy output appreciably exceeds the energy input. The means to accomplish containment require a detailed understanding of plasma instabilities, energy loss mechanisms, and heating techniques. Trade-offs must be made between the number density of fusionable ions (η), the plasma or ion temperature (T), and the containment time (τ_c).

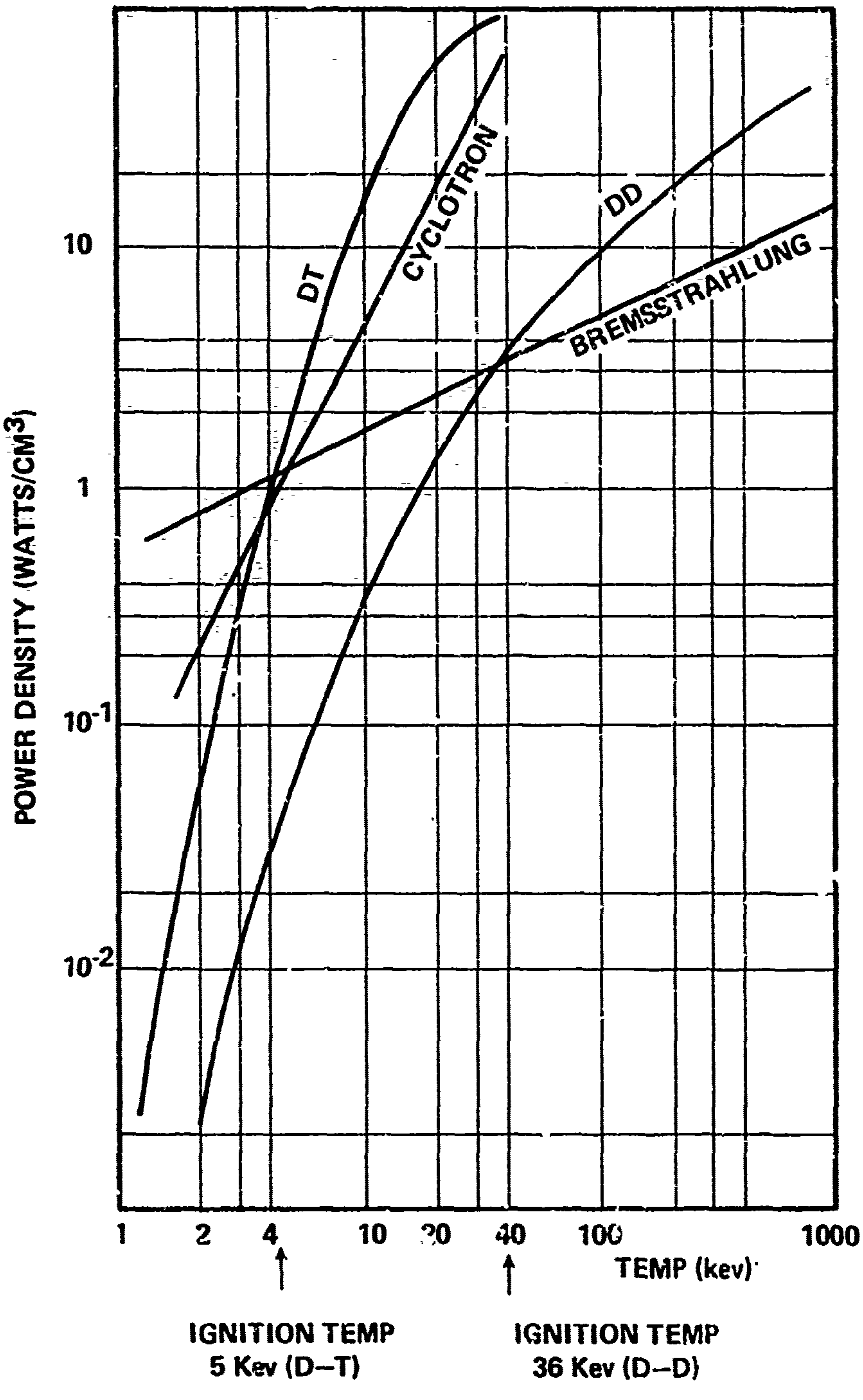

Figure II-8. Power Balance

One of the most promising approaches to the confinement problem has been to suspend the plasma in a strong magnetic field. This is the "magnetic bottle" concept. Various configurations have been studied. They include linear configurations, toroidal configurations, and other configurations that combine certain attractive features (Ref. II-15). However, this approach has continuously been hampered by plasma instabilities. Plasma instabilities are disturbances that grow in either space or time such that the plasma is dispersed. Techniques exist to eliminate gross or macroscopic instabilities. Microinstabilities arise from distribution inhomogeneities and plasma property gradients have not been controlled. Yet most of these microinstabilities appear manageable. At one time, the problem of plasma diffusion losses across magnetic field lines in magnetically confined reactor geometries appeared insurmountable. Recently major breakthroughs have occurred in this area, and the empirical "Bohm" time barrier (the maximum confinement time achievable before diffusion) has been exceeded by two orders of magnitude. Figure II-9 illustrates the current experimental status of magnetically confined reactors. On a T versus natural log ($\eta\tau_c$) diagram, a number of the more promising concepts are shown. The shaded region of the figure corresponds to a gain in net power output over input for a D-T reaction and follows from a simple power balance considering only bremsstrahlung losses. From Figure II-9 it is evident that not one of these current research reactors delivers a useful output.

More recently, it has been suggested that magnetic confinement may not be necessary if the energy required to initiate fusion reactions can somehow be delivered to a deuterium-tritium mixture rapidly enough to allow the reactions to take place before the resulting plasma can disperse. Such a method is available using high-energy pulsed lasers. Current concepts utilize frozen pellets of deuterium and tritium. The laser energy is delivered in two pulses. The first pulse contains enough energy to merely vaporize a portion of the pellet. The second pulse contains

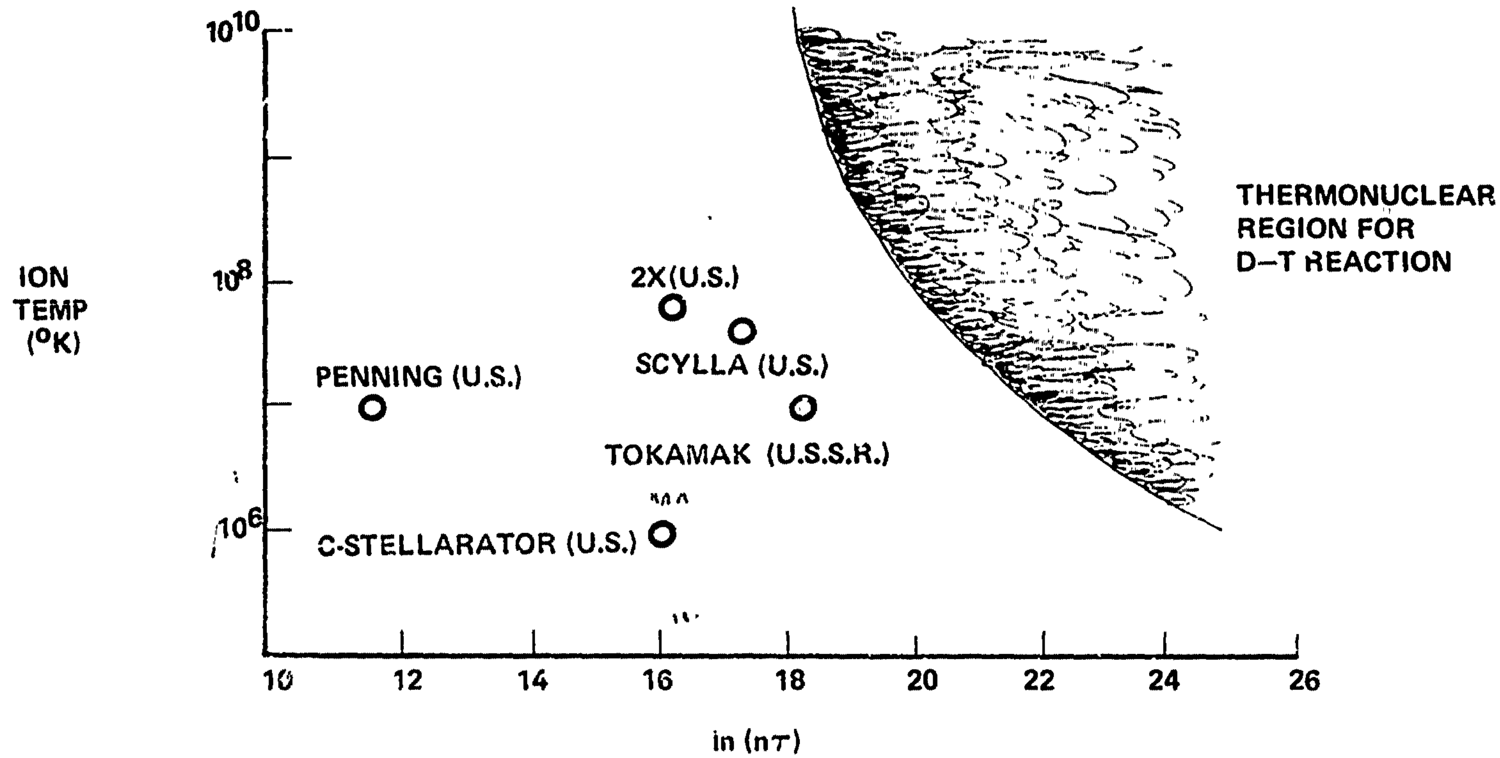

Figure II-9. Experimental Status

sufficient energy to initiate the fusion reactions. These laser pulses are delivered in extremely short bursts of energy ($\approx 10^{-10}$ seconds). Such short pulses are required because the hydrodynamic dispersion rate of gases or plasma is very rapid. This technique correspondingly requires ion densities of higher magnitude than magnetically confined geometries. The details of the physics embodying the interactions between the laser electromagnetic wave and the gas or plasma are still not completely understood. Theoretical results and laboratory tests, however, are encouraging. The development of laser-triggered, unconfined thermonuclear reactions is very attractive and removes many of the inherent disadvantages of the magnetically confined systems. Minus the magnets and cooling system, the pulsed reactor is significantly smaller and lighter. Also, many of the loss mechanisms, such as cyclotron losses, are eliminated from the beginning.

Over the years, a number of propulsion concepts have evolved using the energy of fusion. It should be remembered that the requirements of a propulsion system for space flight are thrust, whereas terrestrial uses for fusion involve the production of energy (i.e., electricity). Classically, for propulsion, the first assumption usually involved a containment technique. Then, some method was envisioned to accelerate matter rearward to produce a thrust. The general picture of fusion propulsion as illustrated in Figure II-10 is seen to be extremely complex. The engineering feasibility associated with thermonuclear propulsion remains to be demonstrated. Using magnets and complete confinement, fusion energy can provide electrical power for advanced electric thrusters or heat a working fluid for expansion through a conventional nozzle. By only partially confining the fusion reaction, a portion of the hot plasma may be allowed to "leak" unidirectionally through a magnetic nozzle to produce a thrust. This is the "Leaky Bottle" concept. Very recently, new concepts taking advantage of high-energy, pulsed lasers to initiate fusion reactions have been described. They involve both external and internal designs. The magnetically contained concepts are shown in Figure II-11 and the unconfined laser-triggered concepts are shown in Figure II-12.

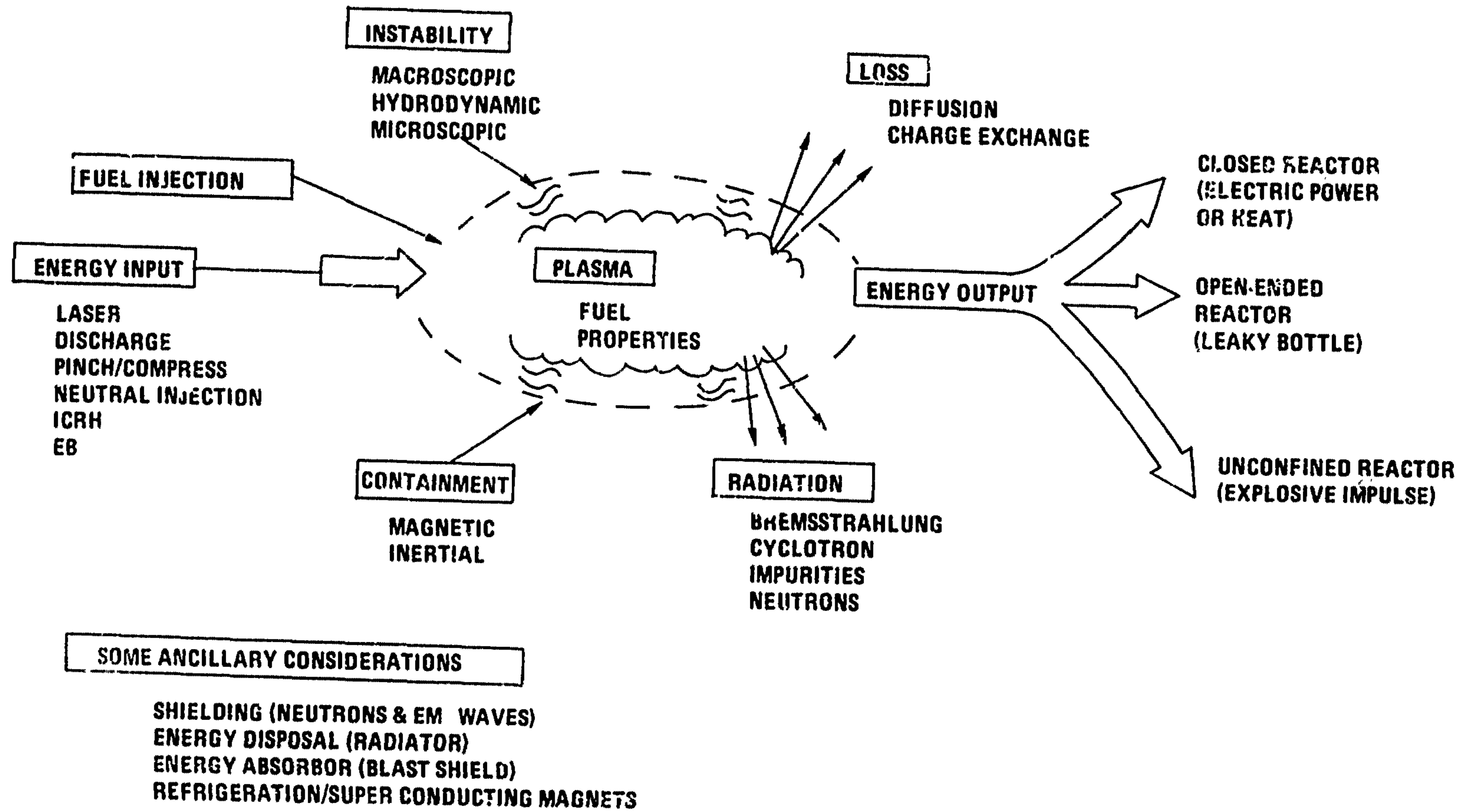

Figure II-10. General Fusicn Propulsion Problem

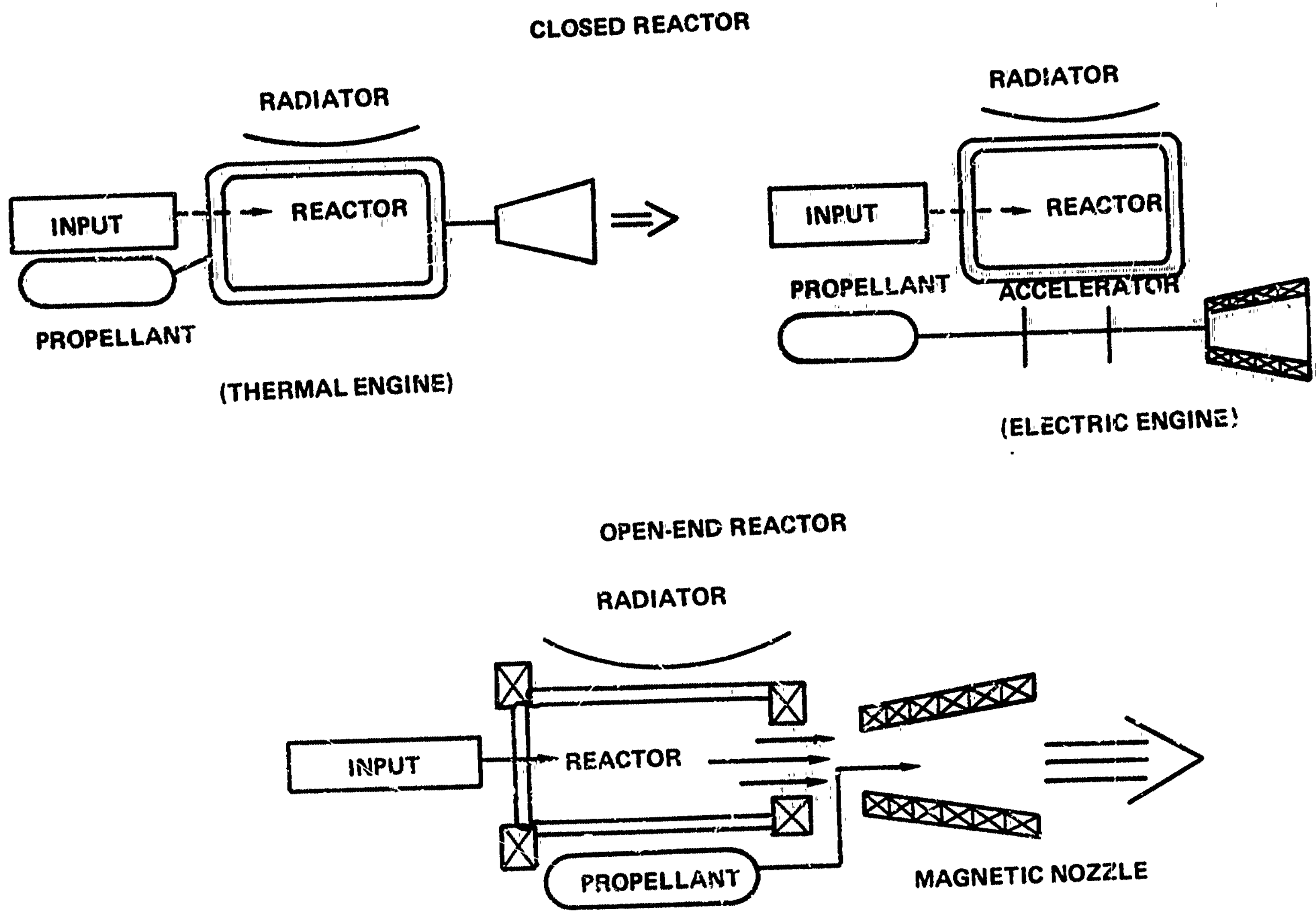

Figure II-11. Magnetically Confined Systems

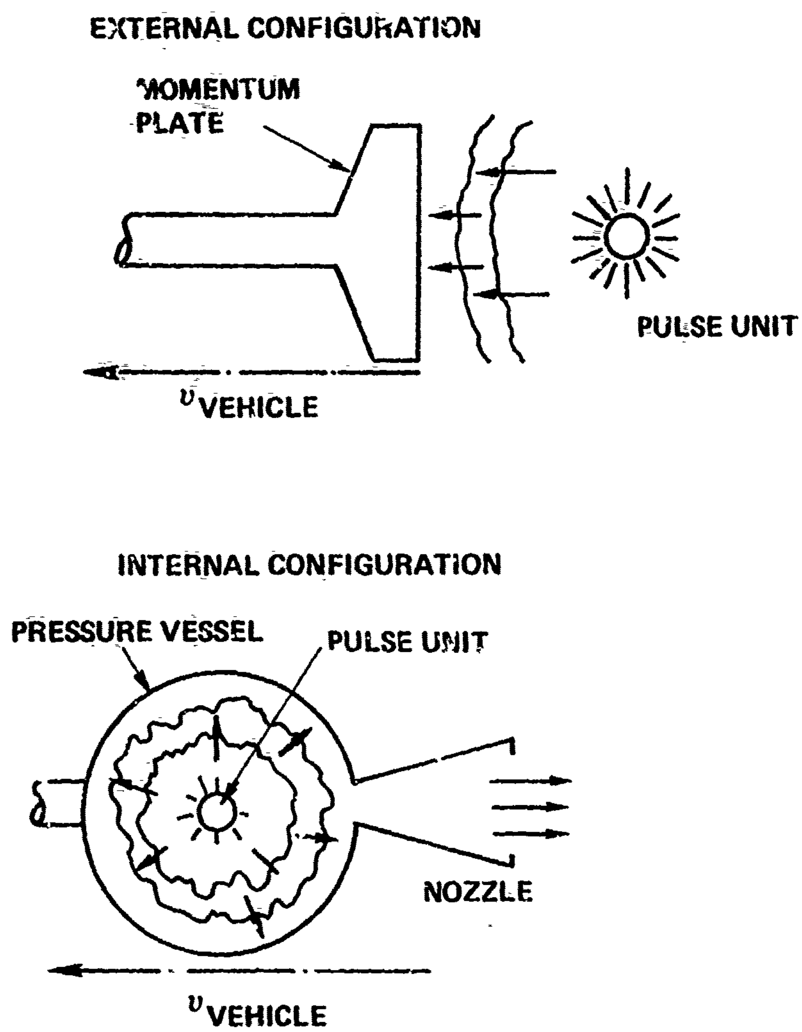

Figure II-12. Unconfined Pulsed Systems

In principle, thermonuclear propulsion offers a vast operating range with performance superior to other propulsion techniques. Because of the tremendous quantities of energy available, fusion propulsion systems can qualify for use on planetary landing, interplanetary and interstellar missions. Fusion propulsion is literally the propulsion technique that can take man to the stars. The superiority of thermonuclear propulsion in comparison with other concepts is illustrated in Figure II-13. Nominal values of performance for the specific fusion concepts discussed above are illustrated in Table II-3.

TABLE II-3. PERFORMANCE OF FUSION PROPULSION CONCEPTS

Concept	Isp	Thrust/Weight Ratio
Electrical Power Generation	10^4	10^{-2}
Working Fluid Heating	10^3	5
Leaky Bottle	10^5	10^{-3}
Unconfined External with Lasers	5×10^3	50

Comparing the thermonuclear propulsion concepts of Figures II-11 and II-12, the broad performance spectrum noted earlier should be obvious. Equally obvious should be the extensive listing of research and development areas that require investigation. For the magnetically confined reactor geometries to be practical, lightweight magnets, cryoplants, energy shielding and disposal systems all require extensive study. For the leaky bottle magnetic reactor geometries, the physics of mixing fusion products with a secondary working fluid for thrust augmentation needs to be elucidated. For unconfined or inertially confined geometries, the pellet design, deployment and initiation systems need to be extensively investigated. Additionally, a lightweight laser initiation system needs to be

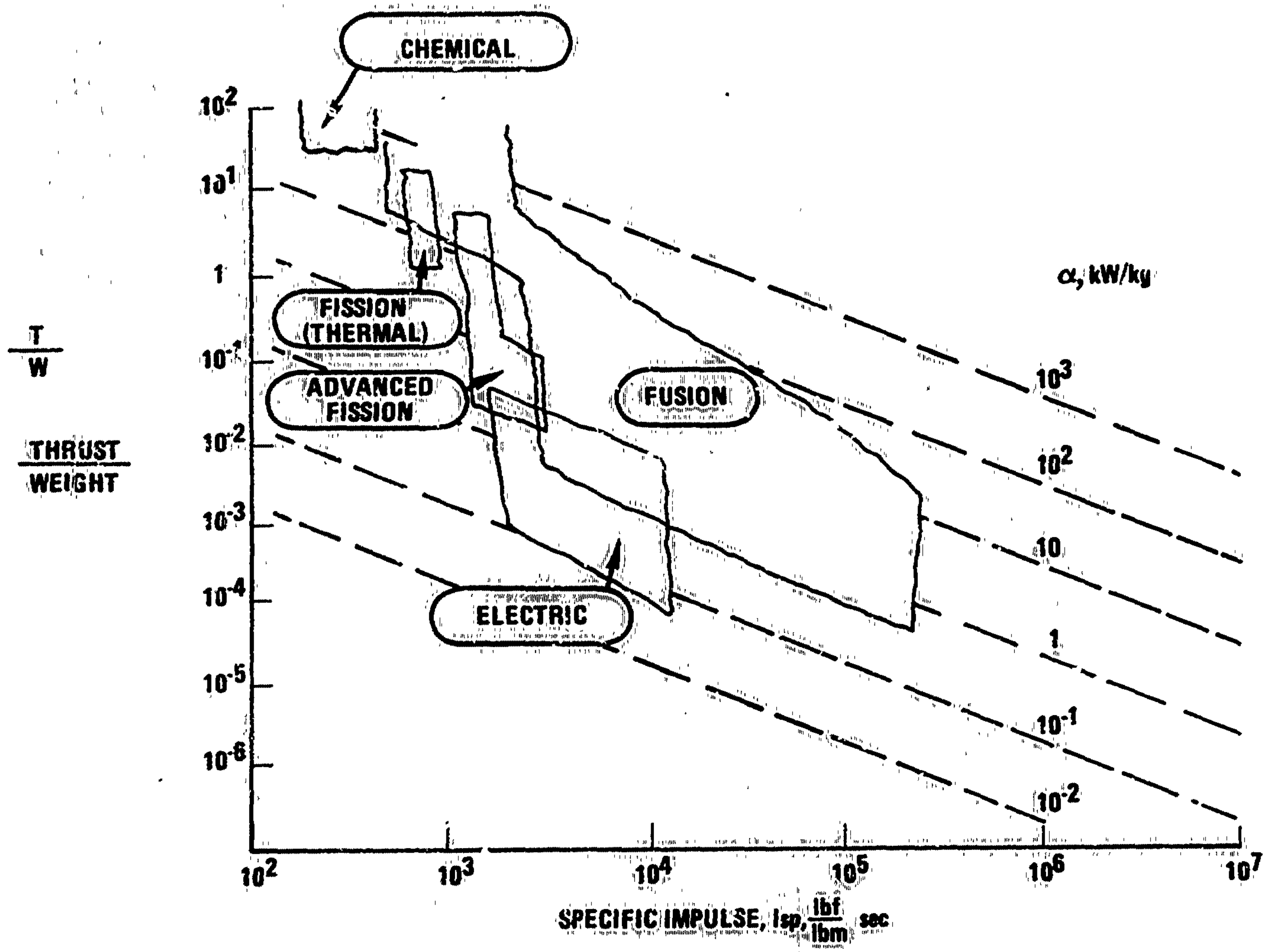

Figure II-13. Performance Features for Propulsion Systems developed. Energy extraction methods, shielding studies, material problems, and neutron heating and activation considerations, among others, require investigation for each concept.

For conceivable military missions in the next 10 to 20 years, it seems clear that reasonable thrust levels are demanded. Only the unconfined laser-initiated geometries operating in the pulsed mode appear to satisfy this demand of reasonable thrust/weight over a considerable time period. In principle, military surveillance, intercept, ballistic and exploratory missions, among others, might be performed using a thermonuclear pulsed propulsion device. This approach eliminates the attendant problems associated with magnetically confined reactor geometries. Accordingly, the pulsed thermonuclear propulsion concept is presently the most promising.

Extensive further work in the pulsed propulsion area has been recently completed (Ref. II-16). This work concentrated on the pulsed thermonuclear propulsion concept utilizing fusion reactions initiated by high-energy lasers. The objectives were to:

1. Define the operation and the details of the operating range of thermonuclear pulsed propulsion vehicles.

2. Conduct and review preliminary designs for pulsed thermonuclear propulsion configurations including major subsystems (both internal and external configurations were studied. (See Figure II-12.)

3. Identify, with priorities, areas requiring further work to demonstrate the pulsed thermonuclear propulsion concepts.

To accomplish the above, efforts were concentrated in the following areas: (1) basic concept, (2) review of laser energy sources and pellet designs, (3) pulsed thermonuclear propulsion performance mapping, (4) environmental considerations, (5) review of actual propulsion system configurations, and (6) areas for further investigation. The results conclusively demonstrated that pulsed thermonuclear propulsion systems afford outstanding performance opportunities. The pulsed internal configuration was found, however, to be severely limited by the inability of materials to withstand a severe neutron and gamma ray environment. In addition, the performance of the internal system in terms of Isp, payload fraction, mission ΔV capacity and vehicle "launch" mass, was found to be always below that of the pulsed external configuration. The external system also exhibited a remarkable capability to perform a multitude of missions other than the specific mission for which the system was designed. The use of extremely small overall vehicle masses to accomplish very ambitious missions also appears possible. Lastly, it appears that pulsed systems offering very high Isp (up to, say, 6500 seconds) can be tested at reasonable exclusion distances where safety from neutrons and gamma rays can be virtually guaranteed.

Shortcomings: The concepts involving magnetically confined thermonuclear propulsion lack considerable proof of engineering feasibility. At present there appears to be no inherent technical disadvantages for attainment of a pulsed thermonuclear device. However, detailed information concerning all aspects of this concept is not yet available; thus, uncertainty is the main disadvantage.

Conclusions: Based upon the considerations, and as a result of extensive discussions with leading scientists of the USAEC, NASA, universities and the USAF who are working in thermonuclear fusion research, and upon the detailed findings of an extensive nuclear pulsed propulsion study, it is concluded that:

1. The scientific feasibility of controlled thermonuclear fusion will be demonstrated within 5 years.

2. For propulsion application on military missions in the foreseeable future, the pulsed mode of operation is preferred.

3. The external thermonuclear pulsed propulsion device provides a broad range of performance where operation is vastly superior to other propulsion techniques. This allows a correspondingly broad range of military missions using thermonuclear propulsion.

Title: Infinite Isp Ramjet

Concept: The fundamental concept of the infinite Isp ramjet is to replace the ramjet combustion process with themal energy supplied by a laser beam.

Attributes: In theory, the infinite Isp ramjet can perform any mission within the capability of more conventional ramjets. Since no fuel is carried, the infinite Isp ramjet is sized by the required payload, allowing vehicles to attain minimum size. The range and time of flight of infinite Isp ramjets are unlimited.

Description: The infinite Isp ramjet is one of a number of concepts that achieve infinite specific impulse or "ideal" operating conditions by interaction with the environment. This system receives both its working fluid and energy from external sources. As originally proposed, a ground-based laser beam is used to transmit large amounts of energy to the vehicle. The laser beam is converted, via a receiver/energy converter, to thermal energy which is transferred to the working fluid (air) by means of a system of heat pipes and a heat exchanger. The heat exchanger is in direct contact with the ram air which is heated before expansion through a conventional nozzle. This concept is illustrated in Figure II-14.

Analysis: The analysis of the infinite Isp ramjet is perhaps best approached on an efficiency basis. As initially envisioned, the infinite Isp ramjet requires a number of discernible processes. The cumulative efficiency of these processes determines system practicality.

Assuming that a high-energy laser beam can be generated, the attenuation of beam energy by the atmosphere must be accounted for. The atmosphere constitutes an absorbing and scattering medium with the specific coefficients strongly dependent on beam wavelength. Utilization

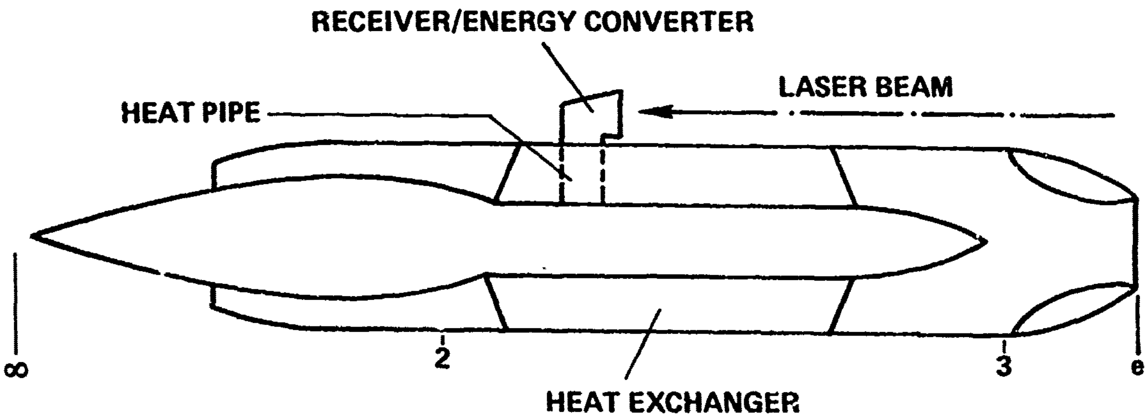

Figure II-14. Infinite Isp Ramjet Schematic of the correct beam wavelength can minimize the attenuation problem, but in general, the beam will suffer energy losses. In the case of the infinite Isp ramjet, a meaningful efficiency measure of the attenuation problem is the beam energy received to beam energy emitted ratio. This efficiency should be a function of the slant range to the laser source, the altitude of the laser source, and the altitude of the ramjet.

Another important efficiency is the percentage of beam energy converted to useful thermal energy. This efficiency will be principally dependent upon the design of the receiver/energy converter unit. If the receiver/energy converter is located remotely from the heat exchanger, an energy transmission efficiency becomes pertinent. In the infinite Isp ramjet, this efficiency would be a measure of the losses in the heat pipes. It might be necessary to eliminate this loss by consolidating the energy converter and heat exchanger design, recognizing that this would induce flight attitude constraints on the ramjet.

Finally, the efficiency of the heat exchanger must be accounted for. There exists a considerable body of literature on the design of heat exchangers in flowing systems which should permit analysis of this efficiency. Detailed design of a heat exchanger was, however, considered unnecessary for this concept evaluation.

A preliminary evaluation of infinite Isp ramjet performance can be made assuming all of the above efficiencies to be in unity. The intent of this study is to determine the minimum necessary power level of the laser source. In order to do this, some evaluation of the thrust required as a function of flight condition must be made. For the purposes of this preliminary investigation, assume that the flight vehicle will be essentially a ramjet nacelle as illustrated in Figure II-14. The vehicle payload, in this configuration, must be carried in the inlet centerbody. For a vehicle configuration as indicated, an approximate zero lift drag variation is assumed as illustrated in Figure II-15. The drag characteristics are optimistic in that no inlet spillage is assumed, and no allowance is made for frictional losses.

If the drag increment due to angle of attack is neglected, then at a given flight condition:

$$\text{Drag} = D = C_{D_O}\, q \cdot A_{2-3}$$

where A_{2-3} is the cross-section area of the ramjet between stations 2 and 3. Assuming that the capture area, A_∞, is approximately equal to the inlet throat area, and assuming a dump Mach number of about 0.2, then:

$$\frac{A_{2-3}}{A_\infty} = \frac{(A/A^*)_{M\,=\,0.2}}{(A/A^*)_{M\,=\,1.0}} = \frac{(1/0.3374)}{1} = 2.96$$

Simplifying, let $A_{2-3} = 3A_\infty$

With the above assumptions, thrust required per unit cross-sectional area is given by:

$$F_{reqd} = \frac{D}{A_{2-3}} = C_{D_O}\, q = C_{D_O} \left(\frac{q}{M_\infty^2}\right) M_\infty^{\,2}$$

The ramjet thrust can be expressed as (heat addition is assumed to occur between stations 2 and 3. See Figure II-14):

$$F = \dot{m}_e V_e - \dot{m}_\infty V_\infty + A_e (P_e - P_\infty)$$

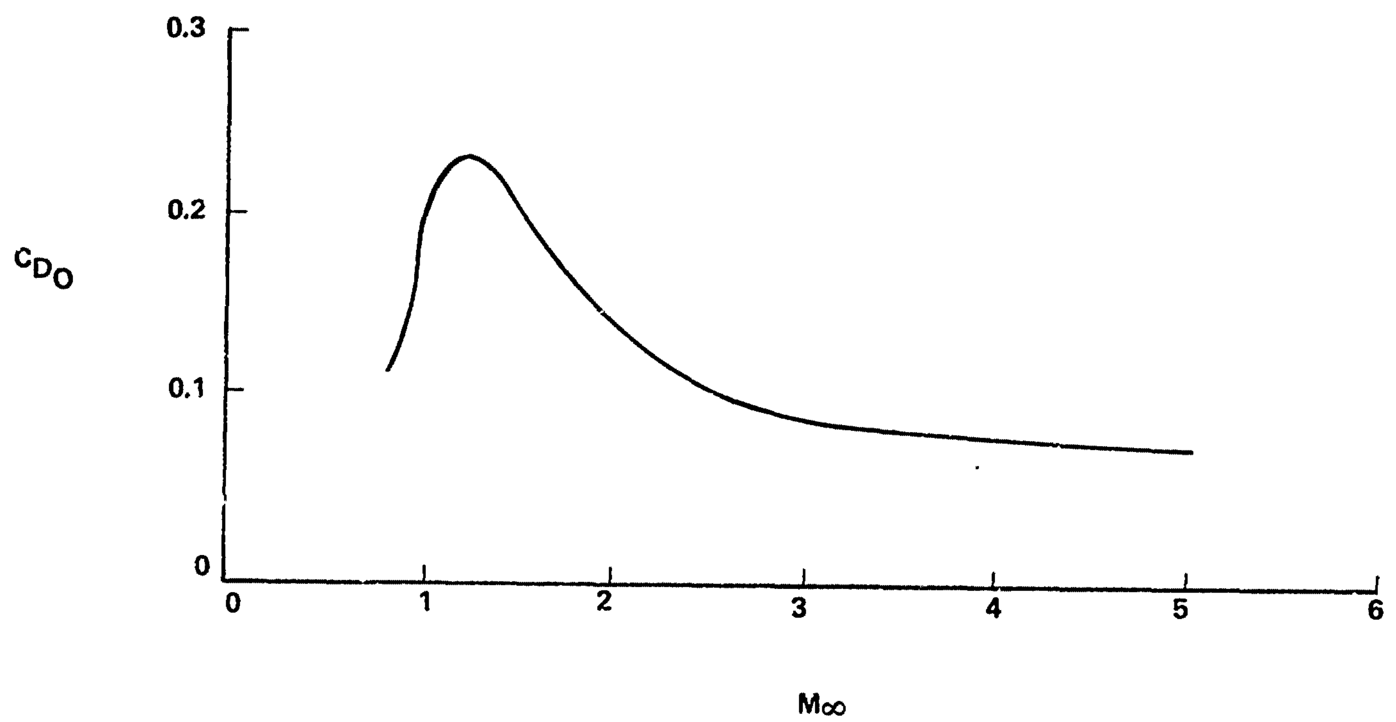

Figure II-15. Ramjet Zero Lift Drag

Introduce the impulse function ($\mathcal{J}$):

$$\mathcal{J} = \dot{m}V + pA$$

Assuming an ideal gas:

$$\mathcal{J} = \dot{m}V + pA = AV^2 + pA = \frac{P}{KT}\frac{\gamma}{\gamma}AV^2 + pA$$
$$= pA\ (1 + \gamma M^2)$$

So, the ramjet thrust becomes:

$$F = \mathcal{J}_e - \mathcal{J}_\infty - P_\infty\ (A_e - A_\infty)$$

or:

$$F = P_e A_e\ (1 + \gamma_e M_e^2) - P_\infty A_\infty\ (1 + \gamma_\infty M_\infty^2) + P_\infty A_\infty - P_\infty A_e$$

$$F = P_e A_e (1 + \gamma_e M_e^2) - P_\infty (A_\infty \gamma_\infty M_\infty^2 + A_e)$$

Introduce the effective exhaust jet velocity defined by:

$$V_j = V_e + \frac{A_e}{P_e A_e V_e}\ (P_e - P_\infty)$$

And the effective exit Mach number defin^d by:

$$M_j = \frac{V_j}{a_e}$$

where a_e is the sonic exit velocity.

In terms of the effective velocity, the thrust may be restated as:

$$F = \rho_e A_e V_e^{\,2} - \rho_\infty A_\infty V_\infty^{\,2} + A_e (P_e - P_\infty)$$

$$F = \rho_e A_e V_e (V_j) - \rho_\infty A_\infty V_\infty^{\,2}$$

$$F = \dot{m}_e V_j - \dot{m}_\infty V_\infty$$

A more practical result is to evolve the equivalent expression of thrust in terms of effective Mach number. Thus:

$$F = \dot{m}_e V_e + A_e (P_e - P_\infty) - \dot{m}_\infty V_\infty$$

$$F = \dot{m}_\infty V_\infty \left[\frac{\dot{m}_e V_e + A_e \; P_e - P_\infty) - \dot{m}_\infty V_\infty}{\dot{m}_\infty V_\infty} \right]$$

$$F = \dot{m}_\infty V_\infty \left[\frac{\dot{m}_e V_j}{\dot{m}_\infty V_\infty} - 1 \right]$$

$$F = \dot{m}_\infty V_\infty \left[\frac{\dot{m}_\infty + \dot{m}_f}{\dot{m}_\infty} \cdot \frac{V_j}{V_\infty .} - 1 \right]$$

where $\dot{m}_f$ = fuel mass flow. Continuing

$$F = \dot{m}_\infty V_\infty \left[(1 + f) \; \frac{V_j}{V_\infty} - 1 \right]$$

where f = fuel-to-air ratio. Proceeding:

$$F = \dot{m}_\infty V_\infty \left[(1 + f) \frac{V_j}{\sqrt{\gamma_e R_e T_e}} \frac{\sqrt{\gamma_e R_e T_e}}{V_\infty} - 1 \right]$$

$$F = \dot{m}_\infty V_\infty \left[(1 + f) M_j \frac{\sqrt{\gamma_e R_e T_e}}{\sqrt{\gamma_\infty R_\infty T_\infty}} \frac{\sqrt{\gamma_\infty R_\infty T_\infty}}{V_\infty} - 1 \right]$$

assuming $R_e = R_\infty$;

$$F = \dot{m}_\infty V_\infty \left[(1 + f) \frac{M_j}{M_\infty} \sqrt{\frac{\gamma_e T_e}{\gamma_\infty T_\infty}} - 1 \right]$$

or:

$$F = P_\infty A_\infty \gamma_\infty M_\infty^2 \left[(1 + f) \frac{M_j}{M_\infty} \sqrt{\frac{\gamma_e T_e}{\gamma_\infty T_\infty}} - 1 \right]$$

For the infinite Isp ramjet, f = 0, yielding:

$$F = P_\infty A_\infty \gamma_\infty M_\infty^2 \left[\frac{M_j}{M_\infty} \sqrt{\frac{\gamma_e T_e}{\gamma_\infty T_\infty}} - 1 \right]$$

All real systems will experience total pressure losses which will cause the ratio M_j/M_∞ to be less than unity. In addition, air will vary with temperature. For this preliminary evaluation, these variations have been ignored. It should be noted, however, that neglecting Mach number and temperature variations is optimistic in both instances.

Incorporating the above assumptions:

$$F = P_\infty A_\infty \gamma_\infty M_\infty^2 \left[\sqrt{\frac{T_e}{T_\infty}} - 1 \right]$$

The thrust available on a unit area basis now becomes:

$$F_{avail} = \frac{F}{A_{2-3}} = \frac{P_\infty \gamma_\infty M_\infty^2}{3}\left[\sqrt{\frac{T_e}{T_\infty}} - 1\right]$$

For sustained flight:

$$F_{reqd} = F_{avail}$$

$$C_{D_O} \frac{q}{M_\infty^2} = \frac{P_\infty \gamma_\infty}{3}\left[\sqrt{\frac{T_e}{T_\infty}} - 1\right]$$

Setting $\gamma_\infty = 1.4$:

$$(1) \quad \left[1 + 2.14 \; C_{D_O} \left(\frac{q}{M_\infty^2}\right) \left(\frac{1}{P_\infty}\right)\right]^2 = \frac{T_e}{T_\infty}$$

The product $\left(\frac{q}{M_\infty^2}\right) \frac{1}{P_\infty}$ is a function only of altitude, and has a constant value of approximately 0.7 from sea level through 100,000 feet.

Equation (1) may be simplified then to :

$$(1a) \quad \frac{T_e}{T_\infty} = (1 + 1.5 \; C_{D_O})^2$$

As given, $\frac{T_e}{T_\infty}$ will vary only with Mach number. In a more precise analysis, an altitude effect would also be present, but the Mach number influence would dominate. The required temperature ratio versus Mach number is shown in Figure II-16 following equation (1a). If it seems surprising that the required temperature ratio declines with Mach number, it must be remembered that the analysis assumes a fixed engine geometry.

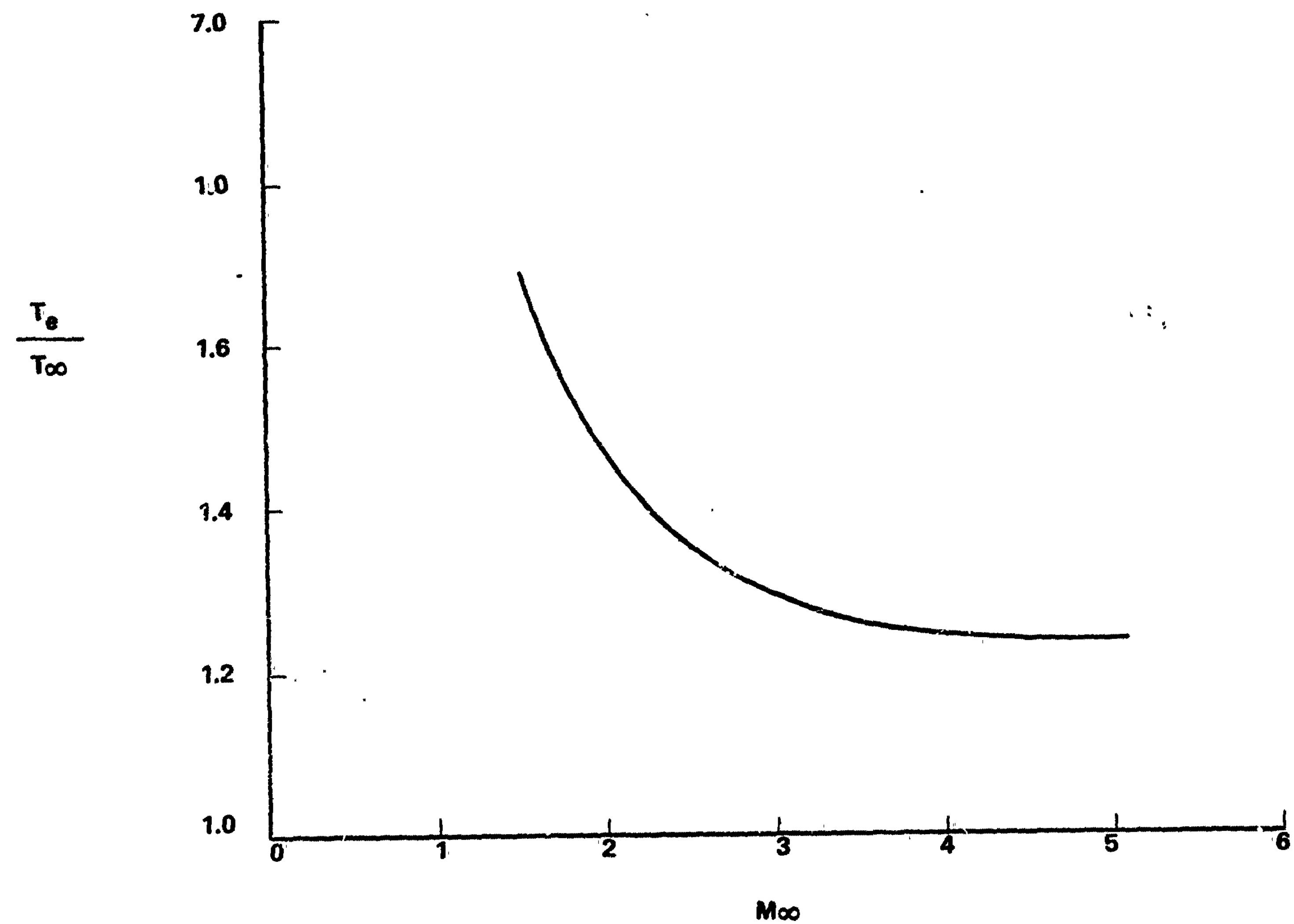

Figure II-16. Ramjet Required Temperature Ratio

At any given condition, mass flow varies as M_∞^2 and the required energy input will begin to rise with increasing M_∞.

Having determined the necessary temperature ratio across the ramjet heat exchanger, it is possible to compute the laser power required for flight at any selected altitude. The analysis assumed a 1.0 square foot ramjet cross section. The heat input required for alternate areas is then the product of the calculated value at 1.0 ft^2 times the actual area (A_{2-3}). The results of typical calculations are plotted in Figure II-17 assuming an air specific heat of 7.0 BTU/lb-mole-$^\circ$R.

Shortcomings: The most apparent shortcoming of the infinite Isp ramjet is the excessive laser power required for operation. For low-altitude operation, it is doubtful that the required power levels can be controlled if indeed they become available. For high-altitude operation, the energy drain on the laser source is still high. It seems unlikely that commitment of laser energy to the task of flying a ramjet can be justified on any logical system basis.

Conclusions: It is concluded that there is no apparent motivation for attempting development of the infinite Isp ramjet at this time. The concept may be technically feasible within 20 years.

Recommendations: It is recommended that no further action be taken on the infinite Isp ramjet at this time. If high-power laser development proceeds favorably, the laser ramjet concept should be reexamined for compatibility with Air Force mission requirements.

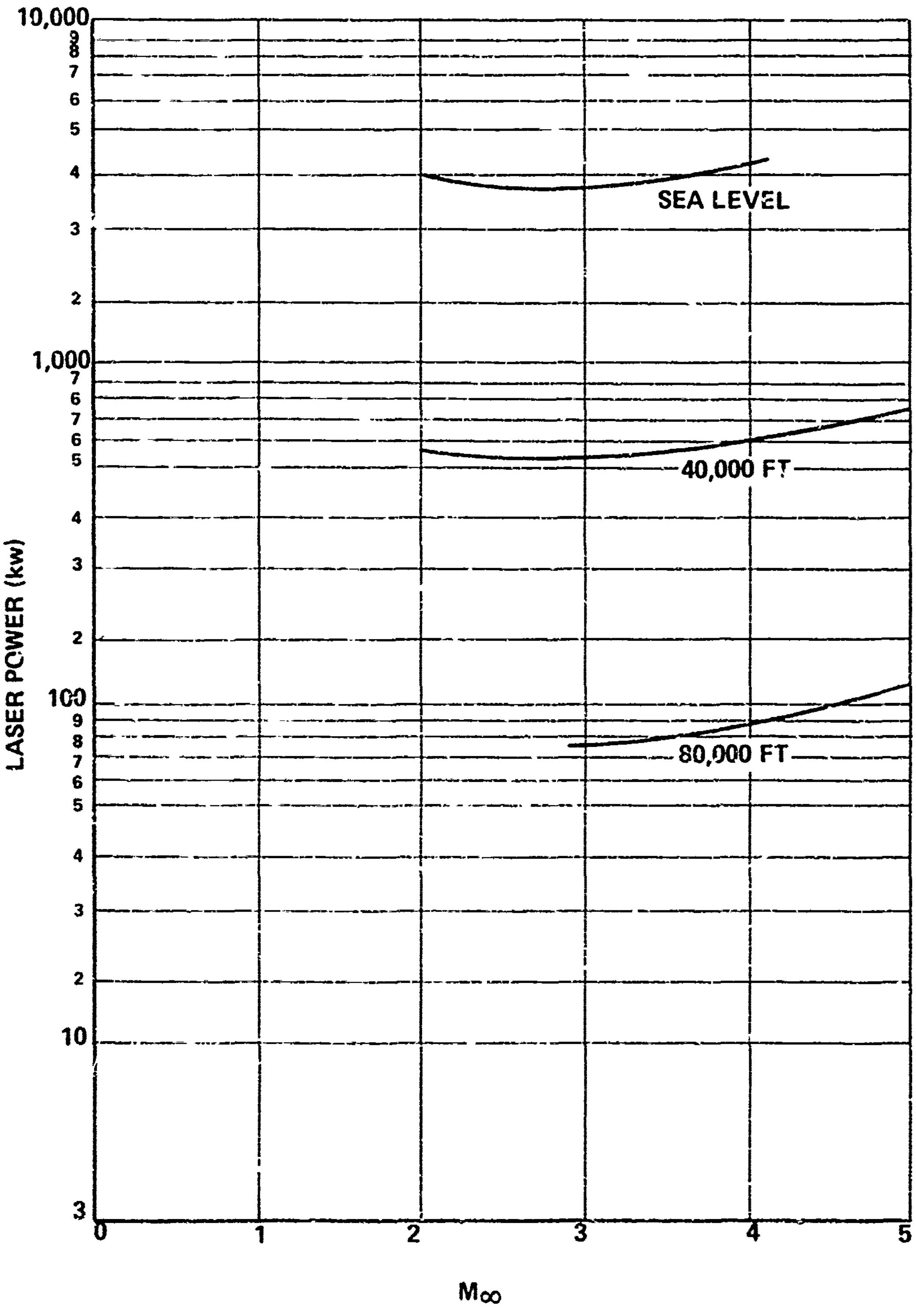

Figure II-17. Required Laser Power

Title: Laser Propulsion

Concept: Energy is beamed to a propulsion system via laser beam from a central location (Ref. II-17). This energy is transferred via a heat exchanger to a working fluid or used directly to heat fluidized particles dispersed in a working fluid. The working fluid, such as hydrogen or ammonia, is used to produce thrust in a conventional manner. The concept is analogous to nuclear rocket technology.

Attributes: Performance levels equivalent to that of nuclear fission rockets are achievable without the associated costs and radiation hazards. The central power station may be earth or space based and the propulsion system may be located in either environment. Since the propulsion system need only carry the working fluid, the added mass fraction normally required to carry an oxidizer or power source is saved. A special case of this concept offers infinite Isp. This is the ramjet, where air is the working fluid, and energy is transmitted to the ramjet via laser beam.

Analysis: The performance of the propulsion system is dependent on the chamber temperature, working fluid, pressure ratio P_c/P_a, and the inert weight of the system. The thrust level will be governed by the maximum energy transfer capabilities of the laser beam and the heat exchanger. Figure II-18 gives the Isp vs chamber temperature for a hydrogen-fueled system operating at a chamber pressure of 1000 psia and 100/1 expansion ratio into vacuum. The maximum Isp depends on the temperature to which the chamber gas can be heated at a given working fluid flow rate. The Isp is, therefore, ultimately governed by the temperature at which the heat exchanger can be operated. Using a heat pipe array to collect and transfer the laser energy, the upper limit would be around 1800°K corresponding to a specific impulse of 734 seconds with today's state of the art in heat pipes. If one uses the colloidal core or "light bulb" concepts from the nuclear rocket industry, the maximum temperature will be limited by the properties of the colloid or the heat transfer mechanisms

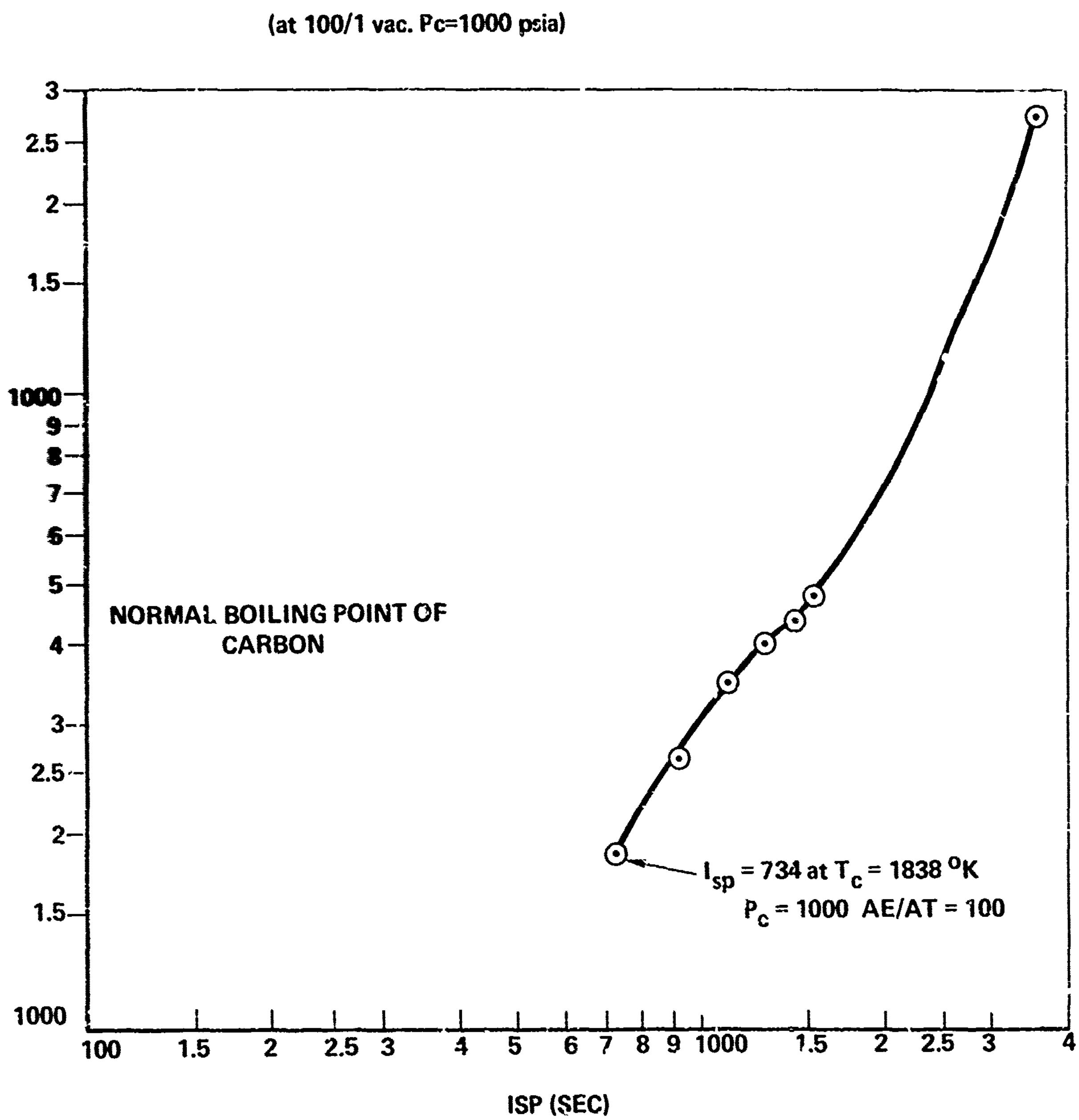

Figure II-18. Performance of Hydrogen as a Function of Chamber Temperature to the fused silica wall. Assuming a colloidal dispersion of carbon in a hydrogen vortex, one might operate at something above carbon's normal boiling point (a chamber pressure of 68 atmospheres). This would give rise to a chamber temperature of around 4500°K with an attendant Isp of about 1500 seconds according to Figure II-18. Possible configurations for the heat pipe and colloidal core concepts are illustrated in Figure II-19. Additional analytical questions involve laser beam divergence, atmospheric diffraction and absorption, spot diameter at altitude, weight of heat exchanger, pointing and tracking accuracy, and laser power and efficiencies to be expected at the central station. The necessary acquisition and tracking technologies are already being developed by the US Air Force, and atmospheric effects can be minimized by locating ground stations in arid regions (Ref. II-18). The laser spot diameter vs distance for various wavelengths is shown in Figures II-20 through 22. The theoretical divergence is given by the equation $\alpha = 1.22\lambda/D_o$. α is the half-angle of the diffraction-limited laser beam. λ is the wavelength of the laser beam, and D_o is the diameter of the objective lens or mirror by which the laser beam is trained on the target. The spot diameter D_2 at an altitude A is given by the equation $D_s = 2A \tan\alpha$.

Figure II-20 indicates that ranges of 200 KM could easily use a collector diameter of one meter if 1 micron laser light is used. The collector diameter of about 4 meters would be better for a 10.6 micron laser beam at the same range. Ranges of 2000 KM look feasible for 1 micron laser light, yielding collector diameters of less than 5 meters (see Figure II-21). Operation beyond this range results in reasonable collector diameters even at 1 micron (see Figure II-22). The 200 to 2000 KM ranges also call for reasonable objective telescope mirror requirements of 0.5 to 3 meters. The Palomar diameter is about 5 meters.

The conversion efficiency of power to laser light is reported at 18 percent for some CO_2 lasers. This effiency should be no problem for

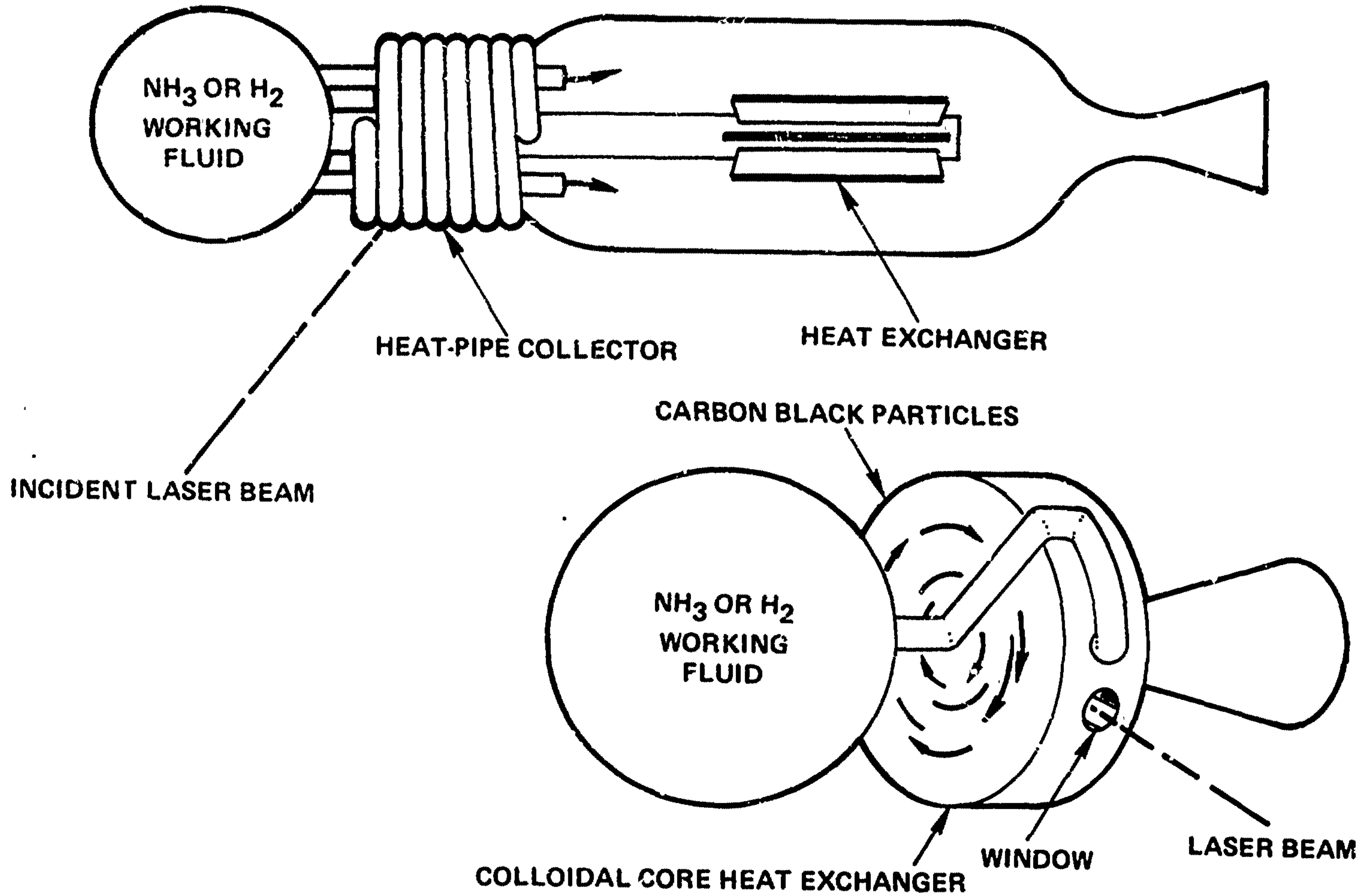

Figure II-19.Laser Propulsion

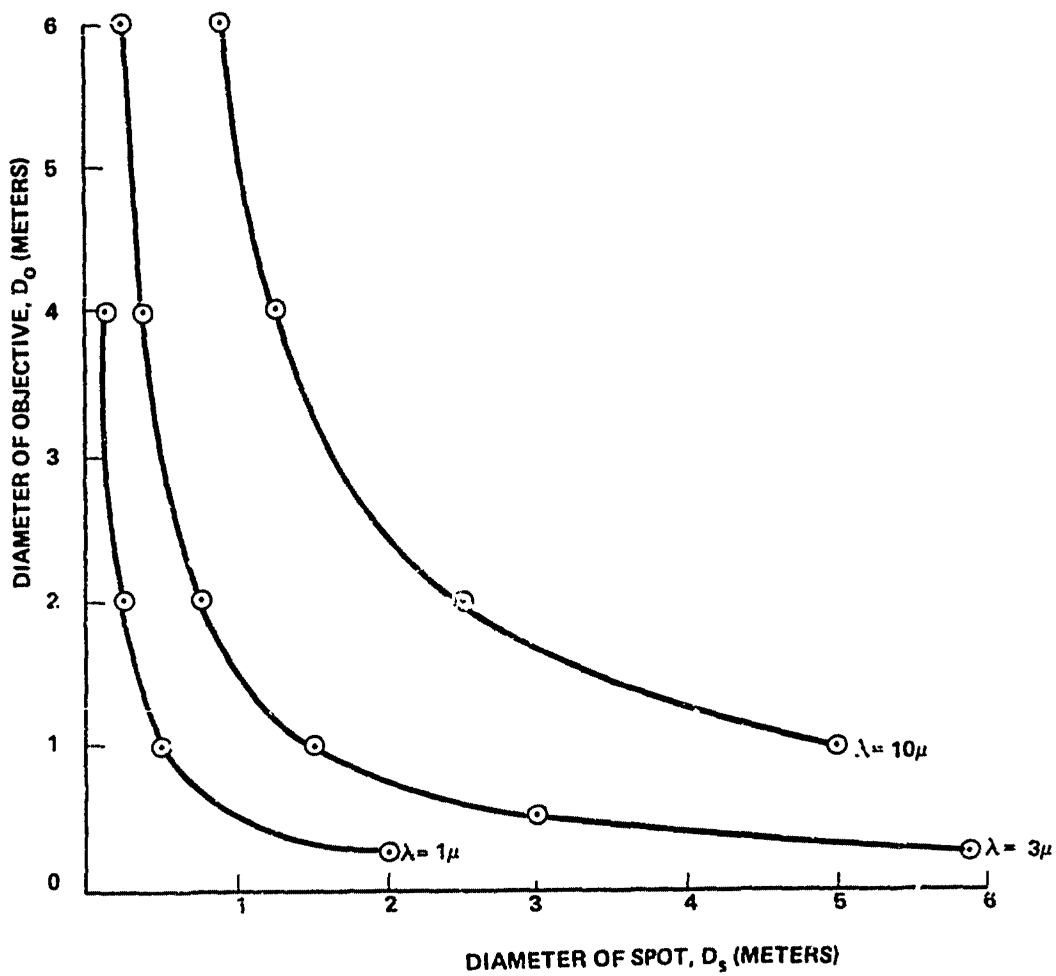

Figure II-20. Diameter of Spot Versus Tracking Lens Diameter for Various Wavelengths of Laser Light at 200 km Distance

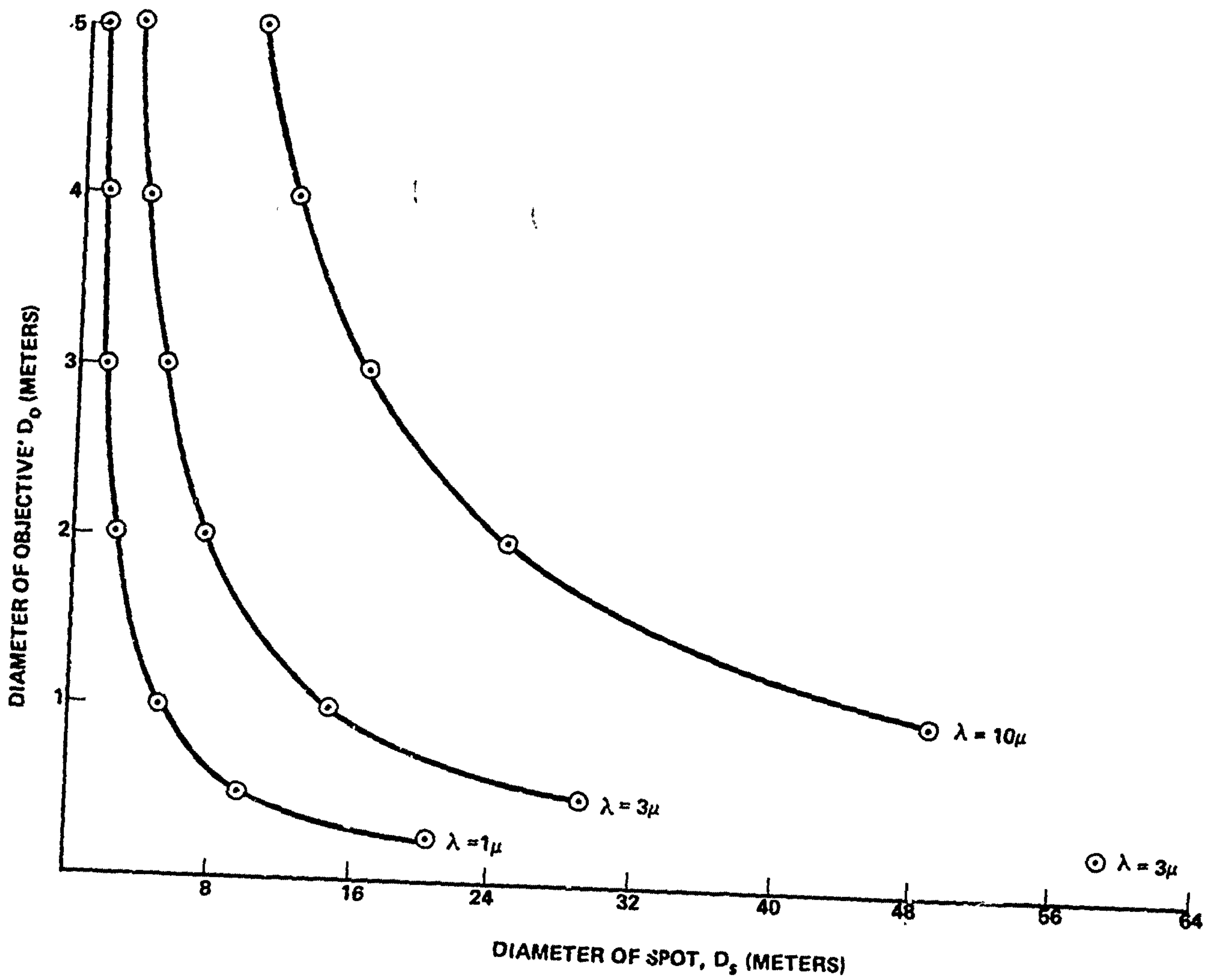

Figure II-21. Diameter of Spot Versus Tracking Lens Diameter for Various Wavelengths of Laser Light at 2000 km Distance

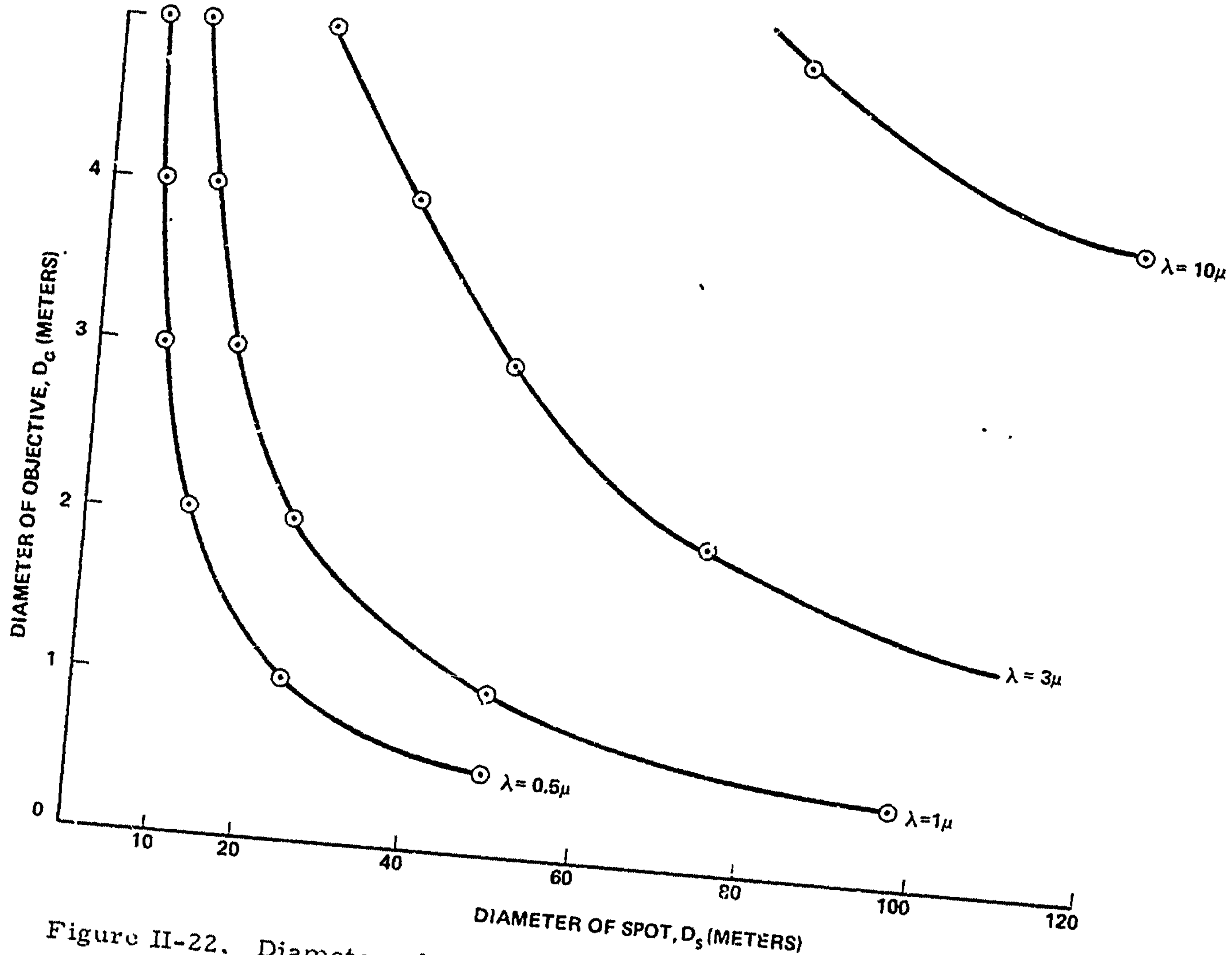

Figure II-22. Diameter of Spot Versus Tracking Lens Diameter for Various Wavelengths of Laser Light at 20,000 km Distance ground-based transmitters. (See Ref. II-19 through II-23 for related work.) The relationship between power delivered to the working fluid of the propulsion system and achievable thrust levels can be examined. A hydrogen working fluid system operating at a chamber pressure of 1000 psi with a 100:1 nozzle expansion to vacuum is used for baseline estimates. Figure II-23 indicates the power in kilowatts to produce 1 pound of thrust as a function of assumed Isp for the baseline system. The estimated limits of the heat pipe concept (Isp = 750 seconds) and colloidal core concept (Isp = 1500 seconds) are also shown. The power requirements for these two concepts are shown in Figure II-24 for various thrust levels. It is apparent that 10,000 pounds of thrust demands power level far in excess of the highest energy, continuous wave lasers reported to date. The power requirement is reduced as the Isp is reduced. The penalty in this case is, of course, that of carrying larger weights of working fluid and the attendant inert tankage weights. These considerations suggest that the proposed systems will be limited to low (1 to 1000 pounds) thrust level applications. Estimates of thrust to weight have not been made. Such estimates will, of course, require parametric studies of heat exchanger weight vs distance from source and operating temperature. Tankage and nozzle weights will be equivalent to conventional propulsion systems operating under these conditions. Beam transmission, collection and heat transfer efficiencies will await experimental data. This data will be needed on beam quality, atmospheric effects on propagation, energy distribution in the beam at the collector and heat transfer characteristics of the collector and propulsion system.

Several additional systems should be considered but will not be analyzed in detail. One is space to space transmission, permitting more optimum selection of wavelengths for beam divergence. Systems might utilize the beam for illumination of solar cells that provide power for electric propulsion systems. This concept appears attractive since

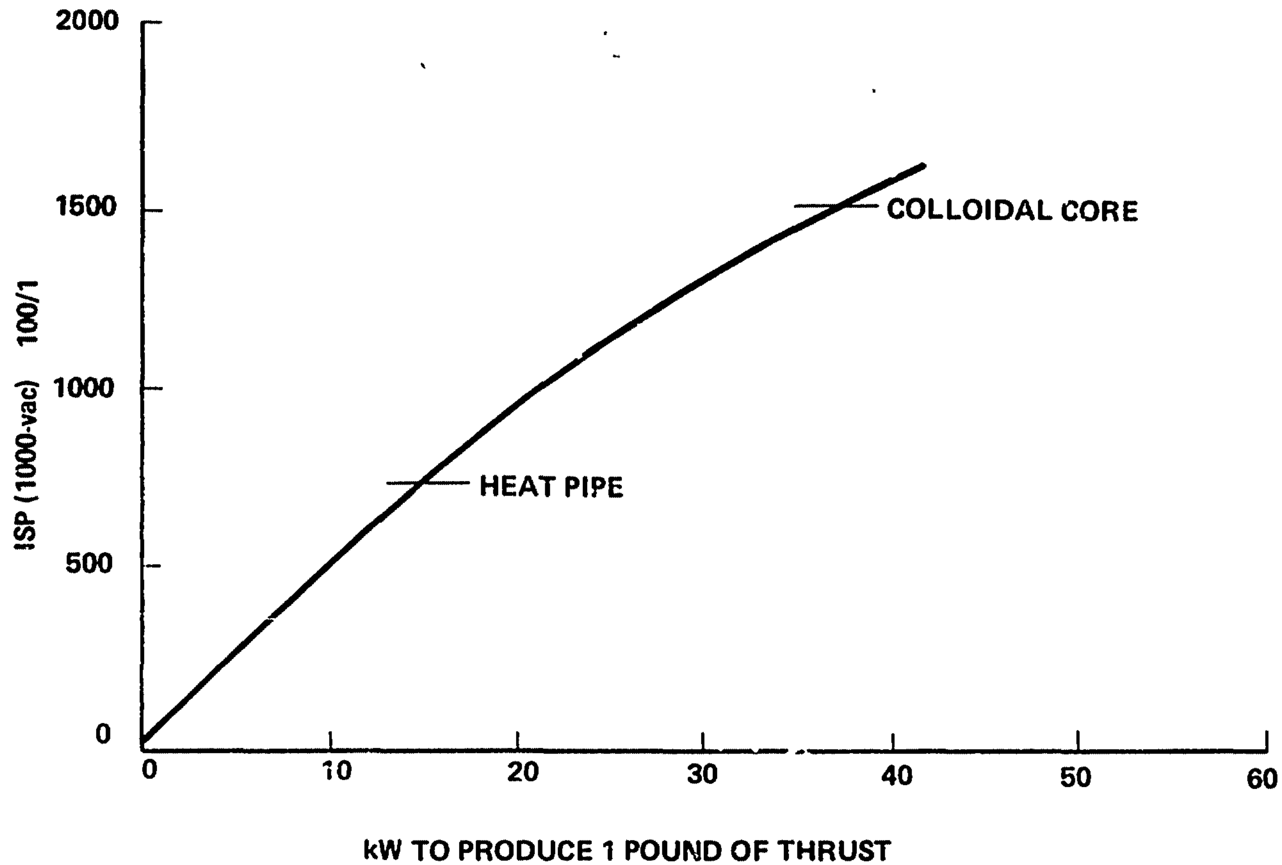

Figure II-23. Isp Versus Power Requirement for H_2 to Produce 1 Pound of Thrust 1000-vac at 100 Percent

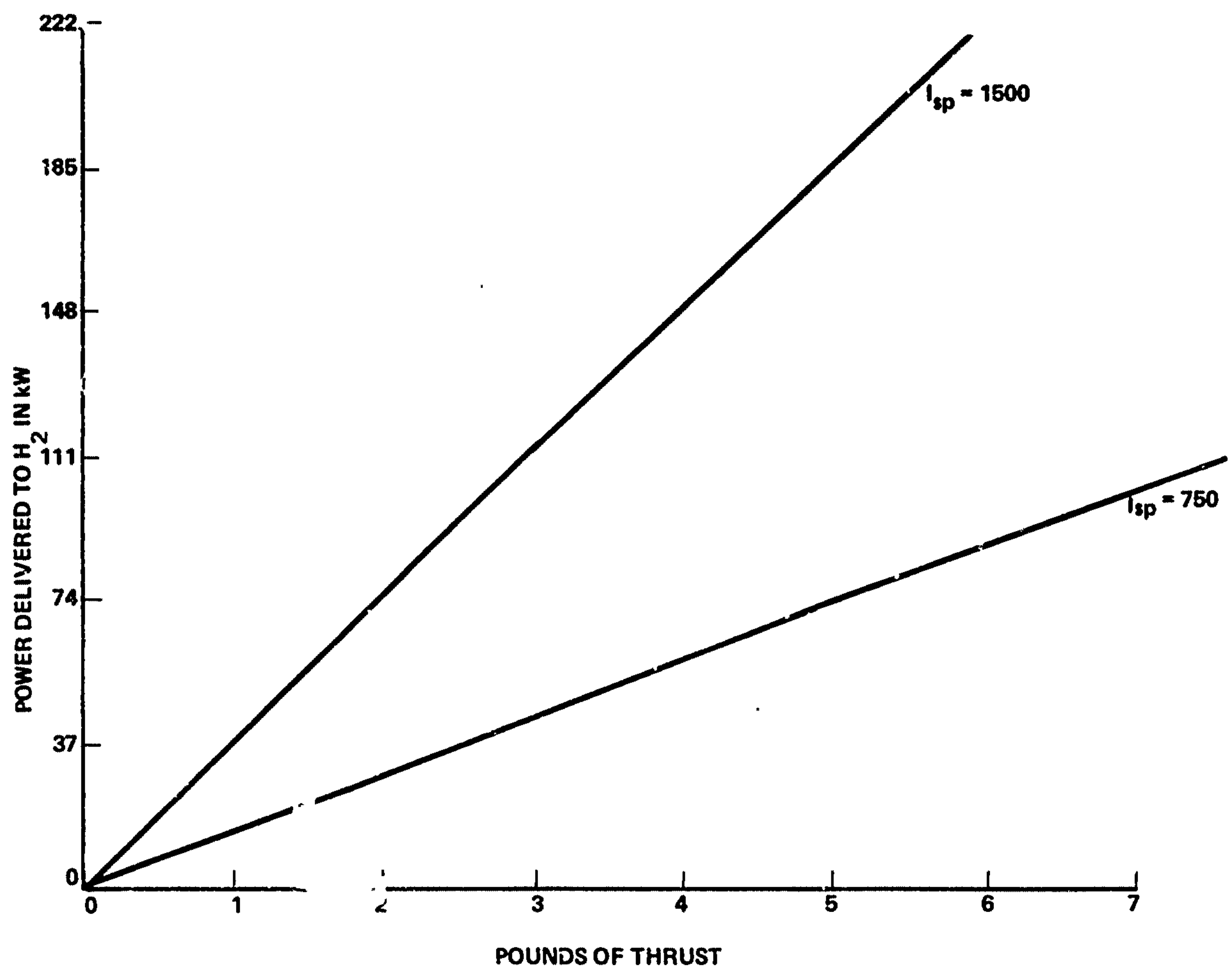

Figure II-24. Power Delivered at 1000 psi Chamber versus Thrust (100/1 Expansion Ratio Vac) solar panels operate well at typical high-powered laser wavelengths. Such a system coupled with electric thrusters would negate the use of normally heavy power supplies and improve thrust-to-weight ratios. The use of laser beams should be considered as a means to pyrolyze or detonate materials on or near an illuminated vehicle to produce thrust. Beam receivers similar to the lasing media have been considered as a potential heat transfer device on the target vehicle. For example, a tube or chamber filled with CO_2 would be an ideal medium to receive the 10.6-micron beam generated by a CO_2 laser. And finally, the recently reported particle levitation from laser beam pressure suggests the use of lasers to accelerate ultrafine particles or gases to extremely high velocities.

Shortcomings: The analysis is based on the most favorable assumptions regarding power transmission by laser beam. Atmospheric effects on ground-based transmitters are uncertain. To minimize atmospheric attenuation and scattering, earth-based laser stations must be installed at high altitude in arid regions. Specific propulsion system designs will have to be compared to nuclear rocket systems to clearly identify mass fraction advantages between the two systems. The laser systems will probably have to operate within 2000 KM of the laser base.

Conclusions: This propulsion system is feasible and will become more feasible with the current rapid development of high-powered lasers and related optics, pointing, and tracking systems. It should be competitive with or superior to nuclear fission rockets.

Recommendations: Make specific mission studies to pinpoint advantages over nuclear fission systems. Demonstrate direct heat exchange from laser beam to colloidal core heat exchanger and measure heat transfer coefficient and maximum achievable temperatures.

Title: Molecular Mole Hill

Concept: This concept "beams" molecules in a preferential direction to provide impulse thrust.

Attributes: This propulsion device features no moving parts, no exhaust nozzle, and no combustion at all. Only the controlled direction of natural randomness of gas molecules is involved. With this propulsion device's capability for operation over long durations at low thrust levels, this concept is extremely attractive for satellite stationkeeping and attitude control.

Description: A container of neutral gas molecules is used. The molecules are allowed to escape through small openings in the container wall to ambient vacuum conditions. A preferred direction of escape is stipulated. The particles efflux from the container (providing a mass flow rate) at a calculable root-mean-square speed. Knowing the state variables of the molecular gas and the number of efflux passages, both thrust and specific impulse can be estimated for a prescribed vessel.

Analysis: Consider a container of gas molecules with X number of efflux ports aligned such that all particles escape in a preferential direction. Let A be the area of each efflux port. Molecules effluxing from each port will have a rms speed v_{rms} given by

$$v_{rms} = \frac{4kT}{m}^{1/2} \tag{1}$$

where k is Boltzmann's constant, T the absolute temperature of the gas molecules, and m the mass of gas molecules. It is noted that v_{rms} of the escaping molecules is greater than the rms speed of the molecules inside the container, since the distribution of effluxing molecules is not the Maxwellian distribution of the molecules inside the container.

Now the rate of efflux of molecules from each efflux port is $n \bar{v}/4$, where n is the number density of gas molecules inside the container and $\bar{v}$ the average speed of molecules. Thus, it follows that the mass efflux $\dot{m}$ from each port is given by

$$\dot{m} = \frac{mNA\bar{v}}{4V} \qquad (2)$$

where N is the number of gas molecules and V the container volume. Equations (1) and (2) are related since $v_{rms} = 1.25\,\bar{v}$.

In steady case, the specific impulse I_{sp} per efflux port is given by:*

$$I_{sp} = \frac{F}{g\dot{m}} = \frac{1}{g}\left(v_{rms} + \frac{AP}{3\dot{m}}\right) \qquad (3)$$

where F is the thrust, P the kinetic pressure of gas molecules inside the container, and g the gravitational constant. However, it is noted that as molecules continually efflux from the container, the container kinetic pressure will decay. Thus, Equation (3) only approximates the situation. Inserting Equations (1) and (2) into Equation (3) yields, using the ideal gas relationship $P = \frac{NkT}{V}$,

$$I_{sp} = 1.42\,\frac{v_{rms}}{g} \qquad (4)$$

*The latter term in equation (3) follows by calculating the number density of molecules in the exhaust beam, knowing the distribution function in the exhaust beam, and then the kinetic pressure in the exhaust beam in terms of this number density and the average translational energy of molecules effluxing the container. The result is: $P_e = P/3$ where P_e is the exhaust pressure.

Now consider a quantitive estimation of the performance of a molecular beam propulsion system. Using a low molecular weight gas, say hydrogen, contained in a 1 m^3 volume at standard conditions. It is assumed that 10^6 efflux ports, each of area 10^{-12} cm^2, are provided over a 10 cm section of the container wall. The developed thrust and specific impulse from such a system, accounting for all efflux ports, follow immediately:

$$F = X F_i \; 10^{-7} \text{ lbs} \tag{5-a}$$

$$I_{sp} = \frac{XF_i}{g X \dot{m}_i} = 320 \text{ sec} \tag{5-b}$$

Representative System Wt 1 lb

$$F/Wt = 10^{-7} \tag{5-c}$$

where the i-th subscript denotes an arbitrary efflux port. From Equation 5, it is seen that a typical molecular beam propulsion system delivers a respectable specific impulse; however, the thrust developed is extremely small.

The time t for the kinetic pressure inside the container to decay to the e-fold of its initial value is given by:*

$$t = \frac{4V}{XA\bar{v}} \tag{6}$$

For the system described here, t is about 10^7 seconds. Thus, the steady approximations made are considered valid. However, if a smaller

*Equation (6) neglects effusion from the exhaust beam back into the container. Including back effusion merely increases the numerical constant in Equation (6).

container is used, say 1 cm^3, there results an e-fold pressure decay time of approximately 10 seconds. Here, the steady approximations become marginal.

Contrasting the molecular beam propulsive device, it is seen that this system appears best suited for missions requiring extremely low thrust. Stationkeeping missions would be obvious examples. However, calculations indicate that the sample molecule beam propulsion system described here can provide the necessary propulsion to perform a ΔV correction of 1 ft/sec-mo for a small 25-pound satellite located in synchronous equatorial orbit. This degree of correction could be used to compensate for anticipated east-west translational drift. This concept could not, however, perform anticipated north-south drift corrections. Also, for the particular system configuration described here, it is noted that the kinetic pressure inside the container will e-fold aft - 2 to 3 months of continuous operation. This then requires that the container be repressurized for further operation. This is not necessarily considered a severe penalty. If, for example, the container gas were hydrogen stored in its solid state for repressurization and then gasified to one atmosphere pressure, only a storage volume of 1 ft^3 would be necessary to replenish the container every 3 months for 7 years. The total weight of hydrogen used would be 5.5 pounds.

Shortcomings: The fundamental weakness of the molecular beam propulsive device is its inherent low thrust capability and attendant small payload maneuvering capacity. One further shortcoming of the molecular beam propulsion device is noteworthy. As the number of efflux ports X increases, ultimately the effluxing velocity reverts from a suitably averaged microscopic speed to a grand or macroscopic velocity. The molecular beam then becomes an equivalent cold-gas propulsion system. A measure of the limit where molecules no longer have physical individuality and thus where continuum effects dominate is given by the relationship between molecular mean free path λ and some characteristic dimension d of the system. d is taken to be the efflux port diameter. Thus,

$$d = \lambda \sim \frac{1}{P} \qquad (7)$$

defines the upper limit of the microscopic approach, and as d increases (or as X increases) the container pressure must be decreased.

Conclusions: The molecular mole hill concept offers moderate specific impulse but very low thrust. Theoretically the concept appears feasible, but the details needed to define an actual system are lacking.

Recommendations: The low thrust levels and small payload maneuvering capacity anticipated with this propulsive device preclude an active interest in this system.

CHAPTER II-2. FIELD PROPULSION

Field propulsion concepts use electric and/or magnetic fields to accelerate an ionized working fluid, or react directly with the environment by electric or magnetic effects. It is in the area of field propulsion that the most revolutionary concepts appear. The ability to perform objective analyses of many of these ideas was diminished because underlying principles transition from the known to the unknown. The category of field propulsion probably contains more ideas than any other concept area. It would be impossible within the time constraints of this study to evaluate the field propulsion area completely, so emphasis has been placed on the more conventional ideas. More radical concepts may be found in the open literature by those interested in pursuing them.

Title: Electromagnetic Thrusters

Concept: In an electromagnetic thruster (plasma engine), thrust is produced by the acceleration of an ionized gas with magnetic fields.

Attributes: When compared to other high-performance engines, plasma thrusters have certain basic advantages:

1. Because the magnetic field acts on the entire plasma, no space charge is formed; thus, the plasma thruster can produce greater thrust per unit area than electrostatic engines.

2. In contrast to chemical engines, plasma engines offer a relative freedom of expellant choice, fine thrust control and almost unlimited restart capability leading to design flexibility.

3. Plasma devices employ a simplified operational concept in comparison with other electrical systems.

4. Plasma devices operate at high specific impulse.

5. When operated in the pulsed mode, plasma devices are ideal for extended orbital control of spinning satellites.

6. Plasma devices can be operated in the atmosphere, a feature which reduces system test costs in relation to other electrical systems.

Description: In a plasma propulsion device, a body of ionized gas is accelerated by the interaction of currents driven through the gas with magnetic fields established either by these currents or by external means.

Physically, there are many ways to establish such interacting currents and magnetic fields in an ionized flow. One basic distinction which may be made is between steady and pulsed systems. In the steady systems, the current density pattern in the gas, the magnetic field, the flow velocity, and the thermodynamic properties of the gas remain constant in time at every point, whereas in pulsed systems, these elements undergo vigorous pulsations in time. Steady systems may be further subclassified as to those which use an externally applied magnetic field and those in which the fields generated by the current patterns in the gas and its driving circuit accelerate the plasma.

Pulsed acceleration systems (Figures II-25 and II-26) subdivide into the series-coupled mode in which the discharge current passes directly through the gas between electrodes in contact with the gas, and inductively coupled modes, in which currents are induced in the gas in response to primary current pulses or oscillations flowing in a circuit entirely external to the gas.

A third type of electromagnetic thruster, the traveling wave accelerator, uses an external array of programmed currents to generate a continuous electromagnetic wave which propagates through the ionized gas, sweeping it along as the wave train interacts with the currents it induces in the gas.

Analysis: For each of the plasma acceleration concepts there are a variety of practical alternatives for electrode, channel, and field geometries as well as the gas type. Operational details such as insulation, injection, and switching also require consideration. Table II-4 indicates some of the possible modes of electromagnetic acceleration which have been studied for possible propulsion application.

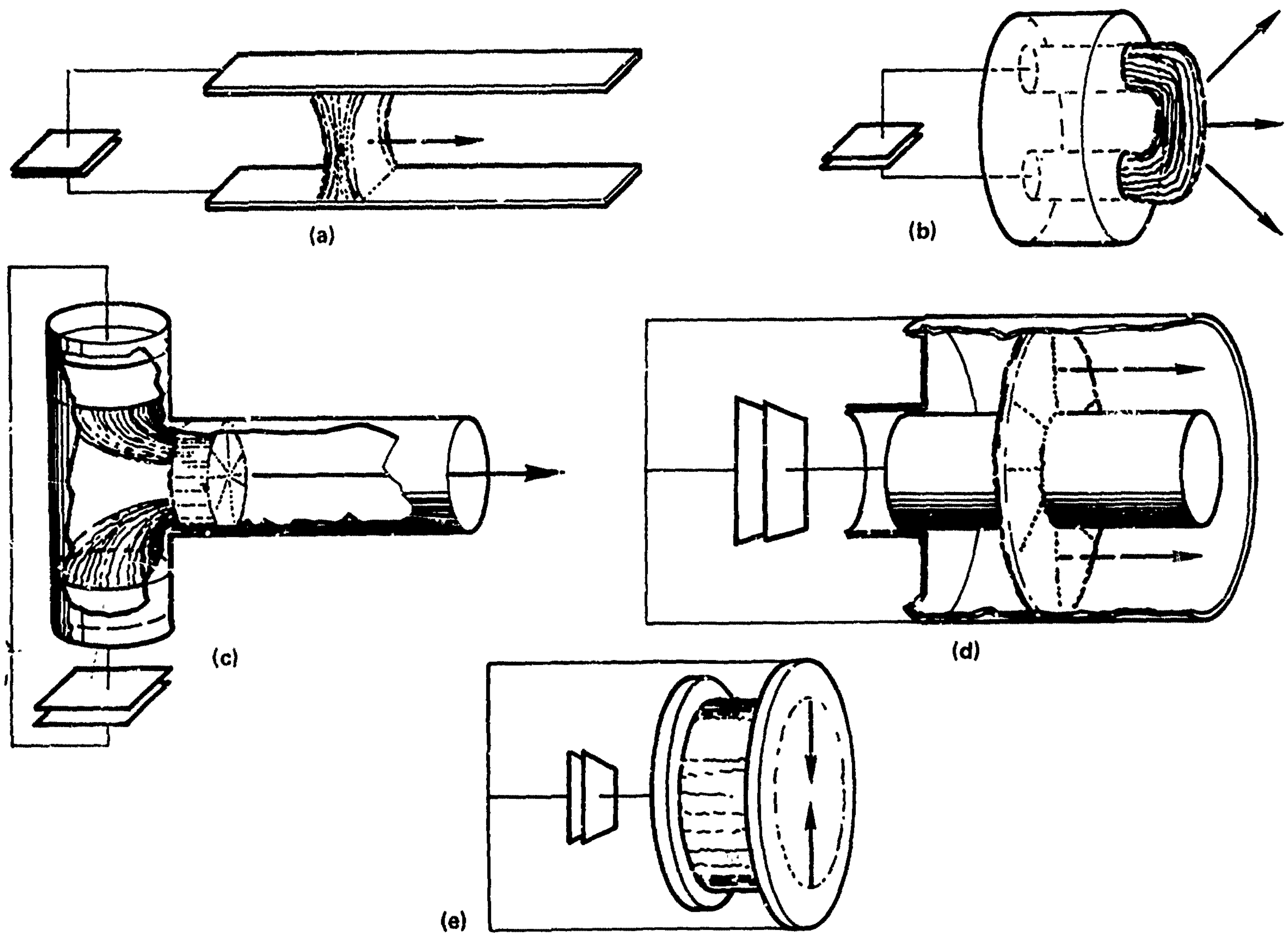

Figure II-25. Various Pulsed Plasma Accelerators (Schematic).
(a) Parallel-rail accelerator; (b) button gun; (c) T tube; (d) coaxial gun; (e) linear pinch.

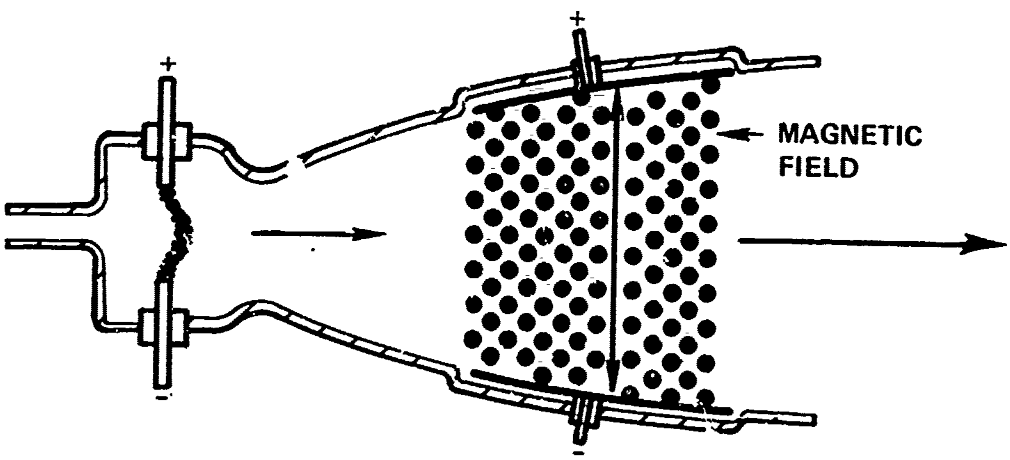

Figure II-26. Lorentz Force or j x B Plasma Propulsion Engine

TABLE II-4. CLASSIFICATION OF ELECTROMAGNETIC ACCELERATORS

Time Scale of Interaction	Steady		Pulsed		Traveling Wave
Source of magnetic field	External coils or magnets	Self-induced	Self-induced		Coil sequence on transmission line
Ionization	External	Internal	Internal		External or internal
Primary current source	Direct current supply		Capacitor bank		Radio-frequency supply
Discharge coupling to circuit	Direct		Direct	Induc-tive	Inductive
Working fluid	Pore or seeded gas		Pore gas; vapor-ized liquid or solid	Pore gas	Pore gas
Channel geometry	Rectangular or coaxial; constant or variable cross section		Coaxial pinch, parallel rail, ablating plug	Theta pinch, conical pinch, loop inductor	Rectangular, cylindrical, coaxial; constant or variable cross section
Other distinguishing features	Lorentz or Hall mode		Internal or external switch		Constant or variable phase velocity

To illustrate the electromagnetic acceleration concept in its simplest form, consider a flow of ionized gas which is subjected to an electric field $\overline{E}$ and a magnetic field $\overline{B}$, perpendicular to each other and to the gas velocity $\overline{u}$. If the gas has a scalar conductivity σ, a current of density $\overline{j} = \sigma(\overline{E} + \overline{u} x \overline{B})$ will flow through it. This current is parallel to $\overline{E}$, and will interact with the magnetic field to provide a distributed body force density $\overline{F}_B = \overline{j} X \overline{B}$ which will accelerate the gas in the direction of $\overline{u}$.

There are factors which limit the effective operation of plasma accelerators. The conductivity of the working fluid is prescribed by its composition, density, and temperature; but the latter two are limited by the tolerable heat transfer to the thrust channel walls. There are also limits to the current density which can be produced by electrodes at given temperatures and local electric field strengths.

The effective electric field $\overline{E}$ must not be so high that the desired uniform discharge breaks down into discrete arc columns. The critical value of this field is a strong function of gas density and composition, and possibly of electrode surface conditions. The strength of the applied magnetic field is limited by the size and type of magnet which can reasonably be carried in the thruster package. Finally, the gas density cannot become so low that cross-field or Hall conduction dominates the conductivity. This in turn depends on the applied magnetic field $\overline{B}$.

Based on analysis and considering factors of this type discussed above, several specific plasma thrusters under development look promising. Experiments in the 1960's showed that arc currents as high as 3000 amperes could be drawn across the electrodes of an electrothermal device without serious erosion. This was accomplished by drastically reducing the propellant flow, and thus the pressure in the arc chamber. Under these conditions, the exhaust velocity of the propellant could be increased to values in the order of 10,000 m/sec with an overall efficiency of 50 percent. Work on these magnetoplasmadynamic (MPD) devices has proceeded for some time at NASA's Lewis Research Center. These efforts have been aimed at understanding the thrust mechanism in order to produce a reliable, efficient MPD device which operates at about 30 KW in steady-state. Such a device could have a specific impulse of 10,000 seconds while operating at 45 percent efficiency. If MPD devices cannot be made to perform satisfactorily in steady-state, it is possible that a quasi-steady-state engine may produce better results. This may overcome some of the inherent heat transfer problems of the steady-state device.

In addition to the MPD thruster, pulsed plasma acceleration devices draw considerable interest. Currently designed to operate at about 170 watts, pulsed plasma engines offer a potential specific impulse of about 1500 seconds and thrust in the low millipound region. A device using a solid rod of teflon which is ionized for use as a propellant has already successfully flown on an LES series satellite. Currently NASA's Langley Research Center is studying methods to improve the efficiency of this engine. As follow-on concepts, both liquid and gaseous pulsed plasma engines are being considered. The two primary problems associated with pulsed plasma thrusters involve the operation of the devices themselves. First, there is the problem of low operational efficiency. Secondly, it is felt that there is a possibility of electromagnetic interference with other systems aboard the spacecraft. Both of these areas will require a considerable amount of further work.

Although MPD thrusters and pulsed plasma engines are the two primary plasma propulsion concepts currently under consideration, there are many other schemes which offer similarly high performance. Funding limitations, engineering problems and lack of large power supplies have limited the scope of the current effort in plasma propulsion.

Since plasma engines require an on-board power supply, they are especially attractive for spacecraft carrying large amounts of on-board power, such as nuclear reactors.

Because of their high specific impulse and relatively low weight, delicacy of thrust level control over a wide range, and essentially unlimited restart capability, plasma thrusters are extremely attractive for attitude control, stationkeeping, orbit adjustment of long-lived earth satellites, or cargo "ferry" missions between earth and lunar orbits. For long-range planetary missions or other extremely high total impulse missions, plasma thrusters must also be seriously considered.

The final decision as to what type of propulsion system is the most desirable depends on the characteristics of both the thruster and the power supply. Today, solar arrays provide most of the long-term spacecraft power. The primary limitation here is one of array area, and hence weight and ease of utilization. As the specific power of nuclear reactors is increased, these energy sources will become more attractive, as will the use of high-energy plasma thrusters. Presently, Rankine cycle nuclear reactors for space power applications are being designed toward systems between 10 and 100 KWe with specific power of about 10 watts per pound. Larger, K-Rankine cycle reactor systems such as SNAP 50, at 3MWe, may deliver 80 watts per pound. Finally, in-pile thermionic devices, such as are being designed at the Los Alamos Scientific Laboratory, may offer 1 to 10 MWe at over 100 watts per pound. The advent of fusion power supplies for space applications will most probably greatly improve on these figures. Thus, the future of plasma propulsion is intimately connected to power supply development as well as mission requirements.

Shortcomings: There are three basic reasons why plasma engines have not been extensively developed and applied to space missions:

1. No mission has been identified which absolutely requires plasma propulsion for its accomplishment.

2. The specific mass of currently available power supplies is too high to warrant widespread use with a plasma engine.

3. Plasma engines have not demonstrated the high operating efficiencies attained with colloid and ion engines.

Conclusions: Plasma thrusters have been the subject of research and development efforts for many years. Their development would lead to systems having high performance (10,000 > Isp > 1000 seconds) at thrust levels dictated by the size of the power supply on board. Compared to other electric systems, plasma thrusters offer more compactness and longer operating life, along with choice of expellant. For space missions with large amounts of available power, plasma thrusters appear to be the optimum propulsion means for a wide range of applications.

However, understanding of basic plasma phenomena is weak, and present developmental efforts favor the experimental approach. More work is necessary before these propulsion devices can be properly designed for widespread application.

Finally, a propulsion system based on plasma acceleration is strongly dependent on the characteristics of the power supply. Total system weights, lifetimes, and efficiencies must be considered along with performance.

Recommendations: Plasma systems appear very attractive for space missions of long duration. Current effort should be directed toward greater understanding of the basic principles. However, continued developmental effort should not be neglected. This effort should be broad enough in scope to simultaneously include work on pulsed, quasi and steady-state plasma devices.

Title: Electrostatic Thrusters

Concept: Electrostatic thrusters generate thrust by accelerating charged particles by means of an electric field.

Attributes: The principal attribute of electrostatic thrusters is their very high specific impulse. For engines with thrust levels ranging from micropounds to millipounds, the specific impulse may vary from 1000 to 10,000 seconds.

Description: Many electrostatic thruster types exist and can be categorized by the size of particles accelerated - ion or colloid, or by the means of particle ionization - contact or electron bombardment. It is essential for the electrostatic thruster to have a neutralizer which injects a stream of oppositely charged particles into the beam such that the beam leaving the engine has no net charge.

The principal activity in electrostatic thrusters currently is with electron bombardment ion engines and colloid engines. Figure II-27 shows a schematic of an electron bombardment ion engine. Propellant is fed through the cathode where free electrons are formed. The path of the electrons from the cathode to the anode is controlled by the magnet surrounding the ionization chamber. Propellant is introduced into the ionization chamber in a gaseous state. The collisions of electrons with the propellant atoms form electron-ion pairs. The chamber will contain neutrals, electrons, and positive ions. As the latter particles migrate toward the downstream end of the chamber, they come under the influence of a strong electric field established between the downstream electrodes. This field serves to accelerate the particles and eject them from the engine. A plasma bridge neutralizer, similar in design to the cathode, completes the engine. Typically, either mercury or cesium is used as the propellant in such an engine.

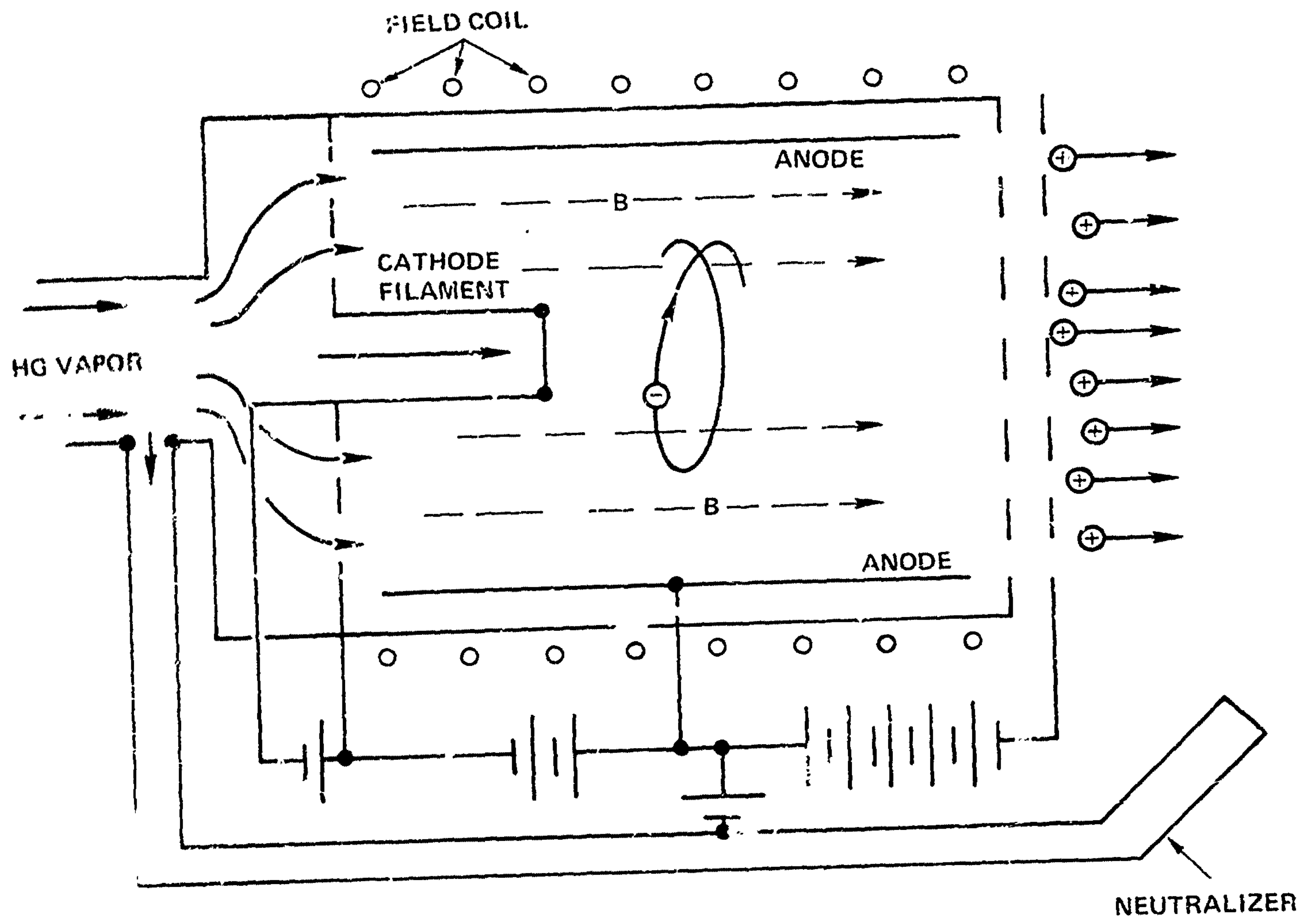

Figure II-27. Electron Bombardment Engine

The colloid thruster is shown schematically in Figure II-28. Propellant is drawn by capillary action to the tips of the needles where it comes under the influence of a strong electric field created between the needle and extractor. The surface of the electrolytic propellant is disrupted by the electric field and submicron sized, electrically charged droplets are torn from it and accelerated downstream, thereby producing thrust. A neutralizer completes the engine. The typical propellant consists of a salt (either NaI or LiI) dissolved in a solvent which is usually glycerol.

Analysis: The area of applicability of electrostatic thrusters appears to be at thrust levels approaching ten millipounds and total impulse values around 10,000 lbf-seconds. Some specific missions that fall in these performance regimes are north-south stationkeeping, maintenance of the line of apsides, and drag makeup. Other missions may present themselves to exploit the peculiar performance characteristics of electrostatic thrusters.

There are several tradeoffs to be considered in integrating such engines into a spacecraft. Generally, military satellites are limited in the amount of power available for propulsion purposes and this will govern the tradeoff analysis. The power required by an electrostatic thruster is directly proportional to both thrust level and specific impulse. Thus, for a given thrust level, the power level required depends upon the specific impulse. The thrust level to do a specific mission is not fixed, however. To perform north-south stationkeeping missions, an engine may have up to 12 hours in which to make the required corrections. By increasing the thrust level, the thrusting time can be decreased proportionately. The spacecraft integrator must evaluate engine lifetime, thrust, specific impulse and power tradeoffs to arrive at a compatible solution. Table II-5 presents some data on electrostatic engine systems of comparable thrust levels.

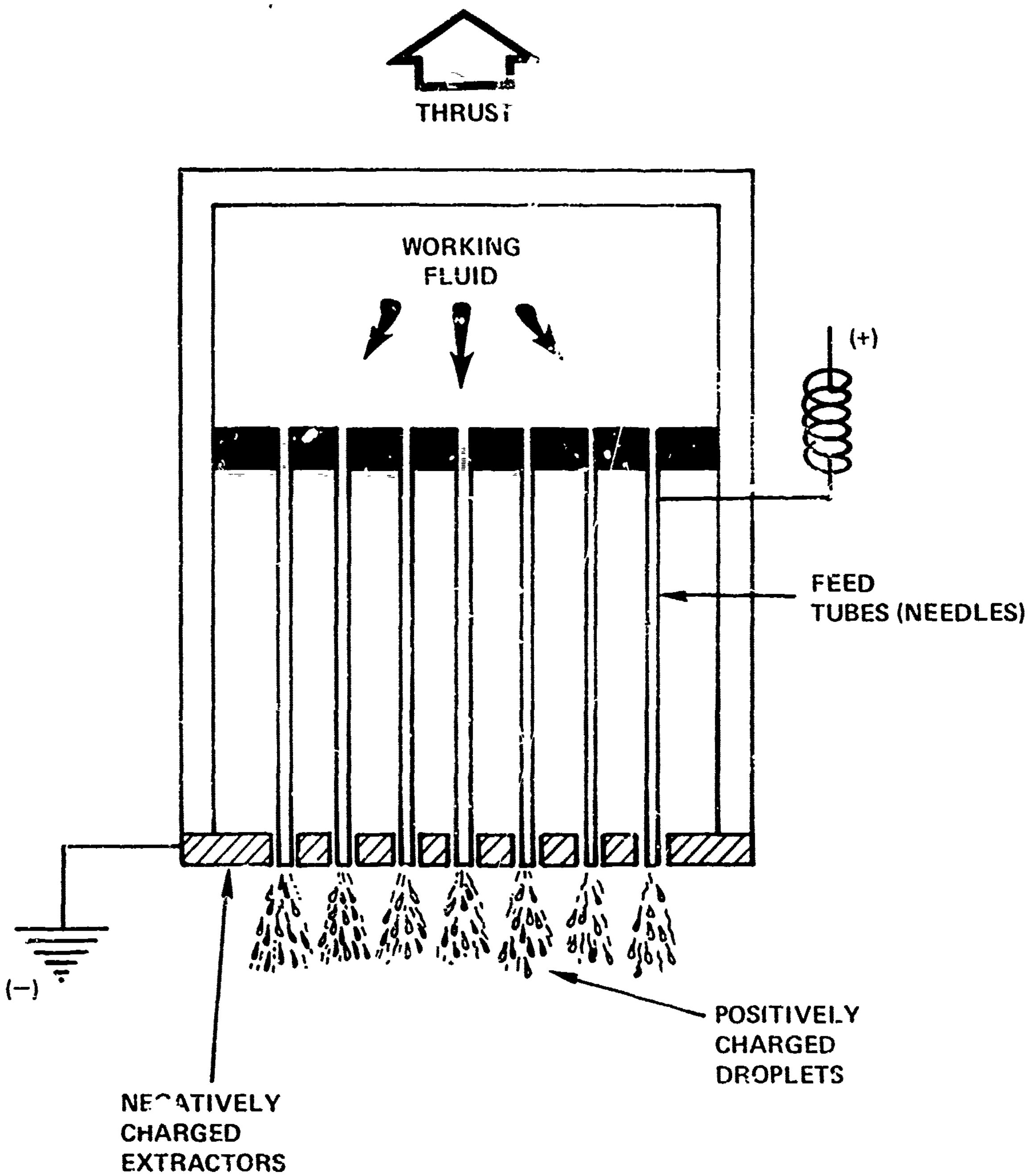

Figure II-28. Colloid Thruster

TABLE II-5. ELECTROSTATIC THRUSTER PERFORMANCE

Engine System	Thrust (mlb)	Isp (sec)	Power/Thrust (watt/mlb)	Efficiency (%)
Colloid	1.0	1500	70	45
Ion (Hg)	0.41	1840	138	29
Ion (Cs)	1.0	2500	145	38

The power to thrust level is important to power-limited satellites where one would prefer the lowest value. The ion engines exhibit much better efficiencies at higher specific impulses.

The electron bombardment ion engine system is well developed using either mercury or cesium. A 6-millipound, 30 cm (in diameter) mercury system was flown on the SERT III satellite and successfully operated for about 4000 hours prior to a malfunction that terminated engine operation. A 5 cm, mercury ion engine is currently under development for the Canadian Communications Technology Satellite to perform the north-south stationkeeping function. A 1 millipound cesium bombardment engine is under development for flight on an Application Technology Satellite (ATS-F).

The mercury system is somewhat simpler than the cesium system. Here, electrical isolation devices, allowing the use of a single tank for cathode, neutralizer, and primary mercury flows, are well developed. The cesium system employs three tanks - one for each function.

The interaction of the engine plume with spacecraft surfaces is a problem under investigation with both the mercury and cesium propellants. Some results to date indicate potential problems that must be considered in integrating either device into a spacecraft. Results from the SERT II flight indicated that very little mercury was deposited on test surfaces; but large quantities of molybdenum, sputtered from the accelerator grids, were deposited in sufficient quantities to be detrimental to solar cells or optical surfaces. This grid erosion problem is attributed to high-velocity neutrals created by charge exchange collisions between high-velocity ions and low-speed neutrals. A reduction in neutral fraction would serve to reduce this problem. The vapor pressure of the metals used in engine construction is so low that they will doubtless condense on any surfaces upon which they impact. The vapor pressure of both mercury and cesium are orders of magnitude above the environmental pressure at typical spacecraft operating temperatures, so it would not be expected that significant deposition of these propellants on spacecraft surfaces would occur. Due to the highly corrosive nature of cesium, the question remains whether even small deposits of this material would cause major problems by chemically attacking the surface. An experimental program to investigate the effects of propellant deposition upon various spacecraft materials is currently under way.

Another aspect of bombardment ion engines that bears consideration is the pulsing mode of operation. Application of electric thrusters to spinning satellites will require pulsing capability to perform certain maneuvers. Pulsed electrostatic vectoring could be employed as a means of generating attitude control forces, but this method does not appear practical for east-west stationkeeping of a spinning satellite.

The colloid thruster is currently in an advanced development status with testing of a one millipound breadboard engine under way. A flight test is tentatively pl.nned aboard a Space Test Program satellite sometime during 1975.

The plume/spacecraft interface problem is currently under investigation for glycerol/NaI propellant. Specifically, it deals with the compatibility ‹ ˙ ¹ spacecraft materials and the propellant. The glycerol solvent has a or pressure which is orders of magnitude greater then the environmental pressure at typical spacecraft operating temperatures. Thus, the deposition of glycerol on sensitive surfaces would not appear to be a problem. On the other hand, the sodium and iodine present in the exhaust plume as ions, elemental species, or in compounds could impose problems when they contact spacecraft surfaces.

Another aspect of colloid thruster design currently under evaluation is the thrust achievable per emitter source. Current engine technology has demonstrated 2.5 micropounds thrust per source, while exploratory investigations are aimed at demonstrating 20 to 30 micropounds per source and 1500 seconds of specific impulse. The immediate objective is to reduce engine complexity and size without reducing performance or life. Special emphasis is placed upon improving efficiency to yield better power to thrust ratios than currently demonstrated.

Electrical pulsing of colloid devices has been successfully accomplished at the single needle level. The potential exists that the device will be applicable to spinning satellites. Additional work with a complete system would appear to be in order for a more thorough evaluation of the technique.

An additional feature of colloid devices is their ability to throttle by trading specific impulse for thrust. This has been demonstrated in limited testing under IR&D.

<u>Shortcomings</u>: The principal shortcomings of the ion engine system with regard to power-limited military satellites is their high power to thrust ratio. The plume/spacecraft interaction is a potential problem that is currently under investigation. The ion engine is not well suited to low thrust operation with specific impulses below 2000 seconds due to mechanical problems of acceleration grid spacing.

The colloid system is quite sensitive to manufacturing tolerances in needle fabrication. This can adversely affect engine efficiency. Performance repeatability of a colloid engine has yet to be demonstrated but is currently being studied. The colloid system is also constrained to specific impulses below 2000 seconds.

Neither the ion nor the colloid system has demonstrated life for a period commensurate with that required by anticipated applications.

Conclusions: The feasibility of both ion and colloid electrostatic thrusters has been demonstrated. The ion engine systems with mercury propellant are the most advanced on the basis of their flight test experience. The cesium ion engine is the next most advanced.

Recommendations: The payoff of colloid systems to power-limited satellites are significant and warrant continued development.

Title: Improved Ion Fuels

Concept: In an electrostatic engine, thrust is produced by the ejection of positive ion accelerated by electric fields. Maximum energy utilization is obtained when all of the ions are identical in mass and velocity. For practical reasons, ion currents (directly proportional to thrust with i velocities being constant) are limited. One avenue for increasing thrust with present electrostatic thruster designs is to find a means of increasing the mass of the ejected ions (increase mass flow) while retaining the properties that make a substance such as cesium a desirable ion fuel.

Attributes: In addition to their attractive physical properties, the proposed ion fuels hold the promise of increasing the thrust of present ion engines up to ten times their current operational limits with only minor modifications.

Description: In an electrostatic thruster, ions are normally g ated by electron bombardment of a suitable fuel, e.g., cesium, mercury, or colloid particles. The ions are then accelerated through a line eries of electric fields and ejected. An electron beam is simultaneously directed in the same direction as the ion beam for space rge neutralization, or in the case of the colloid fuel a beam of negatively charged particles serves to prevent space charge accumulation.

For optimum performance, the charge, mass and velocity of all the ions in the exhaust beam should be identical. Increasing the mass of the ejected ions significantly improves energy utilization at lower specific impulses (assuming no change in ionization potential) because less energy is then required for ionization (which does not produce thrust) per unit mass of propellant. The fuels now used for ion engines have been chosen because of tradeoffs between theoretical, operational, and availability factors.

Cesium is attractive as an ion fuel for its low ionization potential (i. e., 3.8 ev). The low voltages (5 to 8 volts) employed in the electron bombardment process used for generating cesium ions leads to minimal electrode erosion by sputtering. Thus, a cesium engine can be expected to give a long service life. Cesium's low melting point (28.5^{o}C) and moderate volatility (one torr at 279^{o}C) also contribute to making it an attractive ion fuel.

Mercury ions with an atomic weight of approximately 200 are more massive than cesium ions (atomic weight of 133), and mercury ion thrusters provide higher thrust for the same beam currents and ion velocities. Also, the lower melting point of mercury (-40^{o}C) permits its use under a less-controlled thermal environment than cesium. Unfortunately, mercury has the disadvantage of higher ionization energy per unit mass of material than for cesium.

Glycerol doped with sodium iodide comprises a fuel for a colloid engine. This solution has properties that permit its dispersal as charged liquid droplets (Ref II-29). The low freezing point (-50^{o}C) of the solution is also a desirable characteristic. Some variation in particle weight is encountered with the colloid fuel, but at the present time little difficulty is introduced by the non-uniformity of particle weight. However, the high voltages used to disperse the conductive fuel results in the fuel tanks and distribution system having a high electric potential (about 15,000 volts) with respect to their surroundings. Aboard a satellite the higher voltage requirement greatly increases the physical size of the package because of the need for large insulating gaps between the colloid thruster system and other equipment aboard the spacecraft. Such problems do not exist with cesium and mercury thrusters that operate well at lower voltages.

From the discussion above it is possible to formulate characteristics for improved ion fuels for electrostatic thrusters.

1. The fuel should produce ions that have a high molecular weight relative to cesium or mercury.

2. The ions should be singly charged and be capable of being ionized at low voltages, e.g., the first ionization potential should be five volts or less. Also, the second ionization potential should be substantially higher than the first ionization potential (greater than 10 ev).

3. Ions from the fuel should all have the same mass.

4. The fuel should have a melting point as low as possible. A value of -40°C or lower would be ideal, but a melting point as high as that of cesium (28.5°C) could be tolerated.

5. The fuel should have moderate volatility. A vapor pressure greater than 0.1 torr at 250°C would be acceptable. The requirement for volatility could disappear if a liquid cathode were employed (Ref II-30).

6. The fuel should be thermally stable to above 250°C.

An indication of the type of chemical entities that could be used as substitutes for the usual ion fuels is shown in published papers on ammonium (Ref. II-31) and tetramethylammonium (Ref. II-32) amalgams. These materials, while too low in molecular weight and too unstable for the projected use, do have one ionization potential much lower than the second ionization potential, as does cesium.

A material that could possibly meet the requirements of molecular weight and thermal stability while still retaining one low ionization potential is $(C_6H_5)_4As$, tetraphenylarsonium radical. Its molecular weight would be 383, approximately three times the mass of a cesium ion. The thermal stability of tetraphenylarsonium should be considerably better than that of the pseudometals derived from quarternary ammonium salts. With the ammonium radicals, the free electron is completely antibonding; and with a nitrogen atom as the bonding center of the molecule, no d orbital participation would enhance bond strength between the central atom and its ligands. However, in tetraphenylarsonium the presence of the free electron would be stabilized by spreading out through the aromatic substituents and d orbital participation in the bonding between the arsenic atom and its aromatic ligands would strengthen the bonding to the substituents compared to the bonding of protons or alkyl groups to nitrogen in the ammonium radicals.

Since radical species such as tetraphenylarsonium would be predicted to be about as strongly reducing as an alkali metal, the preparation and isolation of such a material requires the application of techniques that would avoid product decomposition by reaction with moisture or atmospheric oxygen. The ammonium radicals were never isolated but only prepared in mercury amalgams from liquid ammonia or aqueous solutions by electrolysis of the corresponding ammonium salt. Obviously, a mercury amalgam is not desired in this case. What is necessary is a solvent that will dissolve some tetraphenylarsonium salts and will also be stable in contact with a material having the reactivity of an alkali metal. Fortunately, such solvents exist, they are the higher glymes such as diglyme $(CH_3OC_2H_4OC_2H_4OCH_3)$, triglyme, and tetraglyme. These solvents form stable solutions with alkali metals at reduced temperatures. Therefore, electrolysis of a solution of a tetraphenylarsonium salt in a diglyme solution at reduced temperatures on platinum electrodes should prepare free tetraphenylarsonium radical. The extension of this concept to include various quaternary phosphonium and stibonium salts would give greater scope for candidate materials.

The preparation of tetraphenylarsonium radical would serve merely as a demonstration that a stable, high molecular weight material having the required electronic characteristics could be prepared. More desirable ion fuels would have much higher molecular weights and would have assymetry built into the molecular structure so that lower melting points would be obtained. Since immediate precursors are commercially unavailable, the more complex fuels would have to be prepared from synthesized materials. From the readily synthesized 1, 2, 3-tritertiary butyl benzene a likely fuel could be $(t\text{-}Bu)_3C_6H_2\ _3SbC_6H_4CH_3$, tris(tritertiarybutylphenyl) tolyl stibonium radical. This material would produce an ion with an atomic weight of 948, approximately seven times as heavy as a cesium ion. If fluorinated substituents on the aromatic groups could be satisfactorily employed, molecular weights could be increased still more without sacrificing volatility. An example of this type could be $\{(CF_3)_3CCH_2\ _2C_6H_3\}_3SbC_6H_4CH_2CF_3$, with a molecular weight of 1853 (approximately 14 times as heavy as a cesium ion).

In summary, a series of pseudometals derived from quaternary phosphonium, arsonium, or stibonium salts has been proposed as substitute fuels for use in ion thrusters. With aromatic ligands these materials would be expected to be more stable than the radicals obtained through electrolysis of tetraalkylammonium salts. Using the proper solvent system synthesis of pure pseudometal species seems very likely. These new materials would be expected to have chemical properties similar to alkali metals and even possibly be good electrical conductors or semiconductors depending upon the number and type of substituents.

Analysis: Electrostatic engines can only be operated under conditions of high vacuum and for this reason are limited to space applications. Their thrust is limited at the present time to the millipound level, e.g., a five-inch-diameter cesium engine has produced 2.3 millipounds thrust at a specific impulse of 2260 seconds and 16 millipounds thrust at a specific impulse of 8800 seconds (Ref. II-33). Looking somewhat ahead, a thrust of 10 millipounds for a colloid engine is projected for 19[illegible]5 (Ref. II-29). A colloid device has operated at 2.9 millipounds thrust using 150 watts of power. Analysis of the tradeoffs in weight of engine, fuel, and electric generating system for earth satellites has indicated that a specific impulse near 1500 seconds is best (Ref. II-29). In spacecraft where an atomic-powered electric generator is employed, higher specific impulses should be highly desirable, especially for trips to the outer planets.

Present metal-fueled ion thrusters are at a stage of development where they are more compact and generate higher thrusts than the colloid devices. However, they lose too much energy in the ionization process to remain competitive with colloid engines. A comparison between the two devices indicates that a colloid ion thruster will give 2.9 millipounds of thrust for 150 watts power at a specific impulse of 1500 seconds while a cesium-fueled ion engine (Ref. II-33) will give the same thrust for 310 watts power at a specific impulse of 2260 seconds. If a cesium substitute were employed that was ten times as heavy, a scaled-down version of the metal ion engine should require about 200 watts power at the same specific impulse as above. With the specific impulse reduced to 1500 seconds, the power required would drop to about 150 watts, essentially the same as for the colloid thruster. Therefore, a pseudometal-fueled thruster of current design could equal the performance of the incompletely developed colloid thrusters without the disadvantage of the high electric potential of the fuel tanks and distribution systems. However, if additional electrical power was available, the pseudometal-fueled device ejecting ions ten times as

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Shortcomings: The shortcoming of the proposed new ion fuels is that no such materials are now available.

Conclusions: Present chemical knowledge indicates that the generation of stable, low ionization potential, massive, free radicals may be feasible. Such radical species could markedly improve the efficiency of metal ion thrusters at lower specific impulses and also significantly enlarge the achievable thrust levels of present ion engines. Basic research answering the question, "Are stable radicals such as described in the text capable of existence?" could be completed in about six months at a low manpower level. Further extension to a radical species obtained with more difficulty, having more optimum properties, could be projected to take an additional 6 months.

Title: Alfven Wave Propulsion

Concept: Alfven wave propulsion uses magnetohydrodynamic waves to achieve thrust (Ref. II-34 through II-36).

Attributes: The power level generated by Alfven disturbances is high enough to: (1) adjust satellite altitudes without propellant expenditure if an electrical power source is available through use of either solar panels or nuclear generator, (2) control satellite attitude by exploiting torques, (3) passively convert satellite kinetic energy to electrical power by operating in reverse, and (4) counteract atmospheric drag effects. The beauty of the concept is that it could be powered by solar energy and, thus, requires no on-board propellants for stationkeeping.

Description: An Alfven wave propulsion engine is shown in Figure II-29. It consists of two flat, rectangular, conducting plates, maintained continuously at a potential difference. This engine moves across a magnetic field ($\bar{B}_e$) at some velocity ($\bar{v}_c$) in a conducting medium like the rarified plasma of space.

Analysis: A conductor moving across a magnetic field has an induced electric field given by the equation:

$$\bar{E}_c = \bar{v}_c \times \bar{B}_e$$

Where:

$\bar{E}_c$ = induced electric field

$\bar{v}_c$ = conductor velocity

$\bar{B}_e$ = magnetic field flux

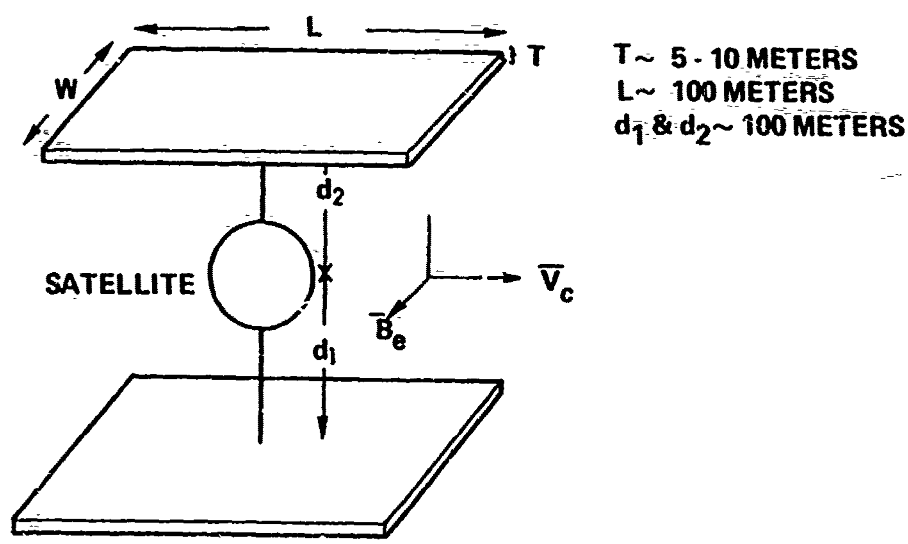


Figure II-29. Alfvén Wave Propulsion Device (See Ref. II-35)

If the surrounding environment is a good conductor, Alfvén waves or magnetohydrodynamic (MHD) disturbances of frequency much less than the ion cyclotron frequency $W_c = eB/mc$ propagate one-dimensionally along the direction of the magnetic field lines. The resulting current in the conductor (Figure II-30) results in a drag force $\bar{F}_D$ according to the equation:

$$\bar{F}_D = \bar{J}_c \times \bar{B}_e$$

where $\bar{J}_c$ is the resultant current density.

Now, if a source of electrical power is available on the satellite, either from a nuclear generator or solar panels, the direction of drag currents can be reversed and converted to a thrust. Essentially, the $\bar{v}_c \times \bar{B}_e$ field must be negated by a larger superimposed field that yields a net $\bar{J}_c$ and $\bar{E}_c$ in the opposite direction. This creates a net thrust $\bar{F}$ in the direction of velocity. The MHD "wing wave" pattern, created by the flow of current from the flat plates, will be at an angle given by:

$$\text{Tan}\,\alpha = \frac{v_c}{v_A}$$

Where: v_A = Alfvén wave velocity (10^7 to 10^9 cm/sec)

$\hat{A}$ = as in Figure II-31

v_c = speed of conductor through field

The Alfvén wave drag power dissipated, lost through F_D, is given by:

$$P_{Alfven} = I_A d_1 B_e \left(\frac{v_c}{c}\right)$$

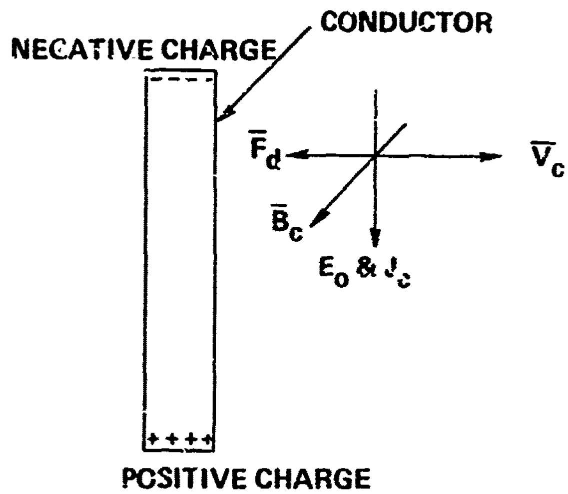


Figure II-30. Characteristics of a Conductor Moving through a Magnetic Field

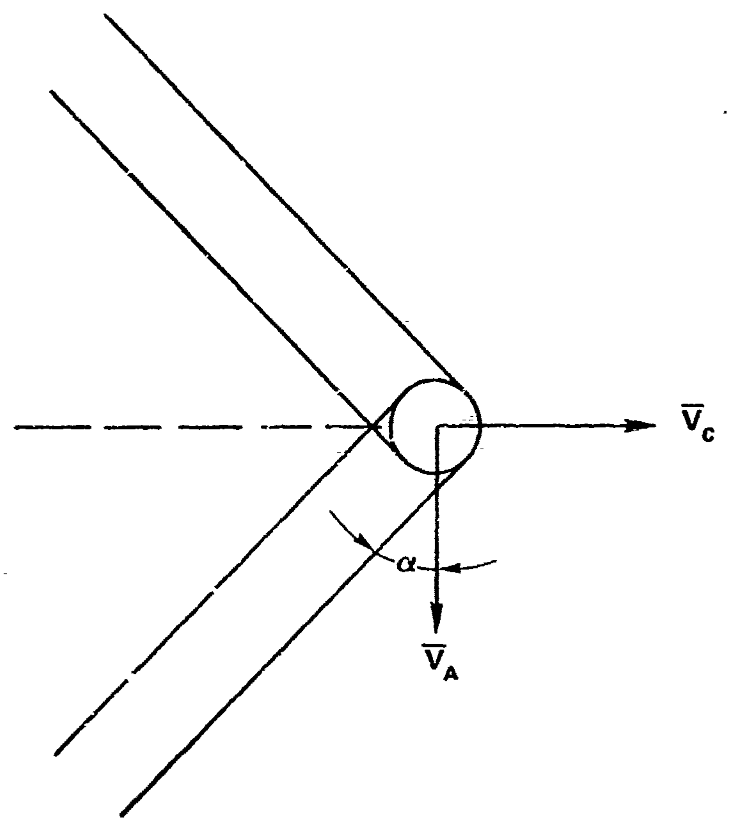


Figure II-31. MHD Wings of Alfven Wave Device (See Ref II-34)

To obtain a thrust, a voltage is impressed in the opposite direction between d_1 and d_2 (Figure II-29) to oppose the current I_A; then an increase in satellite kinetic energy which is proportional to the difference between the Alfv̄en power loss and the impressed input power may be realized. Thus,

$$P_{Kinetic} = VI = I_A d_1 B_e \left(\frac{\bar{v}_c}{c}\right)$$

$$= P_{Alf\bar{v}en} \frac{V - ({}^{v}c/_{c})\, B_e d_1}{({}^{v}c/_{c})\, B_e d_1}$$

Where:
- I = impressed current
- I_A = induced current
- V = impressed voltage
- $\bar{v}_c$ = speed of conductor through field
- c = speed of light
- B_e = environmental magnetic field
- M = length as in Figure II-30

Shortcomings: It is necessary to examine very carefully the basic question of whether a significant external current does indeed flow, and if so, how is it distributed in space. Is the current flow along the earth's magnetic field lines like the transmission lines envisioned in the analysis, or do other effects complicate the current distribution? The evaluation of the Alfv̄en wave propulsion concept will have to wait for a more detailed understanding of the external current distribution, particularly in the vicinity of the vehicle. Alfv̄en waves have never been detected and their presence must be confirmed before this propulsion scheme can be considered valid.

Conclusions: At the present time, Alfv̄en Wave Propulsion is a highly theoretical concept the feasibility of which has yet to be demonstrated.

No real effort has been expended to determine an optimum system configuration. Mission utilization, attractive features and advantages over other propulsion concepts have not been investigated.

Title: Electrostatic Effects

Concept: Propulsion or lift is obtained through the use of electrostatic forces. Several concepts are explored.

Attributes: Utilization of electrostatic forces is an extremely efficient method of propulsion. Not only is there no expenditure of propellant, but the forces involved are conservative in nature. Static electric devices may be charged on earth with a very small amount of mass representing a very large energy density. Because like charges are repulsed, the systems decribed may act to deflect solar radiation. Depending on the charge sign, positive or negative, either electrons or protons may be deflected.

Description: Large electrostatic charges can be placed on spherical conductors. A Van deGraff generator is an example of a device which can be used for this type of charging. The amount of charge that may be placed on a spherical object is limited by the dielectric constant of the surrounding media and the structural strength of the materials involved. Recent studies have indicated that there are a number of materials with dielectric constants between 10^4 and 10^6. These values were measured with low-frequency electric fields. According to Beam (Ref. II-37), the low-frequency response approaches that of a direct current (dc) field. Thus, it is common practice to consider low-frequency dielectric constants to be the same as the dc case. Several applications of highly charged static electric devices are envisioned: (1) Electrostatic Lift - This concept utilizes the forces between similarly charged objects, (2) Electric and Magnetic Field Effects - This concept employs the forces acting on a charged object moving in the earth's electric and magnetic fields, and (3) Storage of High Energy Propellants in Ionized Form (Ref. II-38).

Analysis:

1. Electrostatic Lift - This concept utilizes giant charged spheres arranged symmetrically in the ground just below surface level. These spheres are charged simultaneously at a prescribed rate. Centrally located, relative to the underground spheres, is a single sphere, charged to its maximum, which is lifted into space as the buried spheres are charged (see Figure II-32). The maximum height obtainable is a function of the geometrical arrangement, number of spheres, and their maximum charge. As an illustration, four charged spheres are arranged as shown in Figure II-32, and a fifth initially fully charged sphere is allowed to rise to the point where gravitational attraction balances electrostatic repulsion. In the dynamic case, it may be possible to accelerate the fifth sphere to escape velocity by closely controlling the rate at which the ground-based spheres are charged. The spheres are physically constructed as shown in Figure II-33. A thin film of conductor, copper, is surrounded by a dielectric material, Barium-Strontium-Titanate which has a dc dielectric constant of 1.38×10^4. For the purpose of analysis, a dielectric constant of 10^4 will be used.

a. Charged Sphere Weight: For a copper layer of thickness $t = 10^{-6}$ meters and radius R = 5 meters, the weight is 2.795 kgms or 27.4 newtons. For the $BaSrTiO_3$ of thickness $t = 10^{-2}$ meters and outside radius R=5 meters, the weight is 1.885×10^4 kgms or 1.847×10^5 newtons.

b. Charge on a Sphere: The charge that can be placed on the 10-meter-diameter sphere is 2.78×10^2 coulombs. This is calculated from the equation:

$$q = K\epsilon_o A E_m$$

Where: K = dielectric constant

ϵ_o = permittivity = 8.85×10^{-12} farads/meter

A = surface area

E_m = dielectric strength = 10^7 volts/meter

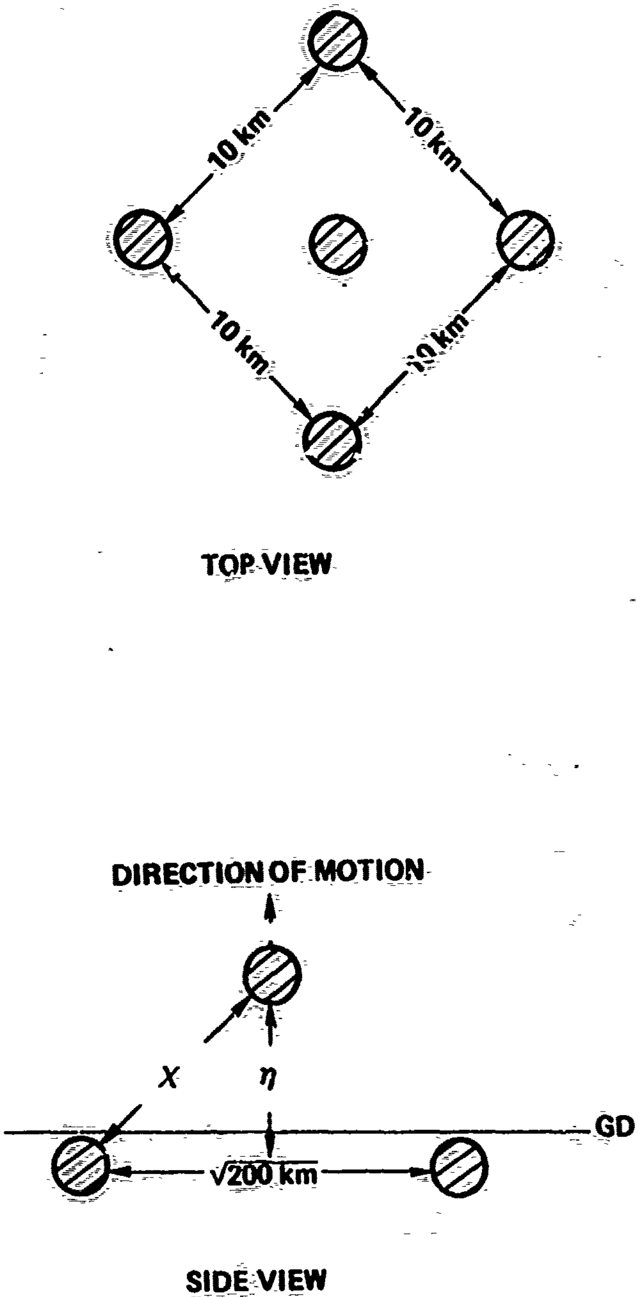


Figure II-32. Arrangement of Electrostatic Spheres

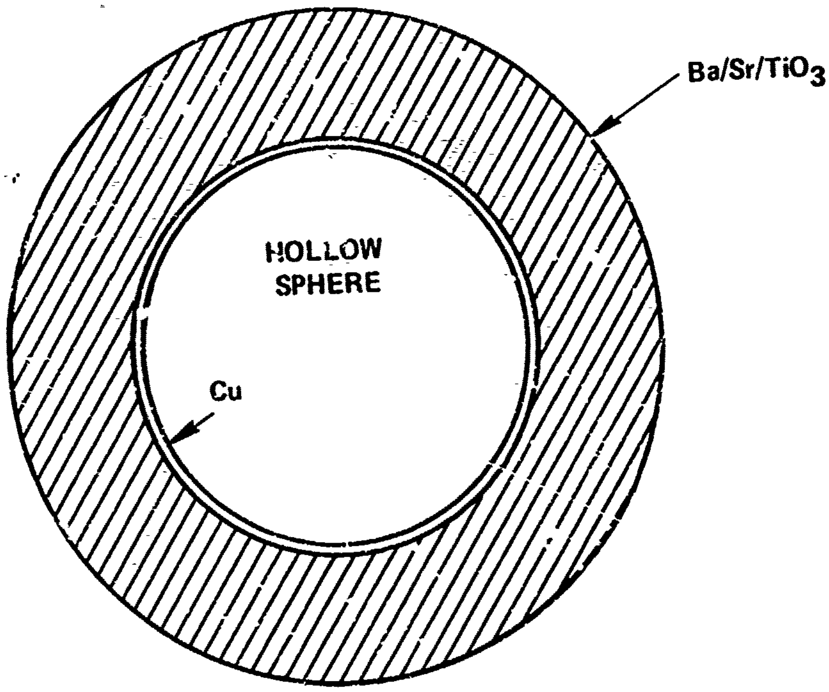


Figure II-33. Electrostatic Sphere Cross Section

c. <u>Force Between Buried Spheres</u>: The forces between charged bodies are extreme at close distances. For the spheres buried in the earth (see Figure II-32), the total force is 3.95-million pounds. Clearly, these spheres need substantial support.

d. <u>Surface Tension</u>: When the spheres are charged, the charges act to repel each other. This repulsion manifests itself as a surface tension which must be supported by the conductor and dielectric. From Jeans (Ref. II-39), the surface tension is:

$$T = (10 \times 10^5)\, \sigma^2 \text{ newtons/m}^2$$

or:

$$T = (28.8\, \sigma)^2 \text{ lbf/in.}^2$$

Where σ is the surface charge density in coulombs per square meter.

For the 10-meter-diameter spheres, $\sigma = 0.885$ coulomb/m^2. This gives a surface tension of 6.5×10^2 lbs/in^2. The yield strength of most materials is in the order of 10^4 psi, so the surface tension effect should not permanently distort the spheres.

e. <u>Lift Force</u>: The repulsion force, due to the four spheres pushing on the fifth sphere, will raise the free sphere to a point where the ratio of force to weight is one. This point occurs at an altitude just under 10^6 meters or 1000 km. This is about 620 miles. The exact solution for the height requires a trial and error solution of the equation:

$$h^2 = \frac{\left[(5 \times 10^3) + h^2\right]^3}{1.83 \times 10^{18}}$$

These results show that a sphere can be continuously supported by an electrostatic field at a height of 620 miles with little loss in energy.

2. Electric and Magnetic Field Effects - It is of interest to look at the forces, electrostatic and magnetic, other than possible buried spheres. Suppose a single, charged sphere, 10 meters in diameter, is moving parallel to the ground at high altitude. The forces consist of (neglecting atmospheric forces-buoyancy, etc.) an electric field force (130 volts/m to an altitude of about 60 km) and a magnetic field force (Lorentz Force). Traveling at 10^3 m/sec (2237 mi/hr), a negatively charged sphere experiences an electric force of about 3.6×10^4 newtons and a magnetic force of about 27.8 newtons. The lifting thrust/weight ratio is about 0.2.

Charged vehicles can also be used in space to provide a thrust for directional changes. Although the magnetic fields are small, a vehicle traveling at high velocity over long distances could completely reverse its direction without any loss in velocity or expenditure of propellant. This zero thrust velocity vector control is feasible for interstellar probes where the radius of curvature is in the order of 10^3 light years (Ref. II-40 and II-41).

3. Storage of High-Energy Propellants in Ionized Form - It has been suggested that a certain configuration of dielectric and semiconducting materials could lead to a capacitor with a dielectric strength greater than 10^8. With this dielectric strength, it is proposed to store ionized hydrogen (Fig. II-34, Ref. II-38). The total charge is estimated at 8.85×10^6 coulomb/m^2 or 4.43×10^6 coulomb/m^2 on each side. Using 1.65×10^{-24} grams for the mass of hydrogen and 13.5 ev per ion for recombination energy, the energy available would be 10^{13} ergs/gm. Utilization of this energy for thrust would lead to performance comparable to metastable chemicals.

Shortcomings:

1. Electrostatic Lift - This concept violates Earnshaw's theorem (Ref. II-39) that states that a static system, consisting of particles which

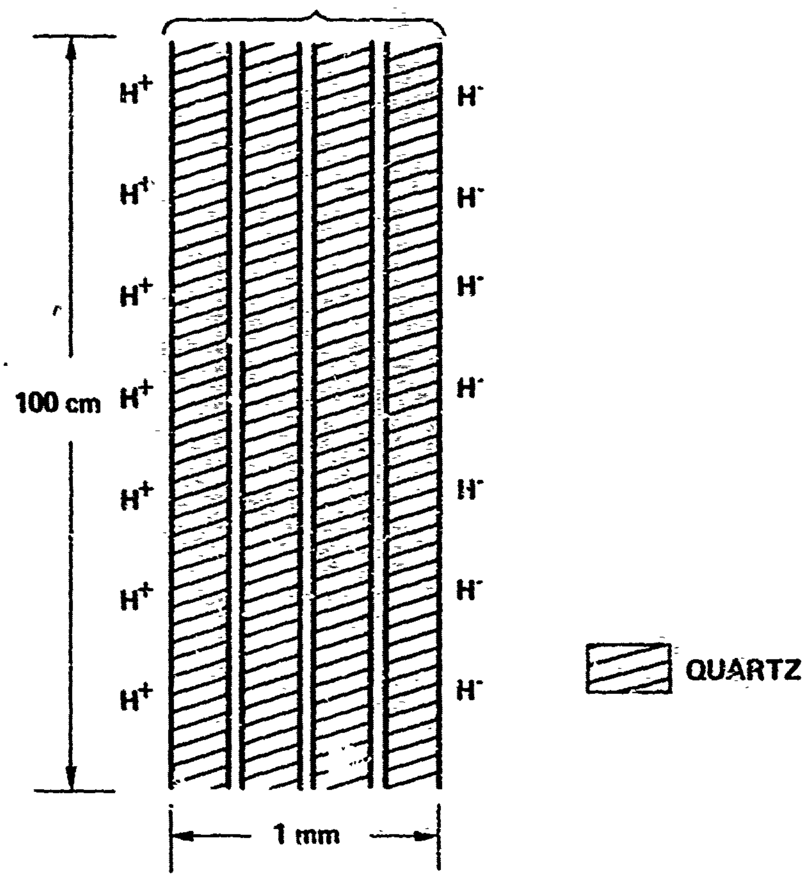


Figure II-34. Sandwich Structure Composed of Quartz (Dielectric) and a Semiconducting Material (See Ref II-38) either repel or attract each other with forces varying inversely with the square of their distance of separation, is inherently unstable. In addition, the analysis of this concept completely ignores the effect of the immense electric fields on the surrounding environment. More than likely, the high fields will cause electrical breakdown of the air and ground with resultant flow of ions to the charged spheres. The accumulation of opposite charge would not only impose a high unit of mechanical and electrical pressure across the dielectric, but it would also nullify the repulsion effect.

2. Electric and Magnetic Effects - Again, the same shortcomings discussed above apply to this concept. The interaction of a highly charged sphere with its environments needs to be more fully analyzed.

3. Storage of High-Energy Propellants in Ionized Form - The pressure on the sandwich structure due to charge attraction is 9.7×10^{24} psi. No known material can withstand stresses of this magnitude. The anomalously high value of 10^{8} is not a true value. Fuller and Ward (Ref. II-42) show that such values result from failure to subtract ionic conduction current components during capacitance measurements. In addition, the charge limit on a capacitor by ion injection is not the breakdown potential but the repulsion between like charges. In general, it appears that the numbers quoted for storage of ionic hydrogen are many orders of magnitude away from present-day realities.

Conclusions: All of the ideas discussed lack theoretical and technical merit. Futhermore, the thrust/weight ratios obtained make these concepts appear unattractive. However, the features of environmental interaction and infinite specific impulse must be recognized as having enormous implications. Additionally, handling and producing charged objects of the magnitude assumed for the analysis may be well beyond the reach of technology for decades to come (see also Ref. II-43 through II-46.

Title: Satellite Drag Make-Up

Concept: The drag force upon a low orbit satellite may be overcome by interaction with the earth's magnetic field (Ref. II-47 through II-51).

Attributes: The scheme needs no propellant to overcome the drag of low-earth orbit. A small amount of electrical power derived from an isotope reactor or solar energy converter would be needed for stationkeeping and satellite rotation relative to the solenoid.

Description: The essential elements of the device are: (1) an earth-orbiting satellite with a large moment of inertia (similar to a gravity-gradient stabilized satellite with eccentric mass attached by a rod or similar structure, i.e., dumbell-shaped, and (2) a superconducting solenoid capable of rotation with respect to the satellite. The solenoid by interacting with the earth's magnetic field is able to sustain a torque on the satellite.

Analysis: To illustrate the principle, assume that the satellite, as in Figure II-35, consists simply of two mass points (m_1 and m_2) connected by a rigid but weightless rod. The rod is in the plane of the orbit and at an angle θ with respect to the local horizontal. For simplicity, assume the satellite orbit to be perfectly circular with a constant aerodynamic drag (centered at the center-of-mass point). For equilibrium to exist with constant inclination θ and angular velocity ω about the center of the earth, the drag force must be overcome by an equal and opposite force.

For the configuration described, if equilibrium exists, conservation of angular moment gives:

$$\frac{d}{dt}\left(m_1\omega r_1^{\ 2} + m_2\omega r_2^{\ 2}\right) = T - Dr_o$$

where

r_o = radius from center of earth to satellite

r_1 = radius from center of earth to m_1

r_2 = radius from center of earth to m_2

T = required torque

D = aerodynamic drag

Assuming uniform circular motion:

$$T - Dr_o = 0 \qquad (1)$$

Also, from conservation of momentum:

$$\frac{d}{dt}(m_o v_o) = R$$

where

$m_o = m_1 + m_2$

v_o = center-of-mass velocity

$R = F_1 + F_2 + D$

and F_1 and F_2 are the gravitational forces acting on masses m_1 and m_2.

For the component perpendicular to r_o in the orbital plane (ϕ_1 and ϕ_2 as shown in Figure II-35):

$$0 = F_1 \sin \phi_1 - F_2 \sin \phi_2 - D \qquad (2)$$

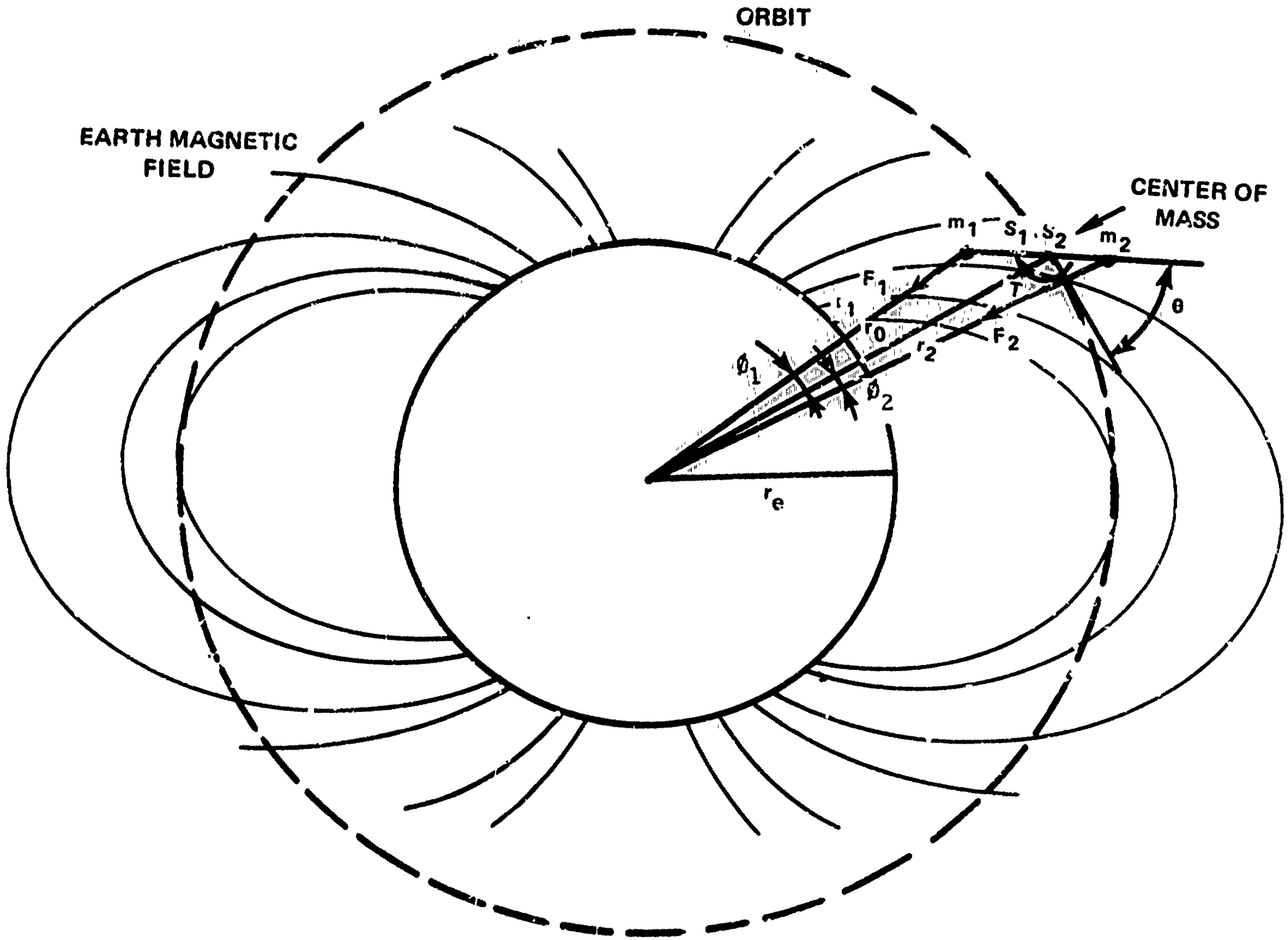


Figure II-35. Schematic of Satellite in Earth Orbit (See Ref. II-47)

And, in addition.

$$F_1 = m_1 g \frac{r_e^2}{r_1^2}, \quad F_2 = m_2 g \frac{r_e^2}{r_2^2}$$

where r_e = earth's radius

Therefore,

$$F_1 \sin \phi_2 = \frac{g r_e^2 m_2 S_2 \cos\theta}{\left(r_o^2 + S_1^2 - 2 r_o S_1 \sin\theta\right)^{3/2}}$$

$$F_2 \sin \phi_2 = \frac{g r_e^2 m_1 S_1 \cos\theta}{\left(r_o^2 + S_2^2 + 2 r_o S_2 \sin\theta\right)^{3/2}}$$

Where S_1 and S_2 are the respective distances of the center-of-mass from m_1 and m_2. Substituting into equation (2), and expanding in terms of S_1/r_o and S_2/r_o:

$$D = \frac{3}{2} \sin(2\theta) \frac{r_e^2 g I_o}{r_o^4}$$

Where the moment of inertia $I_o = m_1 S_1^2 + m_2 S_2^2$. Hence, the drag torce which can be overcome is proportional to sin (2θ) and the amount of inertia. The maximum effect obtained occurs at $\theta = 45^o$, thus:

$$D = \frac{3}{2} \frac{r_e^2 g I_o}{r_o^4}$$

As an example, let $r_o \cong r_e$, $m_1 = m_2 = 1/2m_o$, $S_1 = S_2 = 1/2S$. Then, if $m_o = 10,000$ pounds and the drag acceleration is 1 ft/sec per day, the power P required is:

$$P = WT = 120 \text{ watts}$$

and

$$S = 20,000 \text{ ft}$$

Shortcomings: The analysis is for an idealized "circular" orbit. In reality, all orbits are elliptical. The drag term,

$$\frac{3}{2} \sin (2\theta) \frac{r_e^2 g I_o}{r_o^4},$$

represents only a first-order approximation. The nature and effect of higher-order terms has not been considered. To completely substantiate the drag make-up concept, a detailed study including higher-order terms must be made.

Further analysis of the required solenoid characteristics leads to the conclusion that the total system, satellite plus solenoid, is unstable for all orbits other than a true polar orbit. Consider the equation for the torque (T) on a solenoid coil in the earth's magnetic field:

$$\overline{T} = \overline{M}_s \times \overline{H}_e \tag{3}$$

where

$\overline{M}_s = \mu_o \, Ni \, \overline{A}_s$ = solenoid's magnetic moment

$\overline{H}_e$ = earth's magnetic field intensity

As an example, using rectangular coordinates for a point in space with z - direction parallel to the earth's field at that point, let $\overline{H}_e = h_z \overline{i}_z$ and let $\overline{M}_s = m_x \overline{i}_x + m_y \overline{i}_y + m_z \overline{i}_z$, then

$$\overline{T} = \overline{M}_s \times \overline{H}_e = \left(m_y h_z\right) \overline{i}_x - \left(m_x h_z\right) \overline{i}_y \tag{4}$$

Equation (4) shows that the restoring torque is only in the x-y plane and the satellite is free to rotate about the z-axis. Actually the gravity gradient forces will give a small amount of stability if the z-axis torque is small. A z-axis torque will result if the satellite deviates from true magnetic north. Therefore, the system's successful operation depends on the alignment of the orbit with true north.

Case One shows the ideal case of a true polar orbit where the required solenoid torque equally balances the satellite torque. In Case Two, because of the satellite's deviation from true north, an additional torque is required to stabilize the system. This torque, within limits, can be supplied by gravity gradient torques. The angle then is:

$$\text{Tan } \alpha = \frac{\text{Torque of Gravity Gradient}}{\text{Torque of Reaction}}$$

To minimize perturbing orbital forces caused by interaction between the earth's magnetic field and the solenoid and to obtain maximum torque, the solenoid must be turned nearly perpendicular to the earth's magnetic field, as shown in Figure II-36. However, there is no way to completely avoid the interaction forces on the coil. Eventually, due to these forces, the orbit will be changed significantly. The exact nature of this change is difficult to analyze. If the resulting orbit deviates from true north, then the concept is valid for only a short time.

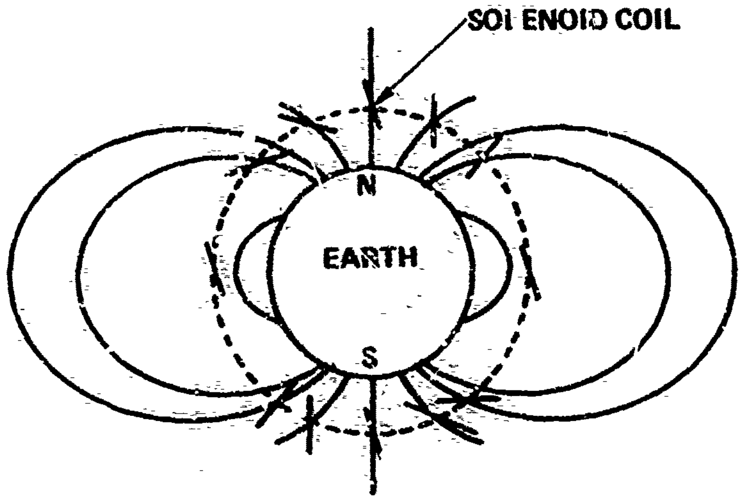


Figure II-36. Solenoid Position in Earth's Magnetic Field

Stability is further hampered by the diamagnetic nature of the superconductor. A normal conductor, moving through a magnetic field, is subject to induced currents which would tend to damp out oscillations of the solenoid. However, a superconductor is affected to a much lesser degree by these currents and, therefore, loses its inherent advantage.

Conclusions: The validity of the concept is highly questionable. A complete in-depth analysis needs to be made of the orbital mechanics, coupled with the magnetic forces and torques, in order to demonstrate concept validity. Furthermore, the fact that this concept is applicable for only polar orbits decreases the value of this concept.

Recommendations: This concept should be developed to the point of ascertaining its theoretical validity. This could be accomplished fairly easily. If this concept should prove valid, it should be considered for polar missions.

Title: Electromagnetic Spacecraft Propulsion

Concept: This concept uses the earth's magnetic field for propulsion

Attributes: The greatest advantage of this concept is that the system is initially charged on earth with a tremendous amount of massless energy which is stored in a low-loss propulsion system. The superconducting magnet itself is a perpetual motion device. This is a compact, efficient method of storing energy. Once in space, stationkeeping energy can be supplied by solar energy or a small isotope reactor. Cryogenic cooling becomes a minor problem because of reduced convection and the low blackbody temperature of deep space. Similar to other low-thrust vehicles, this system is capable of accelerating to very high velocities when operating over great distances for substantial periods of time. This concept also has important environmental qualities in that the tremendously high magnetic field acts as an extremely efficient solar radiation shield by deflecting dangerous, high-energy particles. Because superconductors have no electrical resistance, induction currents are small and lossless. This system could be used to decelerate vehicles approaching the earth at high speed. Militarily, this concept could, with its high magnetic field, destroy, deflect or severely damage incoming high-speed projectiles. Variations of this concept could operate in the same mode envisioned for certain high-speed magnetic transportation devices that skim along a conducting roadbed.

Description: Propulsion is obtained by interaction with the earth's magnetic field in the near-space (within 10 earth radii) environment and then extended to deep space (interplanetary and interstellar). The system envisioned for near space depends upon the interaction of a solenoid coil (Figure II-37) with the earth's magnetic field. The solenoid

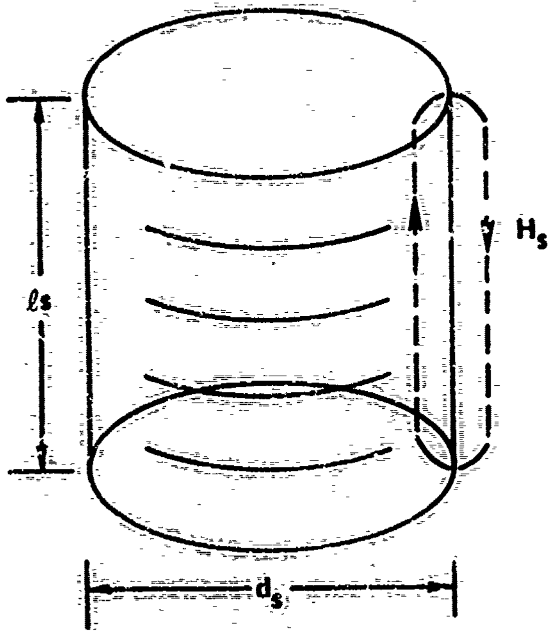


Figure II-37. Schematic of Solenoid Coil
(See Ref. II-54)

coil is employed as a large cryogenically cooled magnet acting as a magnetic dipole with north and south poles which repulse the corresponding magnetic poles of the earth. Figure II-38 shows this concept for a single coil in orbit. This is the same effect obtained with two bar magnets when like poles are forced together.

Analysis:

1. Solenoid Wall Thickness

a. $Ni_s = H_s l_s = H_s N t_s$ assuming the length is equal to the number of coils times the wire thickness ($l_s = Nt_s$) with negligible insulation between wires (Ref. II-52).

$i_s = H_s t$ for a long solenoid or $H_s d_s$ for a single coil (1)

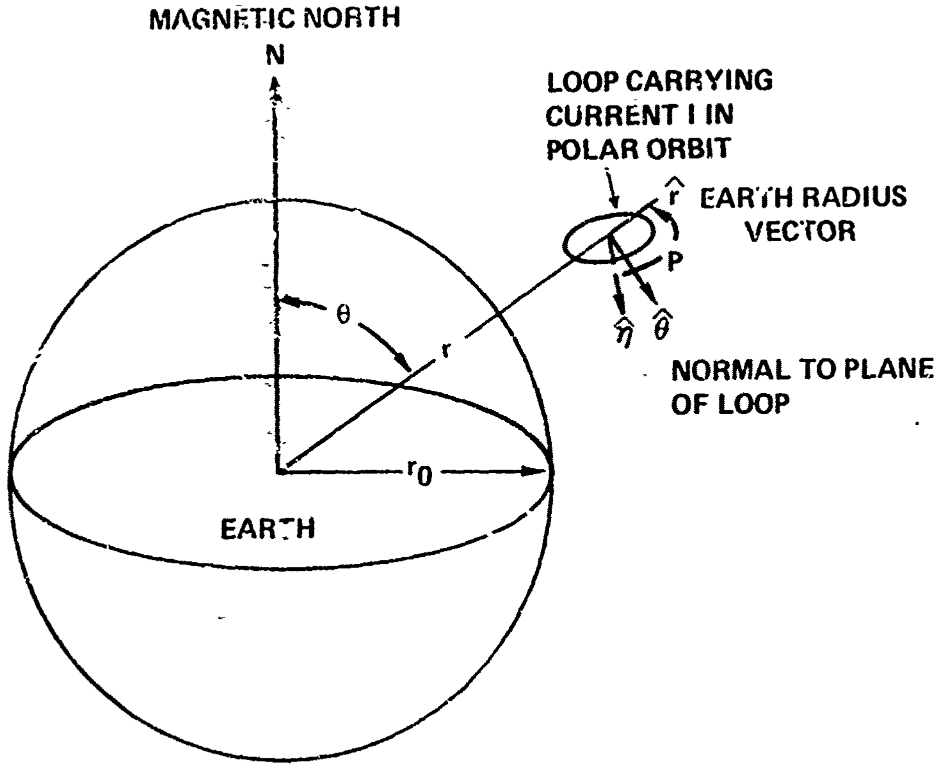


Figure II-38. Diagram of Solenoid Coil in Earth Orbit (See Ref. II-57 and II-58)

where

i_s = solenoid magnetizing current

H_s = solenoid magnetizing force

l_s = solenoid length

t_s = solenoid wire diameter, wall thickness

N = number of solenoid coils = 10^6

d_s = diameter of solenoid loop

b. By definition:

$$J_s = \frac{i_s}{A_w}$$

$$i_s = \frac{\pi}{4} t_s^2 J_s \tag{2}$$

where:

J_s = solenoid current density

A_w = wire cross-sectional area

c. Combining equation (1) and (2) from above leads to:

$$t_s = \frac{4 H_s}{\pi J_s}$$

and since

$$H_s = \frac{B_s}{\mu_o}$$

$$t_s = \left(\frac{4}{\pi \mu_o}\right) \frac{B_s}{J_s} \tag{3}$$

Equation (3) defines the wire thickness in terms of the electric and magnetic properties of the solenoid wire. At this point, it is necessary to assume that the solenoid is a superconducting magnet. It is ridiculous to ever consider operating a solenoid of this magnitude outside the superconducting range. Not only would the heat losses be tremendous, but the power supply, carried along to maintain the large currents, would be a tremendous weight penalty. It is, therefore, reasonable that once this system is in full operation, it should never be turned off.

The critical limit curves for several superconducting materials are shown in Figure II-39. One of the best superconducting materials, Niobium-Tin, is representative of the maximum material capability.

The state of the art for superconducting magnets is a field strength of about 10 webers/square meter at 10^9 amp/square meter. The solenoid coil must have sufficient strength to support the radial pressure (P_r) caused by the interaction of the solenoid current with the magnetic field produced (Ref. II-53).

$$P_r = \frac{B_s^{\,2}}{2\mu_o} \qquad \frac{\mu_o H_s^{\,2}}{2}$$

For a cylindrical, thin-walled pressure vessel, the total hoop stress (S_t) is:

$$S_t = \frac{1/2 P_r d_s}{t}$$

$$\therefore t_s = \frac{P_r d_s}{2 S_t}$$

In terms of the magnetic field:

$$t_s = \frac{B_s^{\,2} d_s}{4 \mu_o S_t} \tag{5}$$

A thickness of 0.01 meter is calculated, based on current density and magnetic field limitations, and a thickness of 0.585 meter is calculated, based on radial pressure. It has been assumed that a new low-density, high-strength material is available with a density $\rho_s = 1.384 \times 10^3$ kgm/m^3 and tensile strength $S_t = 3.4 \times 10^9$ newtons/m^2. The thickness of the wire must be sufficient to support the radial force, and at the same time, as

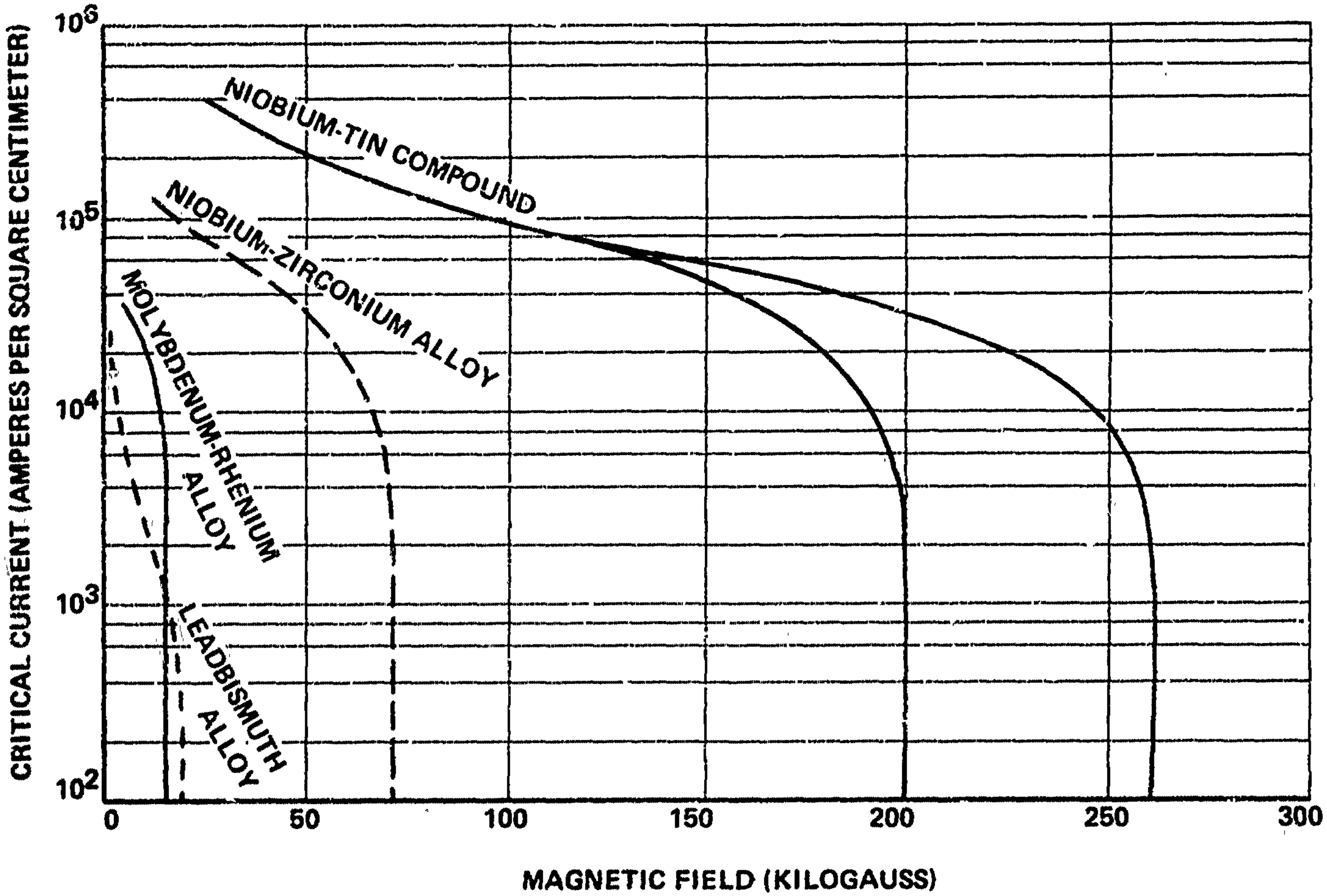


Figure II-39. Status of Superconducting Materials (See Ref II-59) will be seen later, maximum current density must be maintained to yield high thrust/weight ratio. The magnetic flux value for which the two thicknesses are equal is $B_s = 0.1733$ webers/m^2. The thickness is $t_s = 1.757 \times 10^{-4}$m. The current density is still 10^9 amp/m^2.

2. <u>Acceleration</u> - The transitional force ($\overline{F}$) exerted on a dipole placed in the non-uniform magnetic moment, $\overline{M}_s$, in the earth's magnetic field ($\overline{H}_e$) is (Ref. II-54):

$$\overline{F} = \mu_o (\overline{M}_s \cdot \overline{\Delta}) \overline{H}_e$$

where:

$$\overline{M}_s = N i_s \overline{A}_s$$

$\overline{A}_s$ = solenoid cross-sectional area vector

For the ideal case considered here, the earth's field will be treated as a perfect dipole. The earth's field components are:

$$B_r = 2B_o \left(\frac{r}{R}\right)^3 \cos\theta$$

$$B_\theta = B_o \left(\frac{r}{R}\right)^3 \sin\theta$$

where:

B_o = constant = 0.31×10^{-4} webers/m^2

r = radius of earth = 6.378×10^6 meters

R = distance from earth's center to solenoid

Consider the case where the area vector $\overline{A}_s$ is always aligned with the magnetic field. Placing the solenoid in this position also reduces the torque on the vehicle. Then, for the area vector:

$$A_r = A_s \cos\theta$$

$$A_\theta = 1/2 A_s \sin\theta$$

Using the above relations for the earth's field and the solenoid position, the propulsive force is:

$$\overline{F} = -N i_s A_s B_o \frac{r^3}{R^4} (5 \cos^2\theta + 1)\overline{r} + (\frac{5}{4} \sin 2\theta)\overline{\theta} \quad .$$

The weight of the solenoid coil may be approximated by the equation:

$$Wt = g\rho_s \text{ Vol} = g\rho_s (\pi d_s t_s l_s) = 1.315 \times 10^5 \text{ newtons}$$

The ratio of thrust/weight is:

$$F/Wt = \frac{\pi J_s d_s B_o}{16\rho_s g} \left(\frac{r^3}{R^4}\right) \left[(5 \cos^2\theta + 1)\overline{r} + (\frac{5}{4} \sin 2\theta)\overline{\theta}\right]$$

As an example, consider two cases:

a. At the earth's north or south poles, the thrust/weight ratio is 0.452×10^{-4} straight up when the solenoid is located at ground level. And although there is no $\overline{\theta}$ force at any other point slightly offset from the poles component this force exerts an acceleration toward the magnetic equator.

b. At the earth's magnetic equator, the solenoid would experience a thrust/weight ratio of 0.075×10^{-4} radially outward from ground level. Note also that the $\bar{\theta}$ directional force is zero. On either side of the equator, the $\bar{\theta}$ force will tend to accelerate the solenoid toward the equator.

The forces acting on the solenoid may be schematically represented as shown in Figure II-40. When the ball is at either pole (top of the hill), the position is unstable. When placed at the equator (in the valley), the ball will tend to remain there in a stable position.

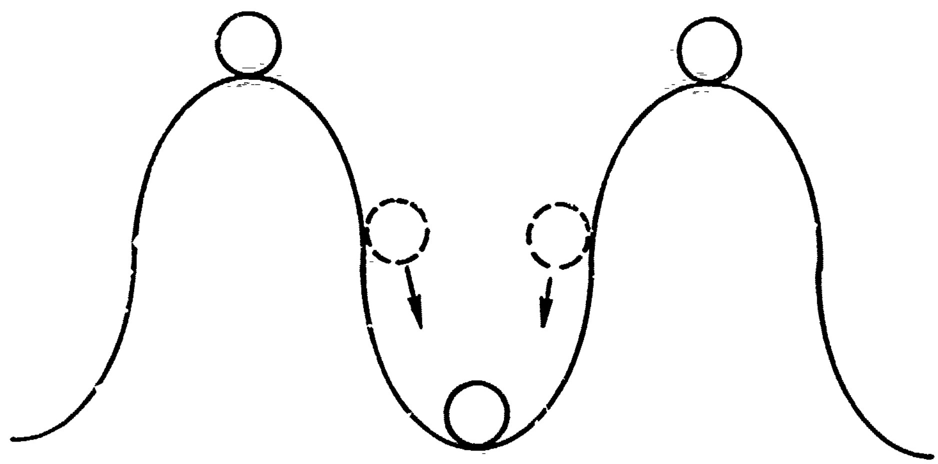


Figure II-40. Orbital Forces on Solenoid

For the solenoid system considered, the thrust is 5.94 newtons (1.35 pounds) and the weight is 1.32×10^5 newtons (2.99×10^4 pounds). The same analysis may be repeated for a 100-meter-diameter, single-loop coil with slightly different results. The magnetic field is 14.35 W/m^2 and a coil thickness of 1.205 meters. The thrust is 179 newtons (40.75 pounds), the weight is 3.34×10^6 newtons (7.6×10^5 pounds), and the thrust-to-weight ratio is 0.537×10^{-4} at the magnetic pole.

3. <u>Torque</u> - There is a torque acting on the magnetic solenoid. This torque tends to rotate the coil into a position such that opposite poles of the earth and solenoid are aligned. In this position, the interactive forces are attractive rather than repulsive. The torque ($\overline{T}_s$) is given by the equation:

$$\overline{T}_s = \overline{M}_s \times \overline{B}_e$$

The maximum torque value for the solenoid system under consideration is 1.18×10^7 newton-meters. The torque occurs when the solenoid area vector $\overline{A}_s$ is perpendicular to $\overline{B}_e$. This means that there is a maximum force of 1.18×10^5 newtons acting at the diameter of the solenoid. If the solenoid is aligned with the magnetic field to within 1°, the torque can be reduced by a factor of 10^2. Clearly, the torque problem needs further analysis to find a means of control that does not penalize the basic interaction concept.

4. <u>Energy Storage Density</u> - The energy E_d that can be stored in a magnetic field is (Ref. II-53):

$$E_d = \frac{B_s^2}{2\mu_o} = 1.2 \times 10^4 \text{ joules/m}^3$$

Converting this to a mass density for the solenoid coil previously analyzed gives:

$$E_d = 8.65 \times 10^4 \text{ ergs/gm}$$

For the single coil, the energy density is 5.92×10^8 erg/gm.

5. The Biological Aspects (Solar Plasma Shield and Effects on Human Beings) - Three types of shielding for manned space flight have received serious consideration. Until recently, shielding concepts have relied entirely upon passive bulk materials. These shields function by degrading and attenuating the energy of the incoming particles until they are either absorbed in the shield or their energy reduced to a harmless level. The thickness of this type of shielding is determined by particle intensity, energy distribution, and mission duration. Passive shielding requirements to provide protection from moderate solar flares are of the order of a few tons. In addition, passive shields do not stop the production of secondary particles.

Active shielding covers two concepts: electrostatic shielding and magnetic shielding. Electrostatic shielding relies upon the deflection of similarly charged bodies. Thus, a positively charged space vehicle would repulse protons. However, this type of shielding will accelerate electrons which, if striking metal, can produce dangerous X-rays. A negatively charged vehicle will accelerate protons which tend to produce neutrons as secondary particles. The unconfined magnetic field produced by a solenoid offers exceptional promise as a shield. Because of the large distances affected by the solenoid's field, the deflection of both positive and negative particles is initiated at considerable distance from the coil. In Figure II-41, the shielded and partially shielded regions for a dipole field are shown. The magnitude of these areas and the particle energies excluded can be calculated. In addition, the problems of secondary particles and X-rays is greatly reduced because fewer particles impact the vehicle. The propulsion system under consideration thus provides, in addition to thrust, an excellent radiation shield. Its high effective magnetic field is able to deflect all but the most energetic particles. Of prime consideration might be the effects of high magnetic fields on human beings. Reports indicate that many people have had parts of their

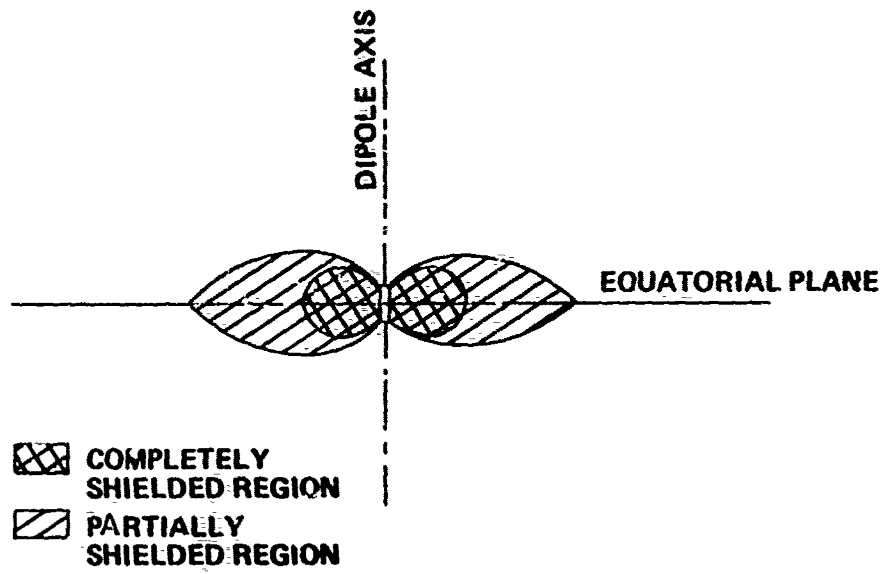


Figure II-41. Shielded and Partially Shielded Regions for a Dipole Field (See Ref. II-60)

anatomy immersed in high magnetic fields for periods of several hours with no noticeable effects. However, further work needs to be done to assure that there are no unusual effects (Ref. II-55).

6. The Meissner Effect - A superconductor, when placed in a magnetic field, will expel the field because of its diamagnetic nature. The coils previously analyzed will not only expel the earth's field from the superconductor but also the area or volume surrounded by the coils. This effect leads to a buoyancy force (F_M) in the direction of decreasing field strength given by (Ref. II-56).

$$F_M = \frac{1}{24} \frac{(d_s)^3 \bar{B}_e}{\mu_o} (\bar{\Delta} \cdot \bar{B}_e)$$

In the radial direction, the Meissner force would be 0.6 X 10^{-4} newtons.

7. Deep Space Magnetic Gradient - The sun is also a source of magnetic fields. The sun's field pervades the space between the planets of the solar system. In deep space where planetary fields are essentially zero, interaction with the sun's magnetic field is possible. Alfvén and Fälthammer (Ref. II-55) report that the general magnetic moment of the sun is $M = 2.1 \times 10^{25}$ webers/square meter. Assuming a dipole field for the sun, the radial force exerted on the solenoid coil at a distance of the earth's orbit is 4.86×10^{-7} newtons. However, this number might be deceiving because it is for the general field. Strong anomalous magnetic disturbances caused by solar "storms" may cause this number to increase by several orders of magnitude.

Shortcomings: Clearly the concept to interact with environmental magnetic fields cannot be used as an earth-to-orbit booster system. The additional weight of a cryogenic cooling system will make this concept even less attractive. The storage of energy in the system is very poor. This low acceleration propulsion concept is most applicable to near-earth missions if the torque problem can be solved.

The application of the magnetic solenoid concept is limited by state-of-the-art superconductors and the extreme weakness of environmental magnetic field gradients. This concept at the present time has no significant value as a propulsion concept.

Title: Superconducting Particle Accelerator

Concept: This concept uses magnetic field forces to accelerate small conducting particles for propulsion.

Attributes: As with all thrusters, high thrust requires high power. The advantage of this system lies in the possibility of achieving nearly 100 percent efficiency based on input power. Additionally, this concept avoids high-temperature materials and techniques (Ref II-61).

Description: The concept utilizes the perfect diamagnetic nature of small superconducting particles. The particles are accelerated in a tubular, linear induction, electric motor by either Meissner forces or Lentz forces. The ideal (100 percent efficient) performance is tabulated in Table II-6 (Ref. II-61).

TABLE II-6. PREDICTED PERFORMANCE OF PARTICLE ACCELERATOR

Field Strength (gauss)	10^2	10^5	10^3	10^5
Accelerator Length (cm)	10^2	10^3	10^2	10^2
Particle Mass (gm)	8×10^{-10}	8×10^{-6}	8×10^{-10}	8×10^{-6}
Final Velocity (cm/sec)	5×10^7	5×10^6	5×10^8	5×10^8
Mass Flow (gm/sec)	2×10^{-4}	2×10^{-2}	2×10^{-3}	2×10
Specific Impulse (sec)	5.1×10^4	5.1×10^3	5.1×10^5	5.1×10^5
Thruster Power (kw)	2.5×10	2.5×10	2.5×10^7	2.5×10^8
Particle Frequency (p/sec)	2.5×10^5	2.5×10^3	2.5×10^6	2.5×10^6
Thrust (grams)	1.02×10	1.02×10^2	1.02×10^3	1.02×10^7

Analysis: This concept will be analyzed by two methods. The first method will utilize the Meissner effect. The second method will show attainable velocities limited only by the critical field strength of the particle material in an induction force accelerator.

1. In a Meissner Force Accelerator, the force (F) on a small superconducting particle is given by (Ref. II-62).

$$F = -\frac{1}{3} \frac{r^3 B}{\mu_o} \frac{dB}{dx}$$

where:

r = particle radius

μ = permeability

B = magnetic field flux pushing the particle

$\frac{dB}{dx}$ = gradient of magnetic field in the direction of force

For a single-loop coil with the particle at the center, an acceleration of 3×10^3 m/sec^2 may be obtained. This number assumes a field flux of 10 webers/m^2, a coil radius of 10^{-1} meters, a linear distance of the particle in front of the coil of 10^{-2} meters, and a particle mass of 8×10^{-9} kgm. The force exerted on the particle is 2.4×10^{-5} newtons. For 2.5×10^6 particles, a thrust of 60 newtons could be realized in the ideal case. The specific impulse would be 3×10^3 newtons-sec/kgm. In a 1-meter length, the particles could be accelerated to a velocity of 7.74×10^3 cm/sec; not a very high exit velocity.

2. In a normal induction accelerator, the projectile being driven by the electromagnetic wave acts as a heat sink for the energy dissipated in it by ohmic heating. Therefore, the limiting velocity is controlled by the

particle's electrical and thermodynamic properties until the melting point is reached. But if superconducting particles are used in a field less than the critical field, an essentially lossless system exists. The force (F) is produced on the particles as a result of Lentz's Law which states that the direction of an induced EMF is such as to oppose the cause producing it. The force is given by:

$$F = 1/2 i^2 \frac{dM}{dx}$$

where:

i = primary coil current

$\frac{dM}{dx}$ = gradient of mutual inductance

In practice, the mutual inductance is quite difficult to solve, but a number of empirical equations have been developed for various geometries. For this case, using aluminum particles, the force may be calculated from the equation (Ref. II-63):

$$F = (3/50)\pi^2 i^2 G(x)A(y) \quad (r/R)^3$$

where:

i = rms current = 0.070×10^5

x(dimensionless) = $2\pi r\ (\nu/10^3)^{1/2}$

r = sphere radius = 0.82×10^{-2} cm

ν = frequency = 10^4 cps

R = coil radius = 10^{-1} meters

= resistivity = 2.7 micro-ohm. cm

$$G(x) = 1 - \left(\frac{3}{4x}\right) \cdot \frac{\sinh 2x - \sin 2x}{\sinh^2 x + \sin^2 x}$$

$$y(\text{dimensionless}) = \frac{Z}{R}$$

$$Z = \text{distance between particle and coil}$$

$$A(y) = y/(1+y^2)^4$$

Use of this equation leads to a thrust of 1.177×10^5 newtons for 2.5×10^6 particles and a specific impulse of 0.588×10^7 (newton-sec)/kgm. These numbers are very close to the theoretical prediction of Table II-6.

Shortcomings: Thom and Norwood (Ref. II-64) indicate that they were able to achieve velocities up to 4.2×10^4 cm/sec compared to a predicted 4.5×10^4 cm/sec (93.4 percent). However, these were individual particles and there was a wide spread in the data. Minimum velocities ranged as low as 2×10^4 cm/sec. The wide variation in velocity was attributed to the accelerator design and use of sliding contacts for the coil. The problems encountered in using a large number of small particles cannot be predicted. There will be a physical interaction effect due to differences in particle position and velocity in the field. Some particles will be accelerated faster than others. Furthermore, the induction coils, power supply, and cryogenic equipment make this concept a low thrust/weight ratio propulsion device. However, further analysis is required to obtain the exact thrust/weight ratio. Also, although the field is below the critical level, the generated eddy currents may be above the critical current density level. In this case, a much lower field would have to be used.

Conclusions: The acceleration of superconducting particles by magnetic induction is based on sound physical principles, appears feasible and may be capable of delivering high performance at very high levels of efficiency.

Recommendations: This concept's feasibility has been demonstrated by Thom and Norwood using normally conductive particles. A program should be initiated to study and demonstrate the concept with superconducting particles. If the concept's feasibility can be successfully demonstrated, a configuration that more closely resembles a propulsion system should be designed and tested.

Title: Antigravity Propulsion

Concept: Utilizing the control of gravitational forces of the earth and other celestial bodies as a means of propulsion.

Attributes:

1. Infinite specific impulse
2. Near speed of light velocities attainable
3. Minimum damage to environment
4. Economic exploitation of space

Description: Two concepts are considered. One utilizes a new physical concept of gravitational absorption (gravity screens) and the second is based on the concept of a unified field theory (using electromagnetic analogies to gravity control). The hardware which will be described is theoretical since fabrication of antigravity devices has not been attempted (as far as this author has been able to determine).

Analysis: Before attempting to control gravity it will be necessary to know exactly what causes gravity. Many reputable scientists including Michael Faraday, Max Born and Albert Einstein have attempted to explain this as yet inexplicable phenomenon by relating electromagnetic and gravitational forces. Other scientists are convinced that an entirely new discovery in fundamental physics is necessary for a full understanding of gravity. The following paragraphs will briefly summarize some current theories to account for the reality of gravitational attraction and the possibility of gravity repulsion.

Physicists generally assume a relationship between electromagnetism and gravity because both obey the inverse-square law which says that the force of both fields decreases with distance in the same mathematical relationship $\left(F = G \frac{M_1 M_2}{d^2}\right)$. Both forces also seem to propagate with the speed of light, though this has never been fully demonstrated with respect to gravity propagation. There are some intriguing differences between the two forces, however, since electromagnetism consists of two identifiable components, an electric field and a magnetic field, while gravity appears to have only one component. In addition, electric charges can repel and attract while gravity always attracts. In conclusion, as the French mathematical physicist Borel wrote:

"There was, however, something rather strange in this phenomenon of gravitation, something that distinguished it from other physical phenomena. This was its utter immutability and its absolute independence of all external actions. Light is arrested by opaque bodies, deviated by prisms and lenses; electrical and magnetic actions are modified by the neighborhood of certain bodies; gravitation alone remains always the same, and we have no means of enabling us to either increase or diminish it. Gravitation is indifferent to all physical circumstances, and it is not affected by the chemical nature of bodies" (Ref. II-65).

The above statements would seem to indicate that control of gravity may always be beyond the reach of man. However, recent discoveries including Dr. Weber's detection of gravity waves and Dr. C. Leiby's determination that a second gravitational component may actually exist (Ref. II-66), leads one to believe that man may eventually be able to predict gravitational radiation as he now predicts electromagnetic radiation.

One such device which uses electromagnetic analogies to create non-Newtonian gravitational forces would contain accelerated masses whose mass flow is like the current flow in a wire-wound torus. In the electromagnetic model, the current (I) through the wire causes a magnetic field in the torus. As the current increases, the magnetic field increases also and creates a dipole electric field. The value of the electric field at the center of the torus is:

$$E = \dot{B} = + \frac{d}{dt} (\mu N I r^2 / 4\pi R^2)$$

where R is the radius of the torus, r is the radius of one of the loops of wire wrapped around it, μ is the magnetic permeability, and N is the number of turns.

If the wires were replaced with pipes carrying a massive liquid, then the analogy between electromagnetic and gravitational fields can be used to obtain the formula:

$$G = -\dot{K} = -\frac{d}{dt}\left(\frac{\eta N T r^2}{4\pi R^2}\right)$$

where G is the gravitational field generated by the total mass current NT and $\eta = 3.73 \times 10^{-26}$ m/kg is the gravitational equivalent to magnetic permeability (Ref. II-67).

From the above equations it is fairly obvious that if such a device is ever to be practical, matter with densities of 10^8 to 10^{15} gm/cm^3 will be required. It is also mandatory that the magnetic permeability of the material be highly nonlinear, similar to the highly nonlinear permeability of iron which allows the construction of efficient electromagnetic field generators. To determine whether such materials do exist, it may be necessary to use the gravity wave detection devices developed by Dr. J. Weber and investigate which materials change the amplitude and direction of gravitational fields. Dr. Weber has already detected the gravity waves emanating from catastrophic celestial phenomena.

The above device would require some major breakthroughs in materials and assumes that non-newtonian gravity forces can be generated using Einstein's general theory of relativity; however, no new or radical change in fundamental physics was required.

The second concept, gravity absorption, will require the development of new fundamental physical laws. In the following paragraphs, the general theories surrounding gravity absorption will be discussed along with some experimentation which has been done in this area. No physical device to utilize the concept will be described since there are still too many uncertainties about the basic theories. According to the hypothesis, all space is permeated with gravitons which move with great speed and can almost freely penetrate matter. The majority of particles pass through the matter without loss of momentum, and an insignificant number are either completely absorbed or undergo elastic reflections. In the presence of two bodies (A and B), the stream of particles from body B, impinging on A, is attenuated by absorption within body B. Therefore, the surplus of flux striking body A from the outer side drives body A to body B. Whether the gravitons are hard particles (such as cosmic rays) or whether they are a sort of electromagnetic radiation is still undecided.

An experiment was conducted to determine whether a barrier could effectively reduce the gravitational attraction of the earth on some body. During this experiment an effort was made to determine the gravity absorption coefficient. In the test a lead sphere was introduced into a hollow sphere of mercury and the weight of the lead sphere decreased by 10^{-6} grams, which is the equivalent of liberating twenty million calories of gravitational energy (Ref. II-65). Many attempts were made to refute this test, but none have been completely successful. However, much more work needs to be done in this area before the hypothesis should be accepted or discarded.

Shortcomings: It is fairly obvious that both of the concepts described above are sorely lacking in details concerning the theory and practical application. This does not mean that nothing can be done to fill the voids of our knowledge, and neither concept should be ridiculed because major breakthroughs could occur and make either concept realistic.

Most of the concepts' weaknesses are in the area of technical limitations. In the case of the first device, the question of mass density and permeability will require years to unravel. Methods must be found to manufacture, contain and control matters with densities from 10^8 to 10^{15} gm/cm^3. The Fermi energy will limit the densities of even thermal neutrons to 10^{-6} gm/cm^3 unless bosons can be formed by the formation of tetraneutrons.

The weaknesses of the second concept are almost too numerous to list. The very concept of gravitons permeating all space is a yet unproven concept, and the type of materials that would be necessary to absorb the gravitons in sufficient numbers to propel a vehicle is completely undefined. Much more work must be done to fully demonstrate the theory before any attempt can be made to develop a usable vehicle.

Conclusions: The feasibility of both concepts must be proven before any work on hardware is initiated. The second concept should be attacked first, since it is a radical new idea which could be proven by some relatively inexpensive experiments. In addition, the second concept, if the theory can be substantiated, offers a more practical device for space travel. It does not appear at this moment that any work is being done on gravity absorption devices, but the concept should at least be investigated. The first concept, though more easily understood and acceptable to technologists, is many years into the future. Even if matter with the density of a white dwarf star could be accelerated around a torus a half mile in diameter, the gravity of the earth would be repelled only for a few milliseconds. This sounds very pessimistic, but major breakthroughs could be made in superdense materials before the end of this century which would allow this device to be practical.

CHAPTER II-3. PHOTON PROPULSION

Photon propulsion uses light pressure to provide thrust. Light provides very high specific impulse, but concepts envisioned today have low thrust-to-weight ratios. The two following concepts are probably representative of photon systems as we now see them. It seems probable that new unique concepts may arise in the future, making this category of propulsion concepts much more attractive.

Title: Antimatter (Photon Rocket)

Concept: This concept utilizes the energy available from mass annihilation for propulsion.

Attributes: The propulsion advantages associated with antimatter are obvious. Mass annihilation provides the greatest energy release known to man and the highest specific impulse of any concept which expels mass to provide a thrust.

Description: A number of antimatter concepts are envisioned to take advantage of the products resulting from a reaction between matter and antimatter.

1. Annihilation products may be used directly by acceleration with electric and magnetic forces (consider the annihilation reaction of a neutrino η with an antineutrino v, yielding a proton p^+ and an electron e^-).

2. Annihilation products may be used indirectly to heat a suitable working fluid for thermal expansion through a nozzle (consider the annihilation reaction of hydrogen and antihydrogen, leaving high-energy γ rays).

3. Antimatter presumably possesses negative gravitational mass, although its inertial mass may be positive. This situation would permit repulsive forces between bodies and give rise to antigravity propulsion.

4. Annihilation products of ordinary quanta give rise to the possibility of a "photon-drive" propulsion system. This system would direct through some sort of reflecting device a beam of light for thrust.

Analysis: To attain awesome velocities near that of light from a propulsion device, complete conversion of matter to energy is needed.

This complete conversion to energy results when atomic particles undergo mutual annihilation with other particles. Because of this annihilation property between particles, one particle is termed the antiparticle of the other. Antimatter consists of matter made up of antiparticles. For example, ordinary matter consisting of neutrons, protons and electrons becomes antimatter consisting of antineutrons, negatrons (anti-protons) and positrons (anti-electrons).

A detailed analysis of an antimatter propulsion system is hardly warranted and necessarily awaits the disclosure of future particle research. However, performance estimates give a thrust/weight of about 10^{-7} and an $Isp = c/g_0 = 3.06 \times 10^7$ lbf-sec/lbm for a possible vehicle configuration (Ref. II-69).

Shortcomings: Identifying potential weaknesses of any antimatter propulsion system is difficult because few engineering details are available. Nonetheless, some appreciable concern may be expressed at the difficulty of:

1. Producing Antimatter - For propulsion application, a mass flowrate of the order of gms/sec of antiparticles is desired rather than the few dozens of antiparticles produced in present machines.

2. Storing Antimatter - Experiments with antiparticles are complicated by their tendency to explode with a burst of energy when they contact normal matter. Containers made of force fields or some other isolating technique are necessary.

Title: Solar Energy for Propulsion

Concept: The solar sail seems to be the only scheme surviving where utilization of solar energy for propulsion is proposed. At any rate, it seems to be the only one of some current interest that is documented to any degree. The basic idea is for a vehicle in space to deploy a sail so as to intercept solar radiation. Radiation pressure on the sail will result in a thrust on the vehicle.

Attributes: Since solar energy is being utilized for propulsion, no propellant need be carried on board the spacecraft. The amount of energy available is unlimited, and the energy is continuously available.

Description: The space vehicle is assumed to be in an orbit around the earth. The sail is then deployed, with the spacecraft attached by shroud lines. The length of these can be controlled from the spacecraft. The configuration resembles that of an object suspended from a parachute. To change orbit, or to escape from an initial earth orbit into a solar orbit, one manipulates the shroud lines so that the sail experiences the full solar radiation pressure when the vehicle is travelling away from the sun; and so that the sail experiences zero (minimum) pressure when it is travelling toward the sun, the sail is collapsed or furled during that portion of the orbit when the spacecraft is moving toward the sun (Figure II-42). In this manner, the orbital velocity is gradually increased and the orbit changes. Eventually, the velocity becomes large enough to allow the spacecraft to escape into a solar orbit.

Similar operations would allow a solar sail spacecraft to reduce its orbital velocity, and thereby change to an orbit closer in to the surface of a given planet - even land.

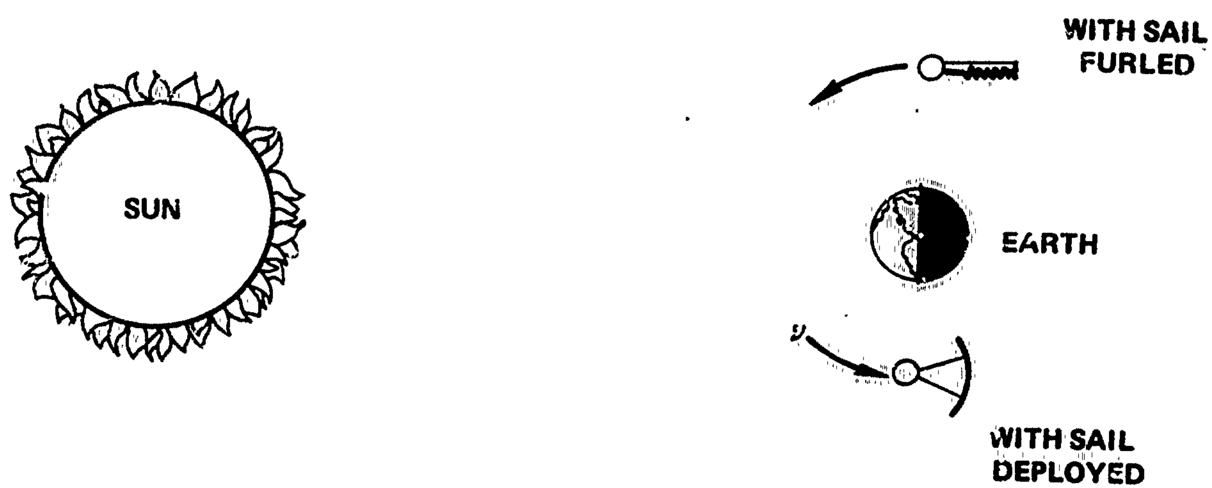


Figure II-42. Maneuvering with the Solar Sail (Not to Scale)

Maneuvers are also possible with a continuously deployed sail (i.e., without having to collapse the sail on approximately one-half of each orbit). This could be accomplished by rotating the sail so that the sail area normal to the radius vector from the sun is alternatively maximum and minimum (on the appropriate portions of the orbit).

Maneuvering in the solar orbit is carried out in the same way. To get to any planet within the solar system, the orbit of the spacecraft is changed so that the new orbit coincides at least partially with the orbit of the target planet. The spacecraft can then be captured by the local gravitational field and a landing on the planet can take place.

Note that motion toward the sun can result in only one way. The spacecraft in a solar orbit reduces its orbital velocity by sail manipulations, allows the gravitational force to overtake the centrifugal force, and thus allows the spacecraft to "fall" toward the sun.

Analysis: The idea of using energy radiated by the sun for propulsion of space vehicles appeared in science fiction literature in 1951 (Ref. II-70). The concept made its entrance into scientific literature in 1958 (Ref. II-71). The general conclusion was that the solar sail concept is valid and that the sail might be competitive thrust-wise with other propulsion schemes (namely ion and photon drives). A more recent publication (1969) in this area proposes to use the sail with radiation originating in a huge laser bank which has been built in a solar orbit (Ref. II-72).

Solar radiation can be described by the empirical formula:

$$S = \frac{3.1 \times 10^{25}}{R^2}$$

S is the energy flux in watts/m^2,
R is the distance from the sun in meters.

At a distance equal to the radius of earth's orbit (1 AU), the above formula yields about 1400 watts/m^2. This then is a typical number for the solar energy available.

The pressure P (newtons/sq meter) exerted by solar radiation on a sail will depend on the energy flux, on the angle between the flux direction and the sail surface and on the nature of the surface. For a non-reflecting surface at normal incidence, this pressure is given by:

$$P = \frac{S}{c}$$

where c is the velocity of light (3×10^8 m/sec). A perfectly reflecting sail will experience twice the above pressure.

Intuitively, one expects the radiation pressure to be somewhat small. The sails must consequently be as light as possible. A typical candidate for sail material is a 2×10^{-4} cm (0.1 mil) thick plastic (such as mylar) sheet which would be aluminized on one side to provide a reflecting surface. Surface density of the resulting material might be 5×10^{-3} kgm/m^2. Assuming for the moment that the sail constitutes one-half of the total mass of vehicle, the acceleration which could be realized is given by:

$$A = \frac{P}{2(5 \times 10^{-3})} = 0.5 \times 10^{-3} \text{ m/sec}^2 \text{ at 1 AU}$$

This is about 5×10^{-5} that of gravity; i.e., the thrust to weight ratio is 10^{-5} to 10^{-4}. Constant acceleration of this magnitude would result in a velocity of 10^4 m/sec in approximately six months. This is the order of magnitude of the escape velocity from earth. Lighter and thinner materials would of course reduce this time; however, the calculation does illustrate the time scale one deals with in solar sailing.

Finally, it is clear that escape from planets nearer to the sun than 1 AU would take less time than the 6 months estimated above for escape from earth's orbit, both because of greater radiation pressures and smaller planetary gravitational fields. Likewise, escape from planets further out than 1 AU would take considerably longer, since the radiation pressures are smaller and gravitational fields larger. Interplanetary trips would necessarily be measured in years.

Shortcomings: The times required for maneuvering operations are extremely long. This objection cannot be disputed but is common to all low-thrust propulsion schemes.

Lifetimes of the sail may not be long enough. The lifetime of the sail is usually assumed to be infinite, and this is commonly listed as a very desirable property of the sail. Erosion rates of surfaces in space are estimated to be as high as 3×10^{-4} cm/yr (due to cosmic ray and solar wind sputtering, sublimation, cosmic dust, and meteorite impact).

In this case, the sail would not last for a year. Certainly, the reflecting surface can be adversely affected.

Maneuvering is unquestionably limited. One has to rely on the slow orbit changes to get from one point to another. The thrust is limited in direction. Because of the "push-pull" nature of this, there is no possibility of any evasive maneuvers (from meteors, for instance).

Other forces on the sail might have to be considered. Solar wind pressure at 1 AU is two or three orders of magnitude lower than radiation pressure. But because solar wind protons can be guided by interstellar magnetic fields, this pressure might become comparable to the radiation pressure further out from the sun. Normally radiation intensity is falling off as $1/R^2$, whereas solar wind density need not fall off at all because it depends on the magnetic field. Thus, the solar wind could exceed the radiation pressure.

The most important practical objection is the enormous size of the sail. The sail areas mentioned in the literature vary from 10^6 to 10^2 m^2 – a disk, say, of 1 to 10 km radius! For a sail of 20,000 kgm and surface density of 5×10^{-3} kgm/m^2, area = 4×10^6 m^2; at 1 AU, the maximum thrust on such a sail would be approximately 36 newtons (thrust/weight = 9.2×10^{-5}) which is less than 10 pounds. This huge sail must be controllable – one must be able to orient it toward or away from the sun. The sail material itself will not be rigid enough to transmit these forces. It is also debatable whether the sail will "keep its shape", so to speak, at other than normal incidence. It appears that some sort of a supporting structure for the sail (a "backbone") will be necessary. This, of course, means more weight and probably many more problems in packaging and deployment of the sail.

Conclusions: The solar sail concept has a certain amount of appeal which probably originates in its simplicity and the fact that it is a basically correct concept. On the other hand, its inherent simplicity is somewhat disappointing in that there are no emergent characteristics – the basic ideas are well understood, and that's all there is to it. The only way to increase thrust is to increase the area of the sail. The limitation imposed by the fixed amount of radiation available cannot be circumvented. Perhaps this is the reason why there does not seem to be any substantial effort in propulsion oriented toward the use of solar sails.

The most that can be foreseen for the sail is for it to function as a secondary propulsion scheme on some future interplanetary (long) mission. The sail would be of a reasonable size (100 m radius), and most probably would require very little manipulation. Main propulsion, escape from planetary orbits and maneuvering would be done by the primary propulsion scheme; the sail would be deployed only when already on the desired course. This would allow some saving of propellant while minimizing structural and orientation problems of large sails.

This would not be to say that solar power cannot be utilized. The statements in the previous paragraphs are meant to apply to sails as a primary propulsion scheme. Certainly solar cells have already proved their worth. In the same line, other electrical energy storage devices and energy conversion devices which provide limited electrical power for spacecraft systems are a very promising field. An upper limit of 10 to 20 KW is estimated for these devices, dictated again by a reasonable collector area and the weight of the storage or conversion device. In addition, solar energy can be utilized effectively for stabilization or orientation functions.

Recommendations: Solar energy is unquestionably very important to most space systems, but as a primary propulsion scheme its future seems limited. Consequently, organizations with interests primarily in station-keeping and stabilization-orientation of spacecraft systems will find solar sails a fruitful field of endeavor.

CHAPTER II-4. UNIQUE CONCEPTS

This concept is discussed separately since it did not logically fit any other category. It is unique inasmuch as it would require a material commitment on the order of the Space Shuttle program to be realized. It is in essence a new "total system" approach to space operation. It involves specialized propellant collection, processing and distribution systems, subsystems and transport vehicles.

Title: The Air-Scooping Orbital Rocket (A-SCOR)

Concept: With present space booster techniques, a large portion of the total mass which is initially placed into low earth orbit often consists primarily of propellants or of propulsive fluids for subsequent satellite or space vehicle maneuvering. When this is the case, the actual mission payload mass is frequently much smaller than the propellant mass.

The essential feature of the A-SCOR (Air-Scooping Orbital Rocket) or PROFAC (Propulsive Fluid Accumulator) concept is to lift only the energy source into orbit at approximately 100 to 150 km, and at that point to collect air as the basic propulsive fluid for further satellite maneuvering or for continuing the journey into space.

The basic essentials of the A-SCOR or PROFAC concept were first proposed by S. T. Demetriades in 1958, and a very substantial portion of the work which has been done since then has been based upon his pioneering work in this field.

Attributes: The primary advantage of such as the A -SCOR propulsion scheme is that the orbiting vehicle would have little or no need for on-board propellants. The A-SCOR vehicle would be capable of remaining in orbit indefinitely and of obtaining all or most of the propellants and any needed life-support oxygen supplies for itself and/or for other space vehicles directly from the upper reaches of the earth's atmosphere.

The basic A-SCOR propulsion concept could be used in conjunction with an extremely wide variety of chemical, nuclear and electrical rocket propulsion systems. For its basic electrical power source, an A-SCOR vehicle might very well use surplus power from nuclear or solar power supplies which would be required aboard the spacecraft anyway for purposes other than propulsion.

In regard to cost, the use of an A-SCOR propulsion system could result in very substantial savings. If used entirely on its own in conjunction with currently available space boosters, it could reduce the cost of placing space payloads in orbit by approximately one order of magnitude. If used in conjunction with the reusable space shuttle system, it could reduce this cost by two or three orders of magnitude (from the current cost of approximately $500 per pound to the general vicinity of 5 dollars to 50 cents per pound). These last figures would then begin to approach the "real limits" of approximately 2 cents per pound quoted by A. Kantrowitz in the March 1971 issue of Astronautics and Aeronautics (Ref. II-73).

Description and Analysis: Available evidence indicates that all of interplanetary space contains small but measurable quantities of matter which might be scooped up by a space vehicle and used for propulsion and perhaps for other purposes.

The basic equation for scooping or collecting this matter is:

$$\dot{m} = \rho A_c V \qquad (1)$$

where $\dot{m}$ = the rate at which mass is collected,

ρ = density of the matter being collected,

A_c = effective area of the collector, and

V = velocity of the space vehicle.

The first objective in this analysis will be to determine those regions of space in which propellant scooping operations are likely to prove feasible in the reasonably foreseeable future and to eliminate from any further serious consideration those regions in which only minimal returns can be anticipated for any time and resources invested in a p pellant scooping research and development program.

Let us assume for the moment that A_c = 1 square meter and then concentrate our attention on ρ and V, the other two factors on the right-hand side of equation (1). The density of interstellar matter (plasma) in the vicinity of our solar system is approximately 10^{-20} kgm/m^2. This density is so low that, even if a space vehicle were to achieve a velocity equal to approximately 90% of the velocity of light, the rate of mass collection would still be only about 2.7×10^{-12} kgm/sec.

This collection rate is so small and the velocity requirements so severe that it would seem reasonable at this point to drop completely, at least for the next few years, any further consideration of a pure space ramjet operating in the interplanetary environment at any considerable distance from a planetary atomosphere. "Electromagnetic scooping"--with an effective scooping radius of perhaps as much as 100 or more kilometers --might become feasible in this region at some time in the relatively distant future when very advanced nuclear or thermonuclear power supplies might become available.

We now find ourselves -- at least for practical military purposes -- confining our attention to that region of space in the immediate vicinity of the planet earth. However, even here the atmospheric densities at representative orbital altitudes of probable primary interest (approximately 100 to 150 km, or approximately 54 to 81 n. mi) are such that it becomes advisable to confine our analysis to space vehicles which are capable of conducting propellant-scooping operations over relatively extended periods of time. Thus, the very general concept of the space ramjet is quickly reduced by practical military considerations to the much more specific concept of the Air Scooping Orbital Rocket (A-SCOR) or of the Air Scooping Orbital Ramjet. If the vehicle under consideration does not actually achieve orbital velocity, then its propulsion system would more logically fall into the much simpler category -- from the conceptual point of view -- of the air-augmented aerodynamic rocket or of the ordinary suborbital atomospheric ramjet/scramjet. The "recombination ramjet", which makes use of the energy stored in dissociated atmospheric oxygen and nitrogen, is a special case which has been studied before, and it will not be covered in this evaluation.

The ultimate design of an A-SCOR propulsion system for a relatively large space vehicle will very probably include both of the following:

a. A high-thrust, low-specific-impulse main propulsion system for any quick changes in orbital parameters required by the specific mission for which the vehicle is designed. This propulsion system would probably be either a LO_2/LH_2 chemical rocket or a nuclear rocket. There is a possibility that a combination of political and technical considerations might make the LO_2/LH_2 chemical rocket a tentative first choice over the nuclear rocket, at least for the initial A-SCOR feasibility demonstration.

b. A high-specific-impulse, low-thrust secondary propulsion system for air-scooping, for atmospheric drag cancellation, and for most orbital changes in which propellant conservation is a more important consideration than rapid maneuvering. This secondary propulsion system would probably be some type of electrical propulsion system because of the obvious requirement that the propellant exhaust velocity must be considerably in excess of the vehicle velocity for any substantial practical benefits to accrue from continuous propellant scooping operations.

The selection of an actual operating altitude for any specific A-SCOR mission would require some very careful systems analysis tradeoff studies. To illustrate this, an A-SCOR vehicle with an effective collector area of 1 square meter would be capable of accumulating approximately 570 kgm (about 1250 lbs) of air per day in a 100 km (54 n. mi) circular orbit. However, the air-scooping capability for this same A-SCOR vehicle would decrease very rapidly to approximately 2.4 kgm (about 5.3 lbs) per day in a 150 km (81 n. mi) circular orbit.

Since the rate of mass accumulation is directly proportional to the effective area of the collector, a very substantial increase in the collection rate could be achieved for any given air-scooping altitude simply by increasing the effective area of the collector from the relatively small 1 square meter which we have previously assumed. Another obvious possiblity, of course, would be to increase the total mass accumulated by simply remaining in an air-scooping orbit for whatever extended period of time might be required to obtain the total propellant mass necessary for accomplishment of the specified mission.

A more detailed description and analysis of various aspects of the basic A-SCOR concept may be found in the references which are included with this evaluation (Ref. II-74 through II-82). To obtain some idea of the relative magnitudes involved in a large A-SCOR system, the following example is offered for a system of the type initially proposed by Demetriades. For this example, an air-scooping vehicle in a 120 km circular orbit will be assumed. To overcome drag, the vehicle will be assumed to be equipped with some type of electrical propulsion system. The total energy required to collect a given mass of air is found to be independent of altitude and is given by the following equation:

$$\frac{P.\Delta t}{\Delta m} \cong \frac{k}{\eta} \frac{V_o^2 A_i^2}{A(A_i - A)} \frac{\text{kw-sec}}{\text{lbm}}$$

Where P = the required electrical power in kw,

Δt = the total collection or scooping time in seconds,

Δm = the total amount of mass collected in lbm,

k = a constant with a value of approximately, 0.21,

η = the efficiency with which the electrical energy input is converted into kinetic energy of the exhaust,

V_o = the orbital speed of the vehicle in ft/sec,

A_i = the total inlet area in ft^2, and

A = that protion of the total inlet area in ft^2 which collects air for drag cancellation by the electrical propulsion device.

If we now assume that the collection interval Δt and the total amount of air to be collected Δm are both fixed, then we can rearrange equation (2) to obtain the following specific equation for the required electrical power:

$$P \cong \frac{k}{\eta} \frac{\Delta m}{\Delta t} \frac{V_o^2 A_i^2}{A(A_i - A)} \tag{3}$$

To obtain the optimum relationship between A_i and A for the minimum power requirement at any given orbital velocity, we can now find $\frac{dP}{dA}$ and set it equal to 0 with the following result:

$$\frac{dP}{dA} = \frac{k}{\eta} \frac{\Delta m}{\Delta t} V_o^2 A_i^2 \frac{2A - A_i}{A^2 (A_i - A)^2} = 0 \tag{4}$$

or

$$A = 1/2 A_i \tag{5}$$

Thus, minimum electrical power is required during actual air-scooping operations when one-half of the incoming air is used by the electrical propulsion system for drag cancellation and the other half is collected and stored.

If we now assume that the A-SCOR vehicle [illegible] signed that frictional and wave drag is negligible by comparison with momentum drag, then equation (5) also implies that the optimum exhaust velocity $(v_{ex})_{opt}$ during scooping operations is approximately twice the vehicle's orbital velocity, or

$$(v_{ex})_{opt} = c_{opt} = 2V_o \tag{6}$$

$$= 2\ (26,000 \text{ fps}) = 52,000 \text{ fps}$$

This optimum exhaust velocity corresponds to an optimum speicifc impulse of

$$(I_{sp})_{opt} = \frac{52,000}{32.2} = 1600 \text{ seconds} \qquad (7)$$

If we had used somewhat less optimistic assumptions in arriving at this figure, the optimum specific impulse during circular orbit scooping operations would actually turn out to be approximately 2000 seconds. Also, after the air-scooping operations have been completed, the specific mission of the A-SCOR vehicle in many cases might very well call for an optimum specific impulse substantially higher than 2000 seconds. These values of specific impulse clearly indicate the need for some type of electrical propulsion device aboard the A-SCOR vehicle, at least during that period of time when actual propellant scooping operations are being conducted. The specific types of electrical propulsion systems which might be considered for use aboard an actual A-SCOR vehicle include electrical or MGD (magnetogasdynamic) ramjets, electrostatic rockets, and electromagnetic rockets. These relatively low-thrust propulsion systems would, of course, be in addition to any required high-thrust chemical or nuclear propulsion systems.

To get some idea of specific preliminary design values for an actual A-SCOR vehicle, the following figures are presented from a preliminary study made by S. T. Demetriades (Ref. II-82). This particular vehicle design is shown schematically in Figure II-43.

Total scooping or collection time (Δt):	100 days
Total mass of air collected (Δm):	100,000 lbm
Length of vehicle:	120 ft
Weight of vehicle (dry):	76,000 to 92,000 lbs
Required radiator area:	12,000 ft

Power required:	4-6 MW
Altitude of operation:	65 n. mi
Area of MGD-orbital ramjet inlet:	795 ft^2
Area of air-scooping inlet	170 ft^2

As can be seen from the general magnitude of the above numbers, the specific A-SCOR vehicle considered in this particular example is very substantial in size and would undoubtedly require a rather considerable investment of research and development resources. However, it represents only one possible end product of considerable preliminary research, analysis, design, and testing. Initial A-SCOR feasibility tests would very probably involve much smaller vehicles and could be initiated with a relatively modest investment of manpower and other R&D resources. To illustrate this last point, a space launch vehicle no larger than a THOR-AGENA or an ATLAS-AGENA would probably prove entirely adequate for preliminary feasibility testing of the basic concept of air-scooping at orbital altitudes and speeds. For this purpose, relatively simple atmospheric-grazing vehicles placed in elliptical orbits and employing non-nuclear electrical power supplies should be able to provide a convincing demonstration of the very substantial practical advantages inherent in the basic idea of air-scooping orbital vehicles.

Numerous unclassified publications have indicated that the Soviet Union has invested substantial resources in the development of an A-SCOR type vehicle. These reports claim that they have already achieved considerable success in both orbital and suborbital air-scooping propulsion programs. The following excerpts from an article in the 21 October 1969 issue of the International Herald Tribune are reasonably representative of the type of statements which the Soviets have made in regard to their A-SCOR propulsion program:

"The Russians today claimed development of a space engine which sucks its fuel from the upper atmosphere."

"The engine would not only have major uses for spacecraft in low terrestrial orbit but could also be fitted to high-flying supersonic aircraft Soviet scientists suggested."

"TASS said the plasma-jet, or ion, engine has developed a thrust velocity of 74 miles a second by using atmospheric nitrogen at heights between 62 and 248 miles. . . ."

"The Russians assert the use of atmospheric nitrogen represents a breakthrough since spacecraft would not have to carry fuel aloft."

"The development also has the valuable side effect of extracting liquid oxygen from the atmosphere, which could have considerable importance in supplying the artificial atmosphere for crews of spacecraft, the Russians said."

"The engine apparently is designed for use in the ionosphere around the earth and not in the vacuum of outer space. . . ."

"Various gases have been used in experiments with the engines. . . . The first Soviet spacecraft for upper atmosphere experiments was Yantar-1 (Amber-1), launched in October 1966 which used argon gas and achieved a jet speed of 28.8 miles per second."

"The latest experiments, TASS said, concentrated on using the fuel available in the ionosphere. Once the engine has been given an initial acceleration, it extracts oxygen and nitrogen from the atmosphere, ionizes the latter, and with a recently improved neutralizer produces a jet thrust of 75 miles per second."

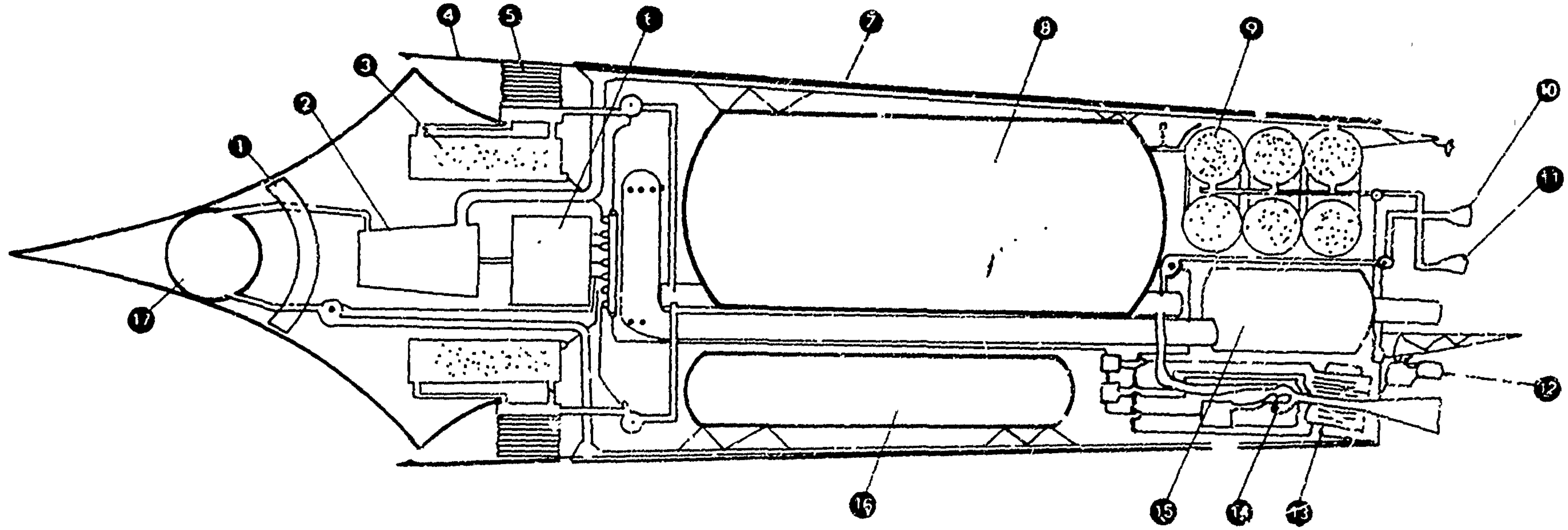
"The report spoke of several 'Amber' flying laboratories. These are believed to have been launched under the catch-all Cosmos label of unmanned satellite research."

Shortcomings: The primary problem is that to date no organization in the United States has been given sufficient financial backing to make a detailed engineering analysis of individual A-SCOR components and of overall systems integration.

Conclusions: The primary conclusion is that the basic A-SCOR concept is theoretically sound and requires no fundamental scientific breakthroughs.

The A-SCOR propulsion system could be used in conjunction with a very wide variety of rocket propulsion concepts. These concepts could be used singly or in various combinations and could include chemical rockets (liquid and solid), nuclear rockets, and electrical rockets (electrostatic, electromagnetic, and electrothermal). Potential A-SCOR mission applications for any nation which chooses to develop such a system would include almost all missions involving earth or planetary-orbiting vehicles. Specifically included in this category would be such missions as the raising of space vehicles from low earth orbits to synchronous altitudes, rendezvous and docking, very low level reconnaissance and surveillance, and the use of aerospace planes, space shuttles, and orbital re-supply vehicles for permanent earth-orbiting space stations.

Recommendations: The basic A-SCOR concept, when viewed in the light of its extremely wide range of possible military applications, not only for this country but for other nations as well, has sufficient potential to warrant a detailed systems analysis. To insure that all available source material is included in this study, the final report should be classified, but this classified report should also be accompainied by an unclassified version to insure widest possible dissemination of its basic conclusions and recommendations. This same recommendation applies with equal force whether or not the United States itself has any current plans or requirements for the eventual development of an American A-SCOR propulsion capability.



1. SHIELD
2. TURBINE
3. CRYOPUMP
4. RADIATOR FOR CRYOPUMP
5. REFRIGERATOR
6. GENERATOR
7. RADIATOR FOR CLOSED NUCLEAR BOILER
8. LIQUID AIR TANK
9. LIQUID HYDROGEN TANKS
10. LIQUID AIR TAP
11. LIQUID HYDROGEN TAP
12. AUXILIARY CHEMICAL ROCKET ENGINE
13. MGD DRIVER
14. PLASMAJET
15. CREW
16. LIQUID OXYGEN TANK
17. NUCLEAR REACTOR AND BOILER

Figure II-43. Schematic of Orbital Vehicle

CHAPTER II-5. SUMMARY FOR PART II

The relationship of chemical energy to rocket performance was described in Part I of this report. It should be obvious that greater sources of energy will provide higher performance possibilities. Table II-7 shows the spectrum of known energy sources.

TABLE II-7. ENERGY SOURCES

Type	Energy Release (cal/gm)
Chemical (O_2/H_2)	3.60×10^3
Free Radical ($H + H \rightarrow H_2$)	5.26×10^4
Nuclear Fission (100%)	1.70×10^{10}
Nuclear Fusion	1.82×10^{11}
Matter Annihilation	2.15×10^{13}

The next step beyond chemical energy in rocket propulsion is generally considered to be the utilization of nuclear energy. Classifying "free radicals as a form of chemical energy, it is important to notice that a discontinuity of five orders of magnitude exists between chemical-type energy and nuclear-type energy. This large order of magnitude step should serve to illustrate the reasons why nuclear-powered rockets are of interest. Based on present technology estimates, nuclear fission systems could be flying by 1980 and nuclear fusion systems before the year 2000. Although antimatter can now be produced in very small amounts, the technology required for matter annihilation rockets is presently beyond the foreseeable future.

But are there other ways? The idea of infinite specific impulse systems provides a glint of another possibility. For an impulse system to be of the infinite Isp variety, all of the working fluid expelled from the vehicle must originate in the environment. Air-breathing rockets and ramjets are examples of propulsion systems whose Isp is markedly increased over pure chemical propulsion by taking a portion of the working fluid from the environment. But they are not infinite because they still carry a portion of the propellant which is used for combustion. The infinite Isp ramjet takes both its working fluid and energy from the environment, and therefore becomes the "ideal" propulsion system. Any vehicle that can "interact" with the environment in such a way that both propulsion and energy are freely provided, must be the ultimate system.

Man is a product of his environment. His ability to envision new propulsion concepts for use in space is to some extent retarded because of his confinement to earth and the sphere of his experience. He continues to project his approaches and intuition gained on earth to the problems of space propulsion. Radical departures from time-honored, well-proved approaches are either discarded or lack visualization. Possibly, not until man truly becomes a creature of space will the restrictions imposed on his imagination be removed and radically new propulsion concepts devised. We are just beginning to understand the true nature of space and to attempt to utilize this environment for our propulsion needs. Many interactive forces are available for propulsion. A few of these forces include gravitational attraction, gas and plasma pressure, radiation pressure, magnetic fields and electric fields. Utilizing these "free" forces will be a true test of man's ingenuity.

Figure II-44 compares most of the advanced concepts that were evaluated. In cases where no system studies were made to calculate thrust/weight or specific impulse, a rough estimate was made of the concept's potential. Estimates are shown as dotted lines. In general, it should be noted that there is a downward slope in thrust/weight as Isp increases to that of eletromagnetic radiation at 3.06×10^7 seconds.

Again, as in Part I, none of the concepts evaluated showed such outstanding promise as to warrant a crash development program. Electric propulsion and nuclear propulsion are currently under development for space missions. These concepts will continue to evolve and develop as technology and mission requirements dictate. The superconducting particle accelerator and infinite Isp ramjet are two promising concepts that should be pursued to greater detail before being either rejected or accepted. The imagination of aerospace engineers should definitely be directed to infinite Isp concepts that react freely with the environment, whether through beamed energy or natural phenomena, because of its fantastic implications. The A-SCOR concept promises exceptional operational advantages to effective utilization of space. It is assumed that the potential of A-SCOR devices for space exploration will dictate their development within the next 15 years.

Obviously, advancements in certain areas of technology could make a number of concepts suddenly very attractive. Improvements in high-energy lasers by several orders of magnitude of energy output or new concepts involving long distance energy transfer would make both laser propulsion and infinite Isp ramjet very attractive. The development of higher current density superconductors, metallic hydrogen, or even room temperature superconductors would make many of the magnetic concepts more attractive. Breakthroughs in antimatter technology or antigravity theory could also revolutionize propulsion.

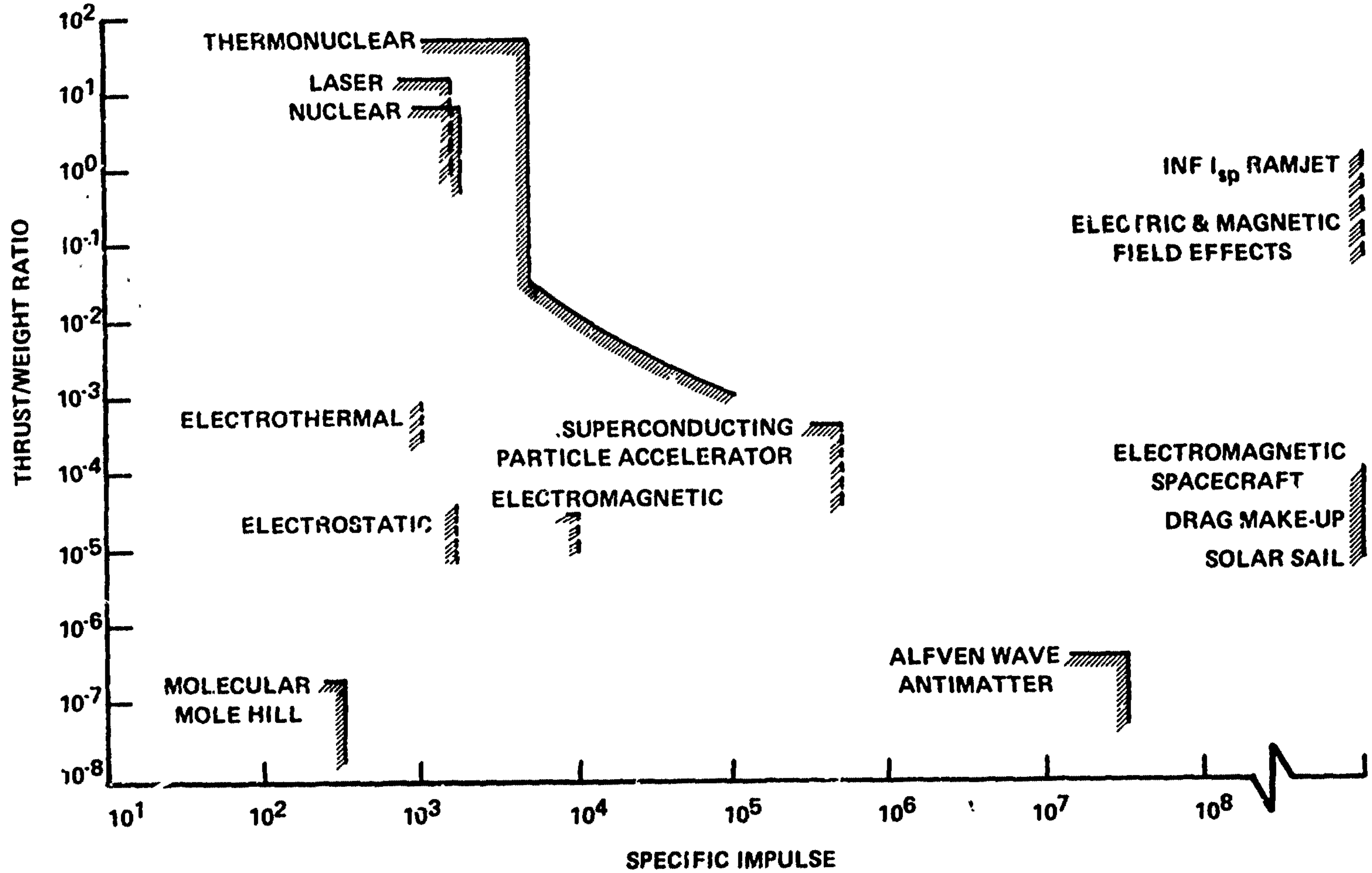


Figure II-44. Propulsion System Summary

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APPENDIX I. IDEAS SUPERFICIALLY EVALUATED

1. METALIDING - A bonding or welding process for special metals. This is a manufacturing process to be used in the production of a system.

2. NITINOL EXPULSION DEVICES - Using Nitinol materials (Ni-Ti Alloy). Liquid propellant tanks could be made to expel propellants with the addition of small amounts of heat to the tanks. Nitinol materials have a characteristic of returning to a set geometry.

3. HEAT PIPES - A relatively new and efficient heat transfer technique. These would be components of a conceptual system.

4. BALLOON-LAUNCHED ROCKETS - Launching rockets from a high altitude airborne balloon. Past studies have indicated that there is not enough gain to be realized from this technique, especially in view of the launch complexities.

5. CATAPULT LAUNCHERS - Catapulting or throwing of rockets (i.e., downhill railroad track, spin-off from a flywheel, etc.). Not enough force gained, high "g" loadings, and systems would be too complex.

6. SLUSH HYDROGEN - Increasing the density and lowering of vapor pressure of hydrogen fuel by partially freezing liquid hydrogen into a slush mixture. This concept does not give enough gain over present systems, and it is similar to the solid hydrogen concept.

7. CHEMICAL/THERMAL "RAILROAD TRACK" - "Tanker" missile lays down a stream of "track" of chemical propellants or thermal energy in space for following missile to pick up and use. The idea appears too vague and complex at present.

DRAG ENERGY - Utilizing heat created by missile drag and feeding it into the propulsion cycle via heat pipes. Idea is vague, theoretical, and probably would not give enough gain.

9. PSYCHOKINESIS - Movement of objects by mental forces. Although recent developments indicate small objects may be propelled by this technique, too much basic research is needed before this method can be considered for propulsion applications.

10. NATURAL ELECTRIC ENERGY - Tapping potential differences between the earth and clouds or between the earth and missile. Adequate techniques have not been devised to utilize this energy effectively.

11. TACHYON PROPULSION - Harnessing theoretical faster-than-light particles. Tachyons have not yet been detected.

12. COANDA EFFECT NOZZLE - A reverse flow nozzle. Test results are negative.

13. DEAN DRIVE - Using "differences in energy" between two decelerated objects to accelerate an entire system. Idea not fundamentally sound.