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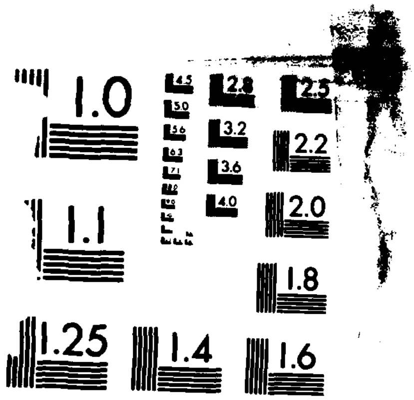
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Interim Report
for the period
October 1984 to
September 1985

Plasma Thruster Development: Magnetoplasmadynamic Propulsion, Status and Basic Problems

February 1986

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FOREWORD

This interim report was prepared for the Air Force Rocket Propulsion Laboratory (AFRPL) under contract AFOSR 84-0394 by the University of Stuttgart, Federal Republic of West Germany. The study was performed during the period October 1984 to September 1985. Project Manager for the AFRPL was Lt Robert D. Meya.

This technical report has been reviewed and is approved for publication and distribution in accordance with the distribution statement on the cover and on the DD Form 1473.

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This report provides an ~~introductory~~ overview of the present state ~~of development~~ and the ~~unresolved~~ problem areas of magnetoplasmadynamic (MPD) space propulsion devices and systems potentially suitable for orbit raising and maneuvering of large space structures, i.e., thrust and power levels of presently most promising steady (or quasisteady pulsed) thruster types, self-field ~~(SF)~~, axial applied-field ~~(AAF)~~ arcjets and Hall-ion ~~(HI)~~ thrusters are reviewed in terms of performance trends, present understanding, uncertainties and ground testing problems, with the conclusion that there is as yet no reliable basis for choosing one type for final development. Typical efficiency vs. specific impulse curves for all MPD thrusters show the dominant role of the propellant on the thrust to input power ratio (as for ion engines), and the necessity of achieving relatively high specific impulse values for each propellant for acceptable efficiencies. The specific impulse (or "onset") limit of self-field thrusters using a variety of propellants is discussed, together with predicted trends from several theoretical models. Considering....

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Block 19 (continued): system complexity and development cost as well as performance, the advantages and disadvantages if pulsed (quasi-steady) vs. continuous thruster operation and the propellant selection criteria are reviewed. Electrode erosion, especially on cathodes, losses and limits imposed by radiation cooling are emphasized as critical problem areas for larger MPD thrusters. Besides the electrode attachments, the unresolved basis theoretical problems of efficient plasma acceleration and of the midstream flow discharge stability are identified, and proposed approaches towards gaining additional understanding are outlined.

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Introduction

With the accomplishment of the successful space shuttle transport system, operation plans for a space station and other large structures in near-earth orbits with correspondingly large electric power supplies have become financially as well as technically realistic for the coming decade. The requirement to raise, maneuver and stabilize these large payloads has reemphasized the future need for electric propulsion systems of substantially higher thrust and power levels than the ion engine or the pulsed ablation plasma thruster systems.

For thrust levels of several newtons and power levels of a hundred kilowatts or larger it is natural to reconsider the magnetoplasmadynamic (MPD) thrusters again as future propulsion candidates.

The potential advantages of MPD thrusters compared with electrostatic ion propulsion systems for the applications considered are believed to be:

- Compactness and ruggedness of the thrusters.
- Reduced complexity, weight and cost of power conditioning (at least for continuous thrusters).
- Competitive system performance with acceptable propellants for the mission-optimal specific impulse range.

For many near-earth missions the cost-optimal specific impulses will be in the range of 1000-3500 s, the system efficiencies should be above ≈25% and the thrust-to-power (or efficiency to specific impulse) ratio as high as possible, according to some recent transport cost and rate studies.[1-4]

At present, these presumed advantages are based on rough estimates only, since no flight systems of the thrust, specific impulse and life levels required for the applications considered here exist anywhere to the writer's knowledge. Additionally, not all of the advantages mentioned apply equally to all the plasma systems under consideration. Also, very significant progress has been achieved recently on ion engine performance (thrust-to-power ratio) in the lower specific impulse range with noble gas propellants and on reducing their power processor complexity and weight. The potential advantages of the MPD propulsion systems must therefore be reevaluated as the necessary data on thrusters and systems become available.

During the past decade in the U.S. and in Western Europe, work on kilowatt level plasma thrusters had been reduced to a few research projects, with no flight hardware in sight except for the small (millinewton) pulsed teflon thrusters which have been in service since 1968. Only in Japan and in the Soviet Union have development efforts on kilowatt-sized plasma propulsion systems led to flight hardware.

With definite requirements within the next two decades for larger electric propulsion systems in sight, a substantially increased research and prototype development effort on plasma thrusters and systems in the U.S. and Western Europe is now required to furnish the basis for final system development decisions (which represent large financial commitments).

At present, there is not a consensus among the experts on which MPD thruster type the technology and demonstration prototype development efforts should be concentrated. Very remarkable progress was achieved in the past few years with pulsed self-field plasma thrusters-- quasi-steady ones using electrodes as well as a short-pulse inductive type. In the older literature, there are some equally promising performance data on applied-field thrusters. However, all of these results are, in this writer's opinion, still beset with uncertainties, both concerning their validity for a final flight system and concerning unresolved component life and other systems problems. In addition, the trading factors between system dry weight and cost (e.g. for pulsed system power conditioning) and thruster performance are not yet available for the model missions considered. Therefore, it appears that increased work on the technologies of several thruster types and other decisive system components must be pursued in parallel for some time together with the application and cost studies.

This position paper is an attempt to furnish an objective overview of the present state of development of newton level plasma propulsion as an aid to the study of more detailed papers on the individual thrusters, electrode phenomena, etc. and on electric propulsion application studies. The emphasis will be on performance trends and factors which dominate them (such as the propellant selection), on comparisons and on present technical limits. The, in the author's opinion, currently still large uncertainties in many of the most promising experimental data will be evaluated as well as possible. No attempt is made here to summarize and evaluate in detail the large body of thruster performance data and system component weight and cost estimates found in the literature. However, the preparation of an up-to-date summary and critical evaluation of the more recent experimental and (sparce) theoretical results on thruster performance, to supplement the existing excellent older summary papers such as Refs. 5 to 11, is strongly recommended.

In identifying and evaluating the importance of unresolved basic theoretical and experimental problems, the author expresses his opinion where (and to some extent how) future research effort could contribute to the successful development of cost effective plasma propulsion systems.

Brief Status Summary of Plasma Propulsion Systems and Historical Background

Present Status

In the Soviet Union[12] about one dozen (according to the published literature) plasma engines of all three major types (discussed below) have been flight tested and used in satellite applications. MPD engine-like plasma accelerators have also been used to inject alkali vapor plasma into the upper atmosphere along earth's magnetic field lines for geophysical research purposes. It has been stated that these engines performed in space "substantially as expected from vacuum tank tests", so that some of the uncertainties of ground testing seem to have been removed. Thruster life times up to 1000 hours and potentially useful specific impulse values and efficiencies have been reported using a multitude of noble gas and alkali metal propellants. These engines[12,13] ranged in thrust from 4.3 to 24 mN and in power level from 100 to 3000 W[12,13] which is substantially smaller than the sizes of several newton and several hundred kilowatt considered here. However, there have also been in the USSR[14] experimental and theoretical electrode studies up into the tens of kiloamperes, so that the basis for scaling up the various thruster types should exist there.

In Japan[15], at least one plasma propulsion system with a quasisteady pulsed self-field thruster was tested successfully in space. A Shuttle-based test of a similar plasma thruster has been made on the Shuttle flight STS-9 on Spacelab as experiment SEPAC[16].

In the United States[10], pulsed solid propellant (ablation) plasma thrusters were developed relatively early and used successfully on numerous satellites since about 1968 for station keeping and attitude control. Development of these thrusters to larger sizes continued. However, it is not clear to this writer whether power levels and efficiencies suitable for the applications considered here can be reached by this method, but if so the electrode cooling and erosion problems to be expected will be as severe as with the other types of thrusters using electrodes.

Also in the U.S.[17,18] at least two hydrogen thermal arcjet engines (1 kW and 30 kW sizes) were readied for flight testing, which however was not carried out for good reasons: first, there was no application in sight and (for the 30 kW at least) no power supply after the cancellation of the SNAP 8 development. Second, there was no reason to doubt that such engines, if they were ever needed, would perform in space as predicted from tank tests.

The life tests which were part of these arcjet engine developments (e.g. 500 hours for the 30 kW engine[18]), and other life tests of radiation/regeneratively cooled arcjet and MPD engines up to about 730 hours established[19] by extrapolation the potential of electrode life on the order of several thousand hours, albeit mostly under conditions of relatively high plasma densities. However, beyond those thermal arcjet engines, there resulted out of the rather substantial R & D effort on plasma propulsion in the U.S. and in Western Europe no flight qualifiable thrusters, nor even the basis for a clear choice of the thruster type and propellant(s' to be developed in the future. However, four favored thruster types and perhaps a half dozen favored propellants have tentatively emerged.

Up to about 1974, many different laboratory thrusters had been tested with nearly 20 different single or mixed propellants. The large body of data on this earlier work was summarized and in some cases critically evaluated in Refs. 5 to 9. The evaluations as to the credibility of the results differ somewhat. Some very high performance data were reported prior to 1967 by a number of laboratories particularly with AAF thrusters using hydrogen or lithium, claiming efficiencies in the 50 to 80% regime and correspondingly high specific impulse values. These data are all now considered erroneous, mostly due to tank gas entrainment (discussed in a subsequent section) and in a few cases due to the use of a thrust plate. The more reliable data (which were repeatable at very low tank pressures) ranged in efficiencies up to about 40% mostly with alkali metals, ammonia and hydrogen; much lower efficiencies were experienced with argon, nitrogen and helium. Later efficiencies in the mid-thirties were achieved also with argon. Thus at least prior to the very high power (multi MW) pulsed thruster experiments of the past decade, the reliable performance data (with efficiencies in the 30-40% range) with acceptable propellants were just barely sufficient for future applications of the type considered here, assuming adequate weight, life and reliability.

The more recent and steadily improving performance data from quasi-steady pulsed self-field thrusters from Princeton University and other laboratories must now be added to the data bank. However, except for the Tokyo University flight test units these pulsed thrusters, like most of the older laboratory thrusters just referred to, are still far from flight qualifiable hardware in terms of cooling, propellant supply, erosion, etc. Also there appear to remain, in the opinion of this writer, serious uncertainties concerning the actually attainable performance in space with a precisely timed propellant valve, and other unresolved system life and weight problems. The existing Soviet and Japanese MPD flight systems have only modest performance to our knowledge, and are of much smaller thrust and power than the systems contemplated here. Thus MPD plasma propulsion has a long way to go to a competitive demonstration system for the thrust range of several newtons, but still appears potentially promising with the proper investment.

Plasma Propulsion Development History - Brief Evaluation

Plasma propulsion research and development[20] started in a major way in the second half of the 1950s, a period of "plasma euphoria" when magnetically confined fusion appeared to be "just around the corner." Although this propulsion work could build on fundamental theory coming from astrophysics, fusion research, early MHD generator studies and arc physics and on a vast body of empirical information from many arc devices (switches, lamps, rectifiers, furnaces, welding arcs etc.), the progress of plasma propulsion since the early 1960's has been somewhat disappointing especially in the U.S. and in Western Europe, as indicated above, in spite of a fairly substantial investment. The reasons for this (as seen by the author) are examined briefly as an aid for the planning of future work.

Plasma propulsion development, in contrast to that of electrostatic ion systems, was held back by the following conditions:

1. The theoretically exceedingly difficult arc discharge and gas acceleration process which is very difficult to make efficient.

2. The large variety of possible plasma accelerator types and geometries-- with electrodes (i.e. applied) or with induced current, with continuous (dc or rf) or with pulsed operation, with or without externally applied magnetic field; with linear, rectangular or axisymmetric geometry coupled with an almost limitless choice of possible propellants-- solid, liquid, gaseous, elements, compounds and mixtures.[5-11,21] This wide range of choices led inescapably to a diffusion of the limited efforts and funds.

3. The age old, still unsolved arc electrode problems (for most thruster types)-- spots, erosion and losses.

4. The initial lack of full appreciation of the ground testing difficulties (tank gas and wall interference), hence no plans of early space tests.

5. The suitability of most plasma thrusters primarily for relatively high power levels and the cancellation of the nuclear power supply development programs which moved potential applications even further into the future.

6. The rather limited funding support compared with that of the ion engines, which ceased almost completely in the early 1970s in the U.S. and in Western Europe before a flight-test prototype thruster and system was reached.

Only now, with the Shuttle in operation, plans of new earth orbital operations and the necessary larger electric power supplies have become financially realistic. These plans have given new incentives for the development of multinewton sized electric propulsion systems and, what is most important, have furnished more definite propulsion performance requirements and missions which can serve as a basis for propulsion system comparisons.

With these new incentives, the insights gained from past work, the large facilities now available for testing and new tools for theoretical analysis, it should be possible to overcome the cited difficulties and evolve one or two cost-effective plasma propulsion systems. The success will be contingent on:

a) careful coordination of the research and development efforts and early reduction of the many choices mentioned above,

b) proof that the potential advantages of plasma propulsion still exist in spite of the latest advances of the ion systems and

c) adequate funding to develop prototype systems (including space tests) for comparative evaluations.

MPD Thruster Types, Performance Trends and Limits

MPD Thruster Types Considered for the Station Keeping, Drag Make-Up, Orbit Raising and Maneuvering of Large Space Structures

The following four MPD thruster types, all rotationally symmetric, have emerged out of the multitude of possibilities as the most promising ones for larger power levels in addition to the existing pulsed teflon ablation thrusters, the development potential of which this writer cannot presently evaluate:

1. The self-field thrusters (here SF-MPD for brevity), operated in a long pulse (ms) "quasi-steady" mode, or continuously (Fig. 1).

2. The axial applied-field (AAF-MPD) thrusters, referred to in the literature variously as MPD-A, Hall arcjets, Hall thrusters or magnetic annular arc (MAARC). These thrusters have a geometry similar to group (1) above except for the external magnetic field coil or permanent magnet. They have generally been operated continuously but could be pulsed (Fig. 2).

3. The low density (original) Hall accelerators or Hall-ion (HI) or radial applied field thrusters, usually operated continuously (Fig. 3).

4. The flat coil induction thrusters requiring sub-μs pulses (Fig. 4, Refs. 22 and 23).

The discussion here will be brief and limited primarily to the first two types, which are under prime consideration in the author's laboratory. This discussion is intended to bring out certain trends, comparisons, current technical problems and apparent limits as seen by the author today. It

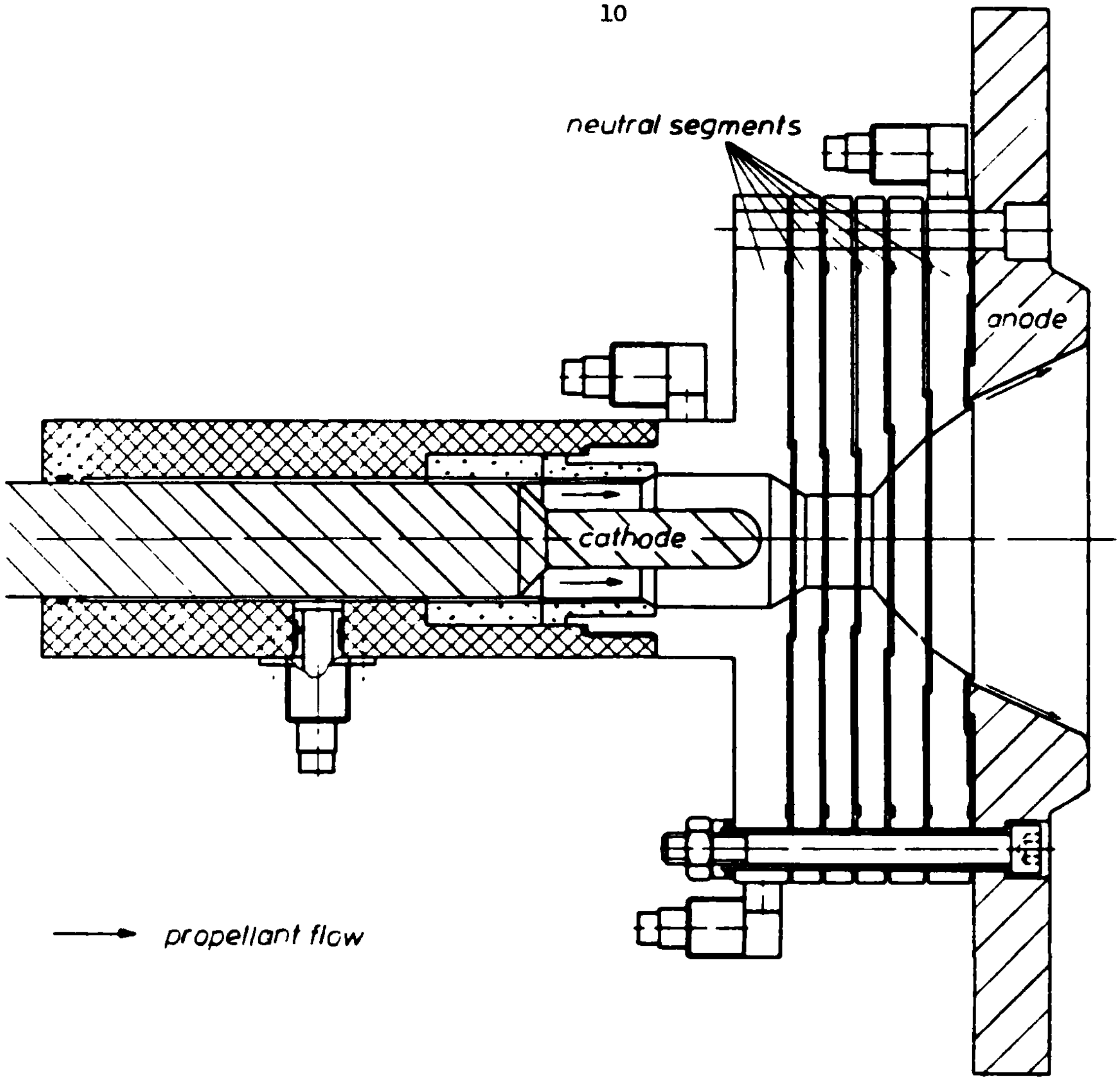


Fig. 1a: Stuttgart University Continuous Water-cooled Research Self-Field MPD Thruster (Capacity about 6 kA or 18 N Thrust with Argon).

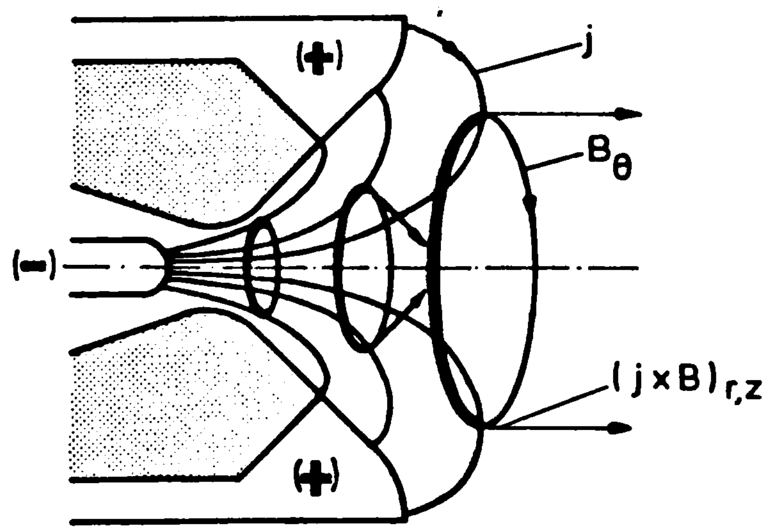


Fig. 1b: Operating Principle of Self-Field MPD Thruster.

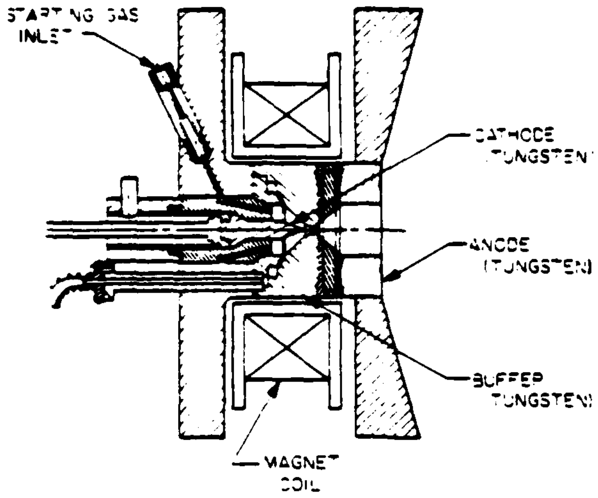


Fig. 2a: Applied-Field MPD Thruster, Lithium-Fed Hall Current Accelerator from Ref. 5.

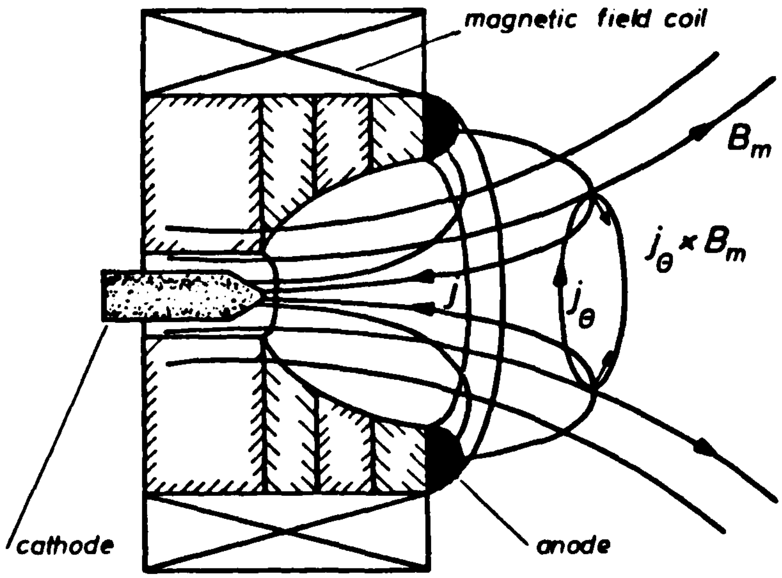


Fig. 2b: Operating Mechanisms of Applied-Field MPD Thruster.

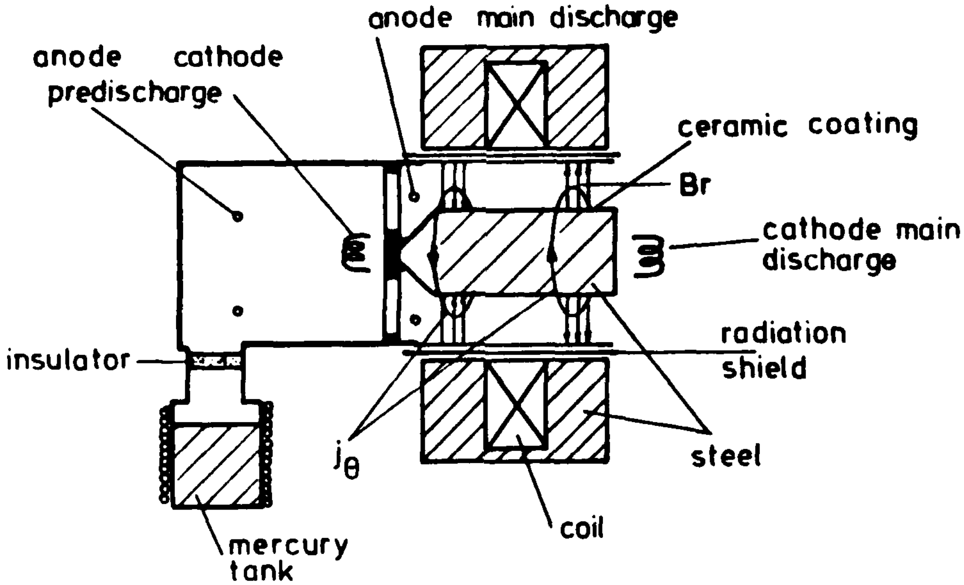


Fig. 3a: Low Density Hall or Hall-Ion MPD Thruster (schematic).

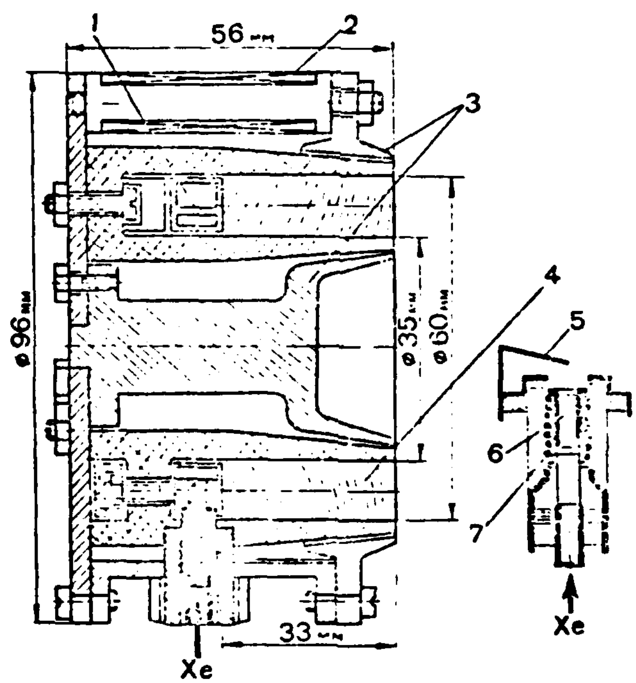


Fig. 3b: Closed Drift Hall-Ion Thruster Flown on the Russian Satellite Meteor I, 1971, from Ref. 13.

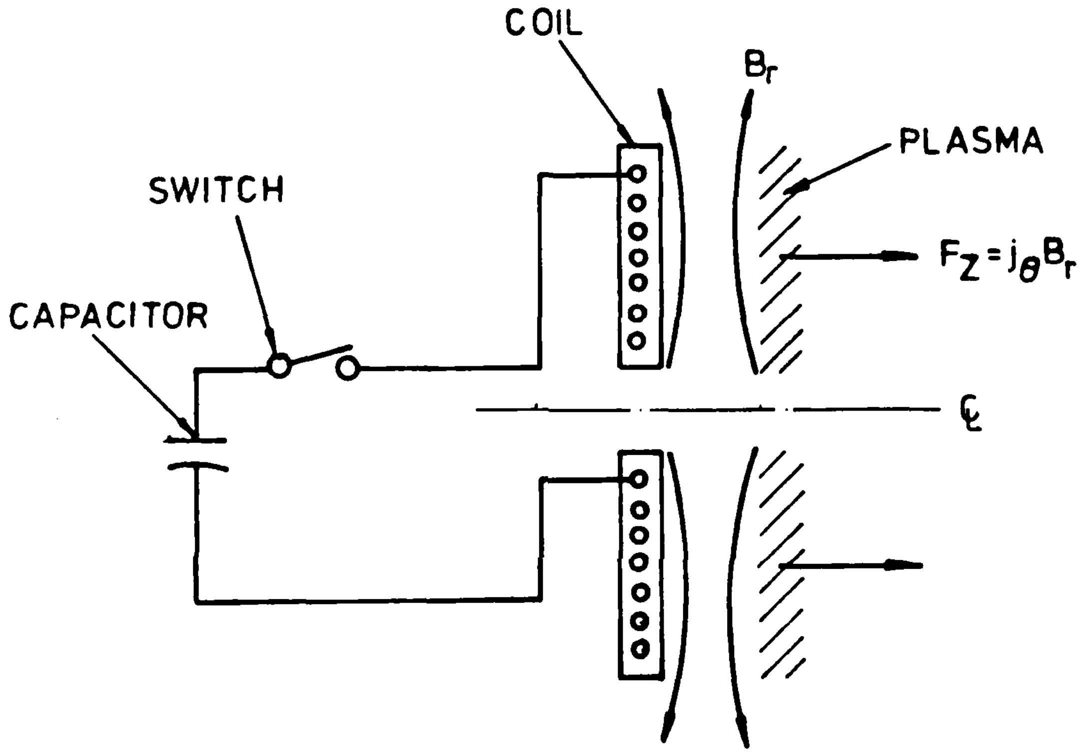


Fig. 4: Flat Coil Induction Thruster Schematic from Ref. 22.

is not intended as an updated replacement of the comprehensive summary papers of the past (Refs. 5 to 11), nor a discussion of the latest experimental or theoretical results on each thruster type, as given in papers devoted to each type. Rather it should be seen as an addition to those other publications.

The thruster of type 4, the refined successor of the early "pinch" and "punch" induction coils used as EM-shocktube drivers and in fusion research, is the only induction type thruster still under active study and development. As such, it avoids the severe electrode erosion and loss problems of the other three groups (discussed in a subsequent paragraph). The author cannot yet evaluate the potential of this thruster and system compared with the other three.

While thrusters (1) to (3) are basically continuous engines, they can all be operated in a "quasi-steady" pulsed mode, that is with pulses which are long compared with the time required to reach a rather steady discharge and flow condition (but <u>not</u> steady state electrode temperatures). Pulsed operation has advantages in thruster performance and cooling and in the ease of preliminary evaluation testing of different geometries, propellants etc., but has obvious system liabilities. The thruster and other system component information required to choose the optimal thruster type and operating mode has yet to be evolved, and the choice may well depend on the mission.

<u>General Performance Trends of MPD Thrusters</u>

The experimental performance curves in terms of efficiency vs. exhaust velocity (or specific impulse) of the thrusters tested so far have varied widely even within each thruster group and with each single propellant, depending strongly on the input power level, the thruster design, the mass flow and other variables (such as the applied field in the AAF-

MPD thruster groups), and, unfortunately, also on the test facility, notably the tank pressure and size.

The efficiency vs. exhaust velocity curves of different MPD thrusters can at least roughly be approximated by an equation which fits ion engines more precisely (Fig. 5 and Table 1):

$$\eta(c_e) = \eta_{max} \frac{c_e^2}{(k_1 U_c)^2 + c_e^2} \qquad (1)$$

where

$$U_c = \{2e \; \Sigma_{i,j}(\alpha_{ij} V_{Ij}(j) + \beta_i V_{Di} + V_{Li}) \; / \; \bar{m}a\}^{1/2}$$

is the Alfvén velocity based on the pertinent ionization $\alpha_{ij} V_{Ij}(j)$ and (if applicable) dissociation $\beta_i V_{\Gamma i}$ and the latent energy of vaporization V_{Li} added here, for i atomic or molecular species and j levels of ionization. k_1^2 is for ion engines always > 1, typically a number on the order of 10, since $(k_1 U_c)^2/2$, the energy invested in the ion production process, is typically 100 to 250 eV per useful ion. For the MPD thrusters investigated, k_1^2 varied fairly widely, and $k_1 < 1$ is possible particularly for AAF-MPD thrusters which frequently accelerate partially ionized plasma (cf. Ref. 5, Table 7). A few values of the Alfvén velocity for singly and doubly ionized propellants (but without the latent heats of vaporization) are given in subsequent paragraphs of this paper (cf. Table 2 and Fig. 8). η_{max}, which is normally near unity for ion engines (e.g. 0.85 by Refs. 24,25), is much smaller for most MPD thrusters. For plasma thrusters, η_{max} can be roughly associated with the thermal efficiency and the following fraction in Eq. 1 with frozen flow and other losses in the acceleration process. While the ion engines follow the curve given by Eq. 1 up to very high exit velocities, most MPD plasma thrusters reach (with increasing specific impulse) a maximum

Table 1: Models of Thruster Types (with approximation (1)) (Ref. 4)

Thruster	Prop.	η_{max}	$k_1 U_c$ m/s	Possible c_e area, km/s	Remarks
Ion	Ar	0.84	23,930	15 - 100	projected
Ion	Xe	0.93	14,240	10 - 100	projected
MPD, applied field	Ar	0.39	8,410	10 - 30	hardware
MPD, self field, quasi steady	Ar	0.36	12,710	5 - 35	hardware
MPD, self field, continuous	Ar	0.22	2,430	4.5 - 12.5	hardware

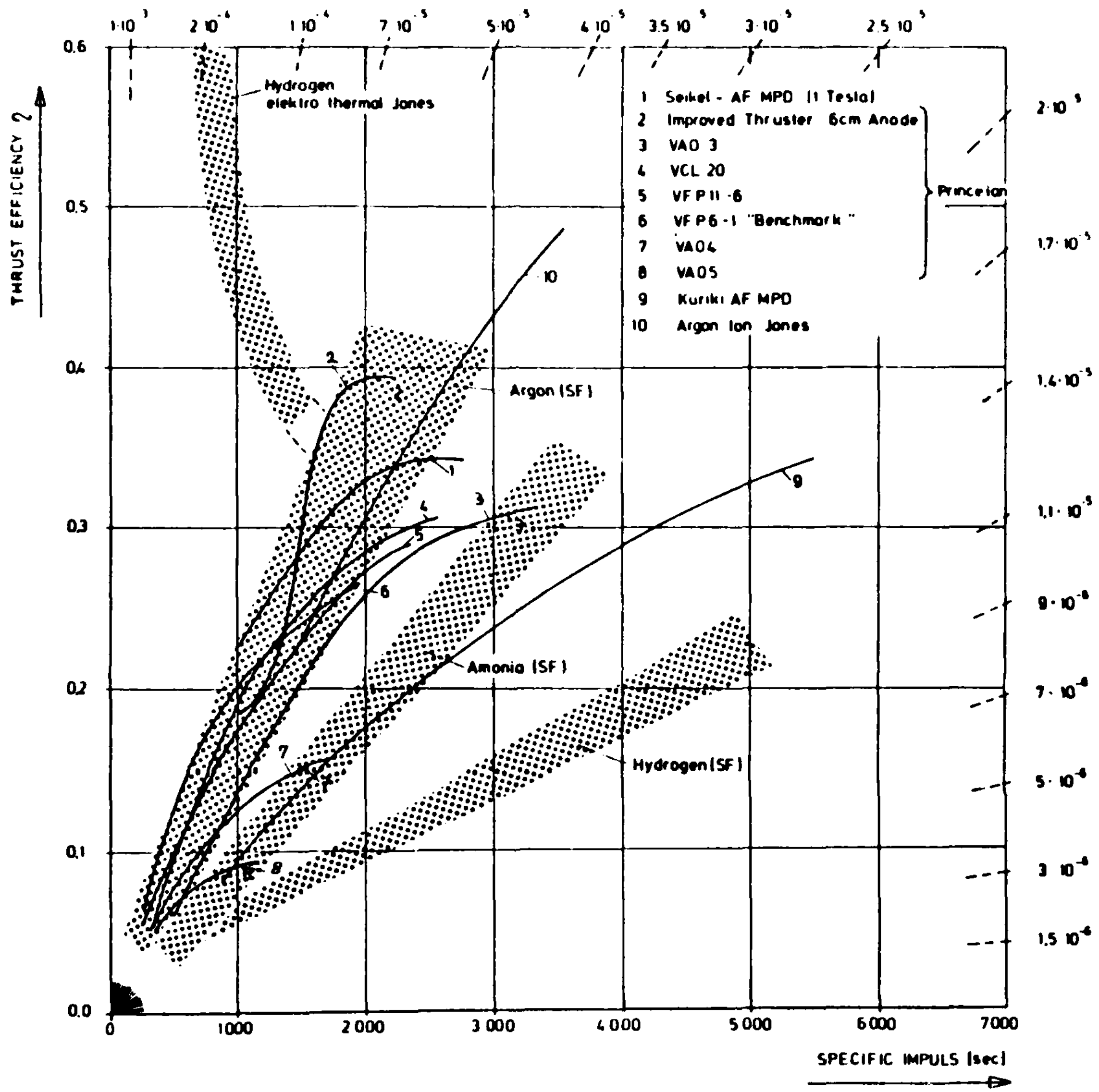


Fig. 5: Various Thruster and Frozen Flow Efficiencies for Different Propellants, Indicating Corresponding Thrust-to-Power Ratios.

efficiency point beyond which the efficiency drops again (departing from the matched curve given by Eq. 1).

Admittedly, the trend of the efficiency curves is quite different at very low exit velocities in the purely thermal regime as long as there is no dissociation or ionization and in some cases no latent heat of vaporization to be accounted for (the latter being supplied by waste heat). But this regime is not of much concern here except possibly for hydrogen.

It should be noted that the maximum thrust-to-input-power ratio is given by the steepest tangent to the efficiency curve from the origin. If $\eta(c_e)$ can be approximated by Eq. 1, this becomes:

$$\left(\frac{F}{W_e}\right)_{max} = \left(\frac{2\eta}{c_e}\right)_{max} = \frac{\eta_{max}}{k_1 U_c} \tag{2}$$

This and the general steepness of the efficiency curve depends primarily on the propellant, since U_c varies over a factor of more than 20, while the thruster dependent factors η_{max} and k_1 vary generally over factors of 3 or less. The newest ion thrusters have reduced the previous k_1^2 values by more than a factor of 2. The steepness of the efficiency (η vs. c_e) curves is of prime importance for the transport rates and transport costs of many typical near-earth missions as long as η_{max} is "sufficiently" high to be of interest (cf. Refs. 1 and 4). The heaviest propellants generally have the lowest values of U_c and thus produce the steepest efficiency curves.

For any one propellant, the thruster efficiency curves of many of the better MPD thrusters fall into a broad band, as indicated in Fig. 5 for three propellants-- argon, ammonia and hydrogen. From past transport cost studies for near-earth missions[1] it appears that the MPD propulsion system efficiency should be at least 20-25% or the thruster efficiency at least 25-30% for the system to be competitive, while specific impulse values above ≈1000 s would be sufficient for many missions. However, with most propellants and MPD thrusters much higher specific impulse values than the minimum must be achieved in order to obtain acceptable efficiencies, e.g. for 30% efficiency with argon, 1500 to 3000 s are necessary, depending on the thruster, and still higher values with the lighter propellants. This general trend of the curves in Figs. 5 and 6 and of Eq. 1 is responsible for the importance of and the striving for the highest specific impulse values with any one given propellant, as discussed in the next subsections.

Whenever the same efficiency can be reached with several propellants, the one which does this at the lowest specific impulse, i.e. the heaviest one, will be preferable because of the thrust/power ratio, if other things are equal.

It should be noted in connection with Eqs. 1 and 2 that the applicable value of U_C (and the losses) can jump to higher values when second level ionization occurs, i.e. when a thruster is successfully driven to higher specific impulse values. This frequently sudden change from first to second ionization level with concurrent change in operating voltage has been observed with some AAF-MPD thrusters, particularly with lithium.[5]

The Self-Field (SF-MPD) Thrusters

The SF-MPD Thrusters (Figs. 1 and 7) have been much more extensively investigated experimentally than the other two groups discussed here, both with continuous and with pulsed quasi-steady operation, with many different propellants. The principle of operation should be clear from Fig. 1 showing how the self-induced circumferential or "pinch" magnetic field (B_θ) interacting with the applied meridional currents produce meridional forces on the plasma normal to the current, mostly axially and radially inward. These forces, called blowing and pumping by Maecker[26] result respectively in direct acceleration or pressure and temperature rise with subsequent expansive acceleration (cf. Ref. 27, ch. 8.9). The pumping effect could also enhance recombination through additional collisions.

The pulsed teflon ablation thrusters which have been in use for many years are also essentially of the self-magnetic (SF) type, but with a rectangular geometry (though rotationally symmetric teflon thrusters have also been tested).

Of the three thruster types discussed here, the SF-MPD thrusters have the lowest arc voltages for a given power level, because the self-induced magnetic fields are relatively weak unless very high currents (tens of kiloamps) are applied, which explains the particular need for pulsed (quasi-steady) operation of these thrusters. Typical values are ≈50-70 V for a ≈200 kW continuous thruster compared with 200-300 V for a ≈6 MW pulsed thruster of roughly the same size. Since the electrode loss voltages change much less rapidly with the current or power level, the thermal efficiencies of these thrusters improve with increasing power. Because of the high currents required, the SF thrusters also have the most severe electrode erosion and cooling problems, though the latter can be reduced with pulsed operation, as will be shown.

The phenomenon of self-magnetic plasma acceleration was first investigated by Maecker[26], Wienecke[28] and others in Germany (ca. 1955) who studied velocity distributions in the cathode jets of carbon arc lamps. A simple integral formula by Maecker[26] gives the electromagnetic thrust fairly precisely for any thruster geometry and assumed or measured current distribution on the cathode and anode and thus also the contribution of the electromagnetic forces to the average axial exhaust velocity in the form

$$c_{EM} = \frac{\mu_0 \cdot J^2}{4\pi \dot{m}} f(\mathrm{geom}) \qquad [\mathrm{m/s}] \qquad (3)$$

$$\approx (1.85 \text{ to } 3.05) \cdot 10^{-7} \frac{J^2}{\dot{m}} \qquad [\mathrm{m/s}] \qquad (4)$$

where J is the total electric current, $\dot{m}$ the propellant mass flow rate, f(geom) is a function of the electrode geometry and the radial current distribution on the electrodes (cf. Ref. 27, ch. 8.9). The numerical values given in Eq. 4 are typical for models tested so far. It is important that the purely theoretical Maecker integral formula for the thrust and the resulting electromagnetic contribution to the average axial component of the exit velocity (Eq. 3)-- which are well verified experimentally-- are independent of any assumed model or rotational (circumferential) uniformity of the discharge. Any arc spokes and/or anode spot attachments have no influence on the result. These details of the arc form and distribution, however, affect the exit velocity distribution, the resulting thrust direction and the efficiency of electromagnetic thrust production. Of course, models are generally used to evaluate the factor f(geom) simply, and herein lie some uncertainties. No such simple and generally valid thrust formula independent of a model exists for any of the applied-field thruster types described herein. (It does for the linear Faraday channel accelerators).

Empirically, it had been known for some time that the current-to-gas-mass flow ratio ($J^2/\dot{m}$) has a limit for each thruster and propellant where stable operation of the arc becomes impracticable. Beyond that so-called "onset" value, pronounced arc voltage fluctuations, severe anode spots and anode erosion, increased anode losses and thruster efficiency drop are observed. Attempts to exceed the onset limit $(J^2/\dot{m})_{Crit}$ are foiled by erosion of the anode (in some cases[29,30]), exposed insulators, and perhaps cathodes. This erosion apparently supplies the additional mass flow needed to limit $J^2/\dot{m}$, as demonstrated e.g. by Suzuki et al.[30]. (See also under Cann Minimum Voltage Hypothesis).

The great importance of the onset or critical $J^2/\dot{m}$ values is clear from the previous discussion (Eq. 1) and from Fig. 1 since practically useful thruster efficiencies (say above ≈30%) can (if at all with a given thruster) be achieved only with each propellant at the highest specific impulse (or $J^2/\dot{m}$) values presently attainable with that propellant, as will be seen.

For SF thrusters, Hügel[7] has correlated the onset $(J^2/\dot{m})$ ratios from different sources and thrusters with different propellants against their atomic weights (Fig. 6) which gave roughly

$$\left(\frac{J^2 M_a^{1/2}}{\dot{m}}\right)_{Crit} \approx 15 \text{ to } 33 \times 10^{10} \qquad \left[\frac{As}{kg}\right] \qquad (5)$$

or with Eq. 4

$$(c_{EM})_{Crit} \approx (3 \text{ to } 10) \times 10^4\ M_a^{-1/2} \qquad [m/s] \qquad (6)$$

where M_a is the dimensionless atomic weight of the propellant, and the empirical constants in Eqs. 5 and 6 depend on the thruster geometry and the power level. The approximate magnetic velocity scale in Fig. 6 has been added by this author. These results agree of course with those of Malliaris[31] and Lien[32] whose data are included. Recent pulsed SF thruster data of Princeton University and the University of Tokyo report electromagnetic velocities by a factor of 2x larger than the upper value of Eq. 6 for argon, or ≈33 km/s.

Hügel[7] calculated the current and gas density distributions for some of these thrusters using several simplified flow models and empirical electron temperatures. He showed that near the onset, the ion density at the anode approaches zero due to the pinch (or radial inward) component of the electromagnetic force distribution. He concluded that at the onset points the ion densities at the anode become too low to

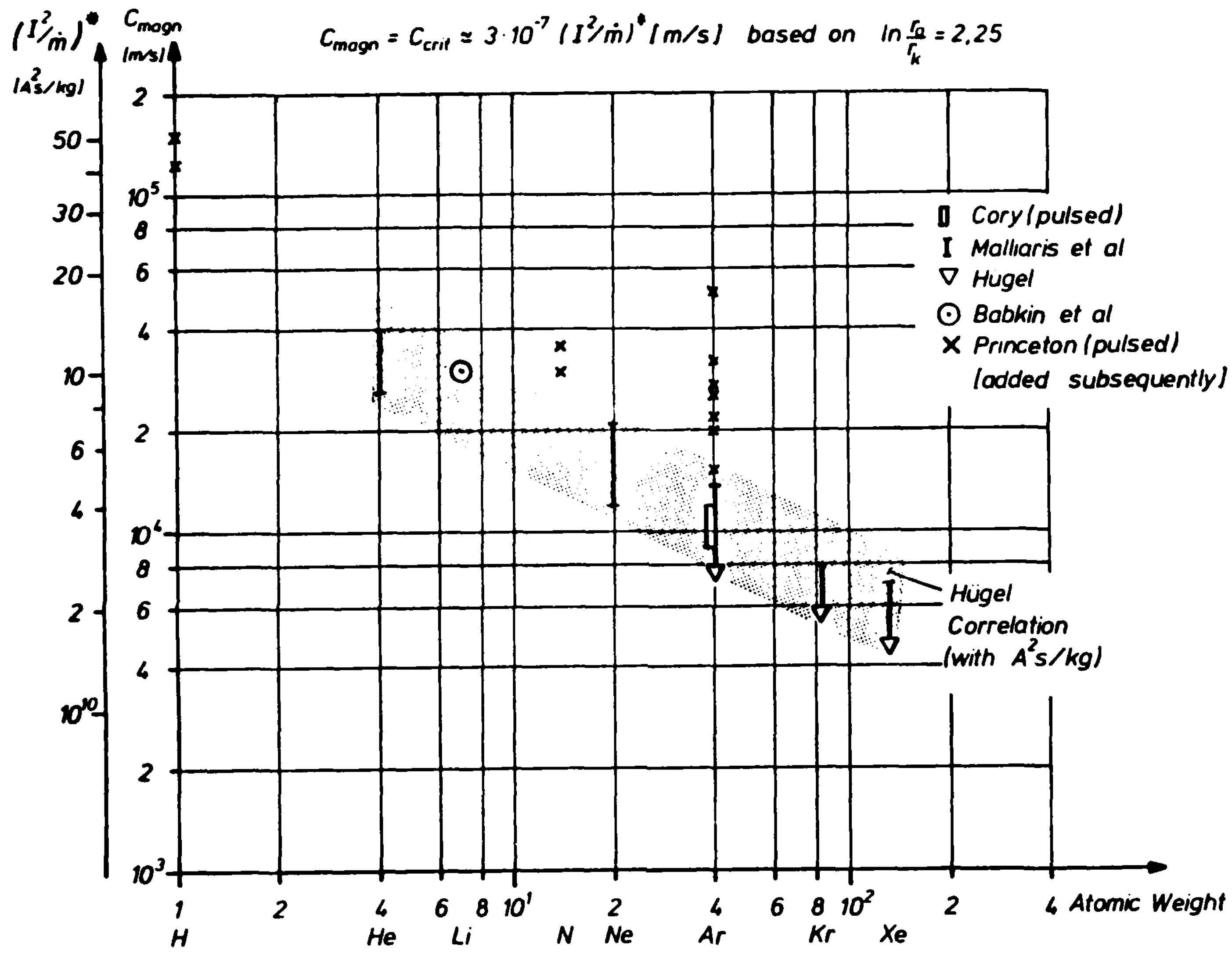


Fig. 6: Self-Field MPD Thruster Onset or Critical Values of $I^2/\dot{m}$ and Corresponding Electromagnetic Velocities vs. Propellant Atomic Weight; Hügel' Correlation, Extended.

neutralize the electron current there, assuming singly-ionized plasma. This explanation is immediately plausible from the fact that the ratio of the magnetic pinch pressure rise (Δp_m) to the mean gas pressure in the jet p_j is proportional to the current/mass flow ratio parameter:

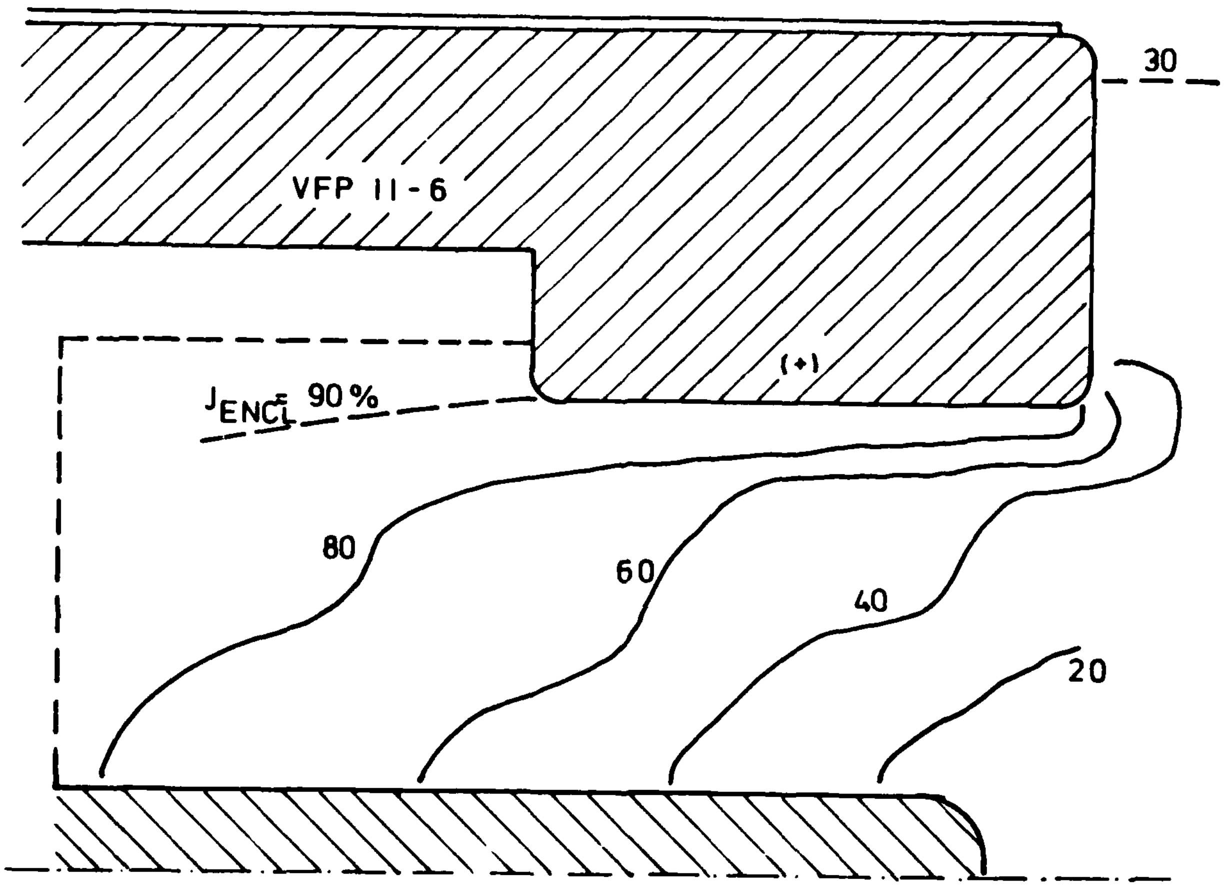
$$\frac{\Delta p_m}{\bar{p}_j} \sim \frac{J^2 M_a^{1/2}}{\dot{m}} \tag{7}$$

as may be verified by simple approximations.

However there appear to be also other theoretical approaches for arriving at critical values of $J^2/\dot{m}$. Cann arrives at this from his minimum arc voltage principle described in a subsequent paragraph. Schrade can show from his arc stability criterion[33] that instability must occur above certain $J^2/\dot{m}$ values.

King[34] has approximated the MPD accelerator flow with a one-dimensional channel theory and from this has estimated values of the Hall parameter in MPD accelerators. He ascribes the onset limit of $J^2/\dot{m}$ and the observed current distributions (Fig. 7) to excessively large Hall parameter values (approaching ≈10) as previously suggested by Rudolph et al.[35]. Incidentally the Hügel model would also indicate increasing Hall parameter near the anode surface as a result of the decreased plasma density there. The induced or "back-emf" ($\delta E_R = -c_e B_\theta$) which was neglected in the Hügel model would add to this effect, as pointed out by Lawless[36] and others.

All these different simplified analysis models, which are not contradictory to each other, predict some of the experimentally observed trends, but they cannot give absolute limits of the performance of thrusters. The discharge behavior near the onset, e.g. the arc attachment on the face of the anode (Fig. 7) is not analyzable with either the Hügel or



CURRENT CONTOURS

Fig. 7: Improved Pulsed (Quasi-Steady) Self-Field MPD Thruster Geometry with Measured Current Density Lines, Princeton (Ref. 34).

the King model. But these models suggest ways in which the limiting values of $J^2\sqrt{M_a} / \dot{m}$, hence the achievable specific impulses with each propellant and generally also the efficiencies could be improved. They are:

a) injecting neutral gas near the anode, to increase the gas and ion density there

b) extending the cathode to achieve (if possible) radial current flow to eliminate the radial ("pinch") pressure gradient

c) lengthening the thruster anodes to reduce current densities and ohmic heating and thereby improve the thruster efficiencies

d) reducing the anode exit diameter, to increase the mean plasma density for a given mass flow rate, counteracting the "anode starvation" and the rising Hall parameter.

Cold (neutral) gas injection near the anode surface has been used in self-field thrusters by the Princeton University Group[37] and by the University of Tokyo Group[38], in each case apparently with success in improving the specific impulse or onset velocity limit. There is a clear limit to the fraction of the propellant which can be injected near the anode, as found in a systematic investigation at Princeton University[37], before arc attachment problems at the cathode (and possibly also midstream arc stability problems) develop. Propellant introduction at the anode has also earlier brought improvements in the performance of applied field thrusters in investigations of Moore et al.[44] (see Ref. 5, p. 32) and of Kruelle[39] and others, but it is not yet clear whether this is for precisely the same reason. For Hall-ion thrusters and for applied field thrusters, the anode generally appears to be the logical place for most of the ions to be generated.

Extended cathodes and long cylindrical anodes have been used by the Princeton University group, both to supply sufficient cathode surface and to counteract the above mentioned anode starvation due to pinch forces which would not occur with a purely radial current flow. The purely radial current distribution was not achieved, but instead most of the arc attached itself to the front face of the thruster anode as seen in Fig. 7. There, conditions are quite different from those treated by the simple flow models of Hügel or King.

King's calculations predicted improving efficiencies through lengthening of the anodes, but a definitive optimization of the length was not possible since wall friction and heat losses were not included.

Reducing the exit area to mass flow ratio and thereby increasing the exit gas pressure leads to lower values of the Hall parameter and should, by King's arguments, lead to increased critical $J^2/\dot{m}$ and specific impulse values (in spite of reduced anode-to-cathode radius ratio). This, however, leads again to higher anode current densities, unless it is compensated by additional electrode length (a point not mentioned by King).

Kuriki et al.[40] use a one-dimensional channel model, assuming alternately low and high magnetic Reynolds numbers, but neglecting tensor conductivity (Hall) effects. They conclude that a converging-diverging channel would be optimal for current distribution, exit velocity and efficiency and that the length should be limited, in contrast to King, who (by extrapolation of his results) predicts a diverging geometry as the optimum.

All this is being further explored at present, both experimentally in several places and theoretically by use of the two simple theoretical models: an axisymmetric model for the conical thrusters (with back cathode), and the one-dimensional channel model for the long cathode geometry thrusters, respectively.

Through such variations, found mostly empirically, (but partly indicated also by the model calculations), considerable performance improvements have been achieved with pulsed SF thrusters recently, particularly at Princeton University and the University of Tokyo. It has been shown that the onset limits and thereby also the achievable specific impulse and efficiency values depend very strongly on the thruster geometry and the distribution of the propellant injection, and further on the propellant used and on the absolute thruster size, the power level and operating regime relative to the thruster size.

For argon, values of $J^{*2}\sqrt{M_a}/\dot{m} \approx 1.69 \times 10^{12}$ $A^2 s/kg$ have been reported[35]-- about 5 times the upper limit of Eq. 5 and Fig. 6. The corresponding magnetic velocity of 33 km/s was about 2 times the upper limit indicated by Eq. 6. Corresponding efficiencies in the low thirties were reported for the pulsed SF thrusters with argon.

The dependence of the performance on the propellant has not been as thoroughly explored as that of the geometry. Very roughly, the maximum exhaust velocities still appear to vary like the Alfvén velocities, or like $M_a^{-0.5}$, and maximum efficiencies at the maximum velocities are expected to be relatively insensitive to the atomic weight of the propellant, though the data of Kuriki et al.[41] show steadily increasing efficiencies with decreasing mean atomic weights (see also under propellant selection).

The effects of the operating point for a given thruster and of the absolute size and power level under purely geometric scaling are only partially understood. Each thruster appears to have a mass flow range at which the best performance is reached. Mead[42] and Kaplan[43] compared two geometrically scaled thrusters (2:1 in size) over a wide range of mass flow values and currents. The results of the scaling experiments were not very conclusive, partly because of (unexplained) excessive erosion with the larger thruster and because the whole question of how scaling should be done and evaluated appears to be not yet well thought out. The essential parameters for flow discharge similarity (which may not be possible with geometrically scaled thrusters) must first be established, and then the effects of variations of these parameters obtained experimentally.

In summary, continuous water-cooled self-field thrusters (in pre-1975 investigations) have, under reliable test conditions (adequate vacuum), reached efficiencies with argon of up to ≈20% at I_{sp} values of about 1400 s, at power levels up to ≈300 kW. The low efficiencies are due to low arc voltages and low thermal efficiencies (40 to 70%). With other propellants, higher I_{sp} values and somewhat higher efficiencies were achieved. Radiation-cooled SF thrusters had higher thermal efficiencies but were generally of much lower power level. For the pulsed SF thrusters of the past ca. 15 years at power levels of 3 to 6 MW, efficiencies into the mid-thirties at I_{sp} levels of 2000 to 3000 s and correspondingly much higher voltages and thermal efficiencies have been reported for argon; with other propellants efficiencies are comparable at somewhat higher I_{sp} values. Thrust-to-power levels run from about 35 mN/kW for argon down to about 10 mN/kW for hydrogen (Fig. 5).

The Minimum Voltage Principle and its Consequences

Cann[44,6] and shortly thereafter Bennett[45] derived on a purely thermodynamic basis (a minimum entropy generation hypothesis backed by empirical data) a simple theorem which, within the limits of certain restrictive assumptions, would predict certain performance trends and limits for the electromagnetic acceleration of gases in any device using an arc discharge (i.e. any MPD thruster type).

It must be pointed out (for the reader who wishes to save the time) that the theorem and its conclusions are highly controversial, and are rejected by many expert investigators on various grounds to be discussed. In the opinion of this author, the theorem (even if it should not apply precisely) has already predicted many observed performance trends so remarkably well and given enough insight into the behavior of MPD arc devices and stimulated enough thought that it deserves discussion. While the theorem is basically purely thermodynamic and not tied to any discharge form (like axisymmetric), the limiting assumptions make it in a sense still a theoretical model thermodynamically which may apply more or less well in different cases. Also, Cann and Bennett used the thrust formulas from various models to evaluate some of the results. The theorem is introduced at this point for comparison with the earlier self-field thruster data, though Cann had particularly the lower density discharges of the AAF-MPD thrusters in mind.

Ordinary electric arcs (without externally imposed magnetic fields) are known to always assume that diameter and temperature distribution which, for the particular current flowing, results in the lowest possible *voltage or lowest resistance*. This long accepted empirical fact, the "Steenbeck Minimum Voltage Principle" of the German literature, has later by Peters[46] been shown (not surprisingly) to correspond to the case of minimum entropy production (for a given current and set of other conditions).

Cann and then Bennett postulated that this should apply also to arcs in MPD thrusters with strong external or self generated magnetic fields and gas crossflow; this is (1.) the first basic assumption. Like any other minimum entropy or minimum energy principle in physics, this cannot be proven purely theoretically. The following three additional assumptions were made:

2. Only ionized gas particles are initially being accelerated electromagnetically (i.e. only they play an important part in the thrusters in question).

3. Any recovery of ion or electron thermal energies (including those from possible recombination) can be neglected.

4. During any virtual changes of the arc condition at constant current, the thrust would remain constant.

The applicability and implications of these assumptions will be discussed further on. With these assumptions, one can partially differentiate the energy (or power) balance equation for the thruster, using the ion mass flow ($\dot{m}_i$) as the variable at constant current and thrust, and reach the following conclusions:

1. A definite minimum mass flow ($\dot{m}_{cr}$) must be ionized (and dissociated where applicable) by the arc to achieve the minimum voltage.

2. Half of the power converted by the arc (less electrode and other direct heat losses) must go into internal energy (mostly ionization and dissociation energies) which were assumed to be unrecoverable.

3. Consequently, the thrust efficiency cannot exceed half of the thermal efficiency, thus it must be below 50%.

4. The electromagnetic part of the exit velocity cannot exceed a certain value-- the Alfvén velocity (U_c) for the degree of ionization which is applicable.

5. The "minimum" arc voltage also becomes a simple function of the applicable Alfvén velocity U_c.

Analytically stated,

$$\dot{m}_{cr} = \frac{T}{U_c} \tag{8}$$

or

$$(c_e)_{max} = \frac{T}{\dot{m}_{cr}} = U_c \qquad (9)$$

and

$$(V)_{min} = TU_c/J + V_A + V_c \qquad (10)$$

where the so-called Alfvén velocity U_c is defined as[47]

$$U_c = \left[\frac{2 \sum_{ij} e(\alpha V_{Ij} \quad \beta V_{Di})}{\bar{m}_a}\right]^{1/2} \qquad (11)$$

for i species and $\bar{m}_a$ the effective mean mass of the atoms in the jet. V_{Ii} and V_{Di} are the applicable ionization and dissociation potentials; α_i and β_i, the ionization and dissociation fractions. U_c thus depends on the level of ionization in the beam. V_A and V_c are the loss potentials of the anode and cathode attachments, respectively (discussed in a subsequent section). In some of his publications of the theorem, Cann[6,48] has introduced additional terms (e.g. for the enthalpies of the electron gas and the neutrals) into the energy equation, which slightly modifies the formulas given above.

The Alfvén velocities of some typical propellants, i.e. the noble gases, the alkali metals and some molecular gases, both singly and doubly ionized, are plotted in Fig. 8 and listed in Table 2 against the mean dimensionless atomic weights (when fully dissociated) of the heavy particles in the beam, M_a. It is striking how closely each group falls on a straight line with slope differing not far from one half. Thus, for engineering purposes, the Alfvén velocities of each group may be approximated by

$$U_{cj} \approx k_j (M_a)_j^{-1/2 \pm \varepsilon_j} \quad , \text{ with } \varepsilon_j \ll 1 \qquad (12)$$

Table 2: Alfvén or Critical Velocities c_{crit} for Various Propellants

Propellant	Molecular Mass kg/kmol	c_{crit}^{+} km/s	c_{crit}^{++} km/s
H_2	2.016	55.1	—
He	4.003	34.4	61.7
NH_3	17.03	27.5	33.0
N_2H_4	32.05	24.9	31.2
Li	6.94	12.2	47.5
N_2	28.01	16.4	26.0
Na	22.99	6.6	20.1
K	39.10	4.6	13.2
Ar	39.95	8.7	14.5
Cs	132.90	2.4	6.5
Ne	20.18	14.4	24.4
Xe	131.30	4.2	7.0
Kr	83.80	5.7	9.4

$^{+}$ every atom singly ionized
$^{++}$ every atom doubly ionized, if possible

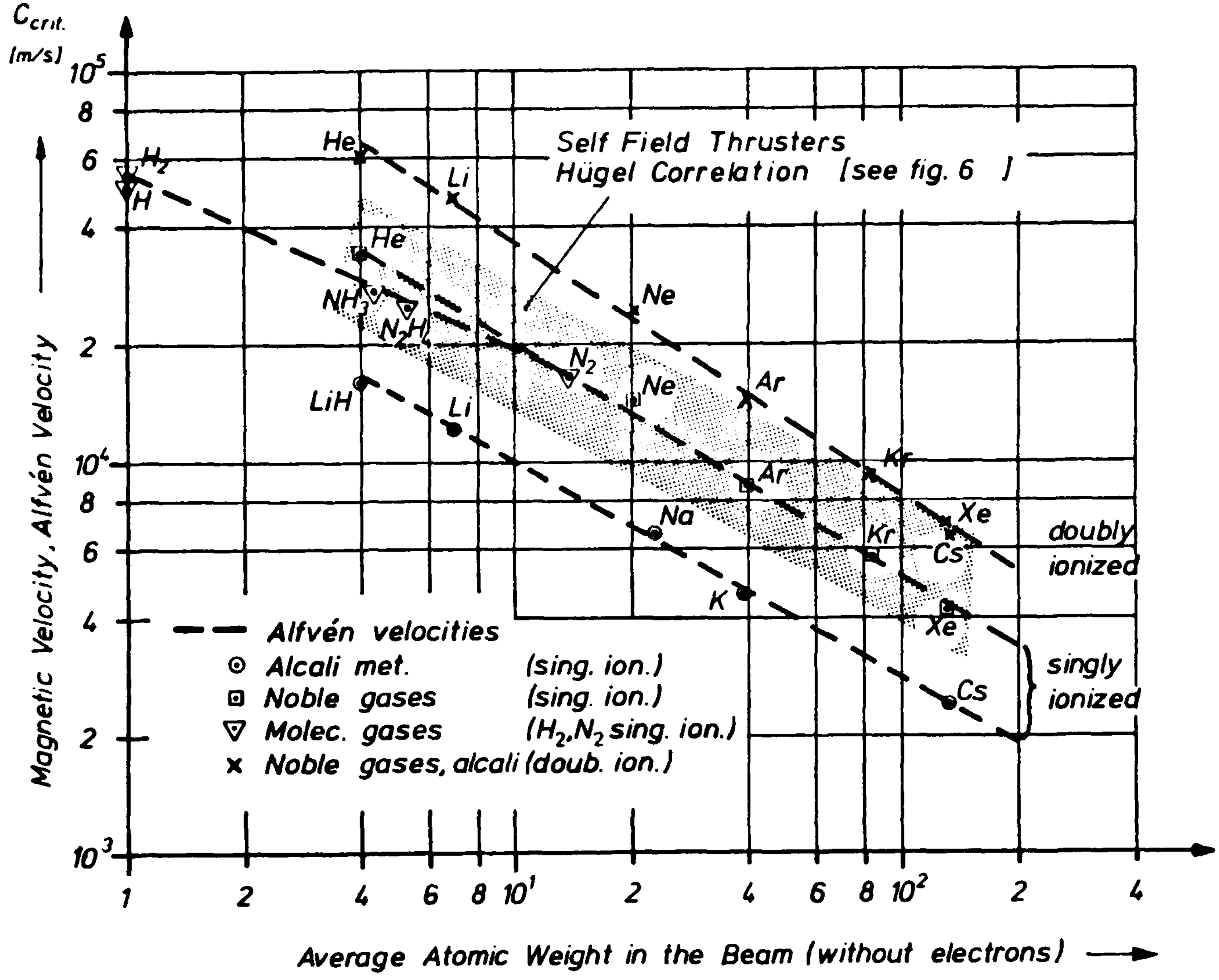


Fig. 8: Alfvén Velocities for Various Noble Gas, Alkali and Molecular Propellants vs. Atomic Weight (Mean, Dissociated) Compared with Hügel Correlation for Self-Field Thrusters.

with k_j and ε_j constants for the group (j) and level of ionization. For the doubly ionized particles, the alkali and noble gases fall on the same line.

The dotted band in Fig. 8 is the Hügel correlation of the self-field thrusters of Fig. 6 (expressed as magnetic velocities) which included He, Li, Ne, A, Kr and Xe. The Hügel correlation band falls right over the line for singly ionized noble gases. The agreement is remarkable in view of the uncertainties in both the experiments and their evaluation. The Princeton University pulsed thruster data for argon fall considerably above the doubly ionized noble gas line, which is no contradiction in view of the extremely high enthalpies reached in these tests.

Cann[6,48] has examined the voltages, maximum exit velocities and corresponding mass flow data of many SF and AAF-MPD thruster tests in the literature and obtained remarkable agreement with predictions based on his theorem and appropriate equations for the electromagnetic thrust. He has also identified many test data which go beyond his predicted limits, and listed these including some of his own as "questionable data" on the basis of tank pressure and tank gas recirculation, incorrect thrust measurements, erosion or other sources of error. Cann interprets the theorem not as an absolute limit but as the empirically most probable condition of the arc, which will be difficult to exceed. He postulates that, in cases where less than the critical mass flow is supplied into the thruster, the arc will usually "find" the additional mass flow required for minimum voltage from the tank, from the electrodes (e.g. anode spots) or from insulators at the onset condition, or if this is prevented the voltage will increase to produce higher levels of ionization and thereby a higher value of U_c. All this agrees with many experimental observations, as already pointed out, including those of Rudolph et al.[49], Hügel[7] and Suziki et al.[30]. In the last mentioned experiments the measured jet speed also equalled the Alfvén velocity of the eroded insulator (hood) material.

Kuriki and Suzuki[50] quote observations and results similar to those just discussed. However, they arrive at two critical or transition velocities, U_C and u_{tr} given by

$$u_{tr} = [U_C^2 + 3k(T_i + T_e)/m_i]^{0.5} \tag{13}$$

where T_i and T_e are the ion and electron temperatures, respectively. They thus include the thermal internal energies of the ions and the electrons (surprisingly not the enthalpies) in their transition velocity, but it is not clear whether this is based on a minimum voltage hypothesis. They state that these two transition velocities (or $J^2/\dot{m}$ values) bound (with increasing currents) three regimes they call "heating", "ionizing" and "EM acceleration", respectively. At the lower transition velocity (or current) the thrust vs. current curve would steepen to the J^2 slope, and at the higher one (u_{tr} above) the voltage vs. current curves steepen to the J^3 slope predicted for the EM acceleration regime. Though they observed at onset similar voltage instabilities and change in anode attachment described by all other investigators, they do not, like most others, consider either of the transition points as limits for the operation or performance of pulsed SF thrusters. Rather they advocate as the goal advancing higher into the "EM acceleration" regime without producing excessive ablation. Their experiments did produce appreciable ablation of the plastic anode hood, which makes interpretation of the data difficult. Their measured exit velocities went up to about u_{tr} (Eq. 13), calculated assuming doubly ionized argon.

De Villers and Burton[51] state they found "no direct causal link" between U_C and the exit velocity of MHD channels, but that U_C can be used to determine the field ratio E/B. To this writer this sounds contradictory, since E/B is the limiting velocity of such channels. Baksht et al.[52] agree with U_C as the critical velocity and, like Hügel, ascribe this limit to "anode starvation". Yoshikawa et al.[53] recently obtained critical $J^2/\dot{m}$ values with four propellants which, for at least two of the propellants, agree quite well with the "theoretical" values derived from the Alfvén velocities.

The objections raised by various investigators against even considering the minimum voltage theorem are mainly:

a) that the main assumptions on which it is based (1 to 4 above) are not approached by any one MPD thruster type and that it is therefore an over-simplification. In particular, that the conversion of thermal energy modes in the magnetic nozzle is not properly taken into account.

b) that reliable experimental evidence exists which contradicts the conclusions.

c) that one can prove from first principles (by solutions of the conservation equations) that the conclusions are false, e.g. that even with magnetic acceleration alone, higher than 50% efficiency is in principle obtainable; that therefore invoking any such theorem is superfluous and misleading.

The writer agrees substantially with objection (a), in particular that conversion of the thermal enthalpies of the ions and electrons should be more properly taken into account; this can be quite substantial-- ≈4 eV per ion or more for 16,000 K (counting also the enthalpy of the electron gas). Actually, the enthalpies of typical pulsed thruster experiments indicate much higher temperatures, typically between 25,000 and 40,000 K and higher. In some cases, even possible ionic recombination cannot be fully excluded. Thus, the conclusions of the theorem in its present form are only approximations, even if the basic premise is correct. Unfortunately, when conversion of these thermal energies is included, the conclusions become less definite, and the matter is complicated by the fact that the conversion of thermal enthalpy and the efficiency of electromagnetic thrust generation are interrelated.

Objection (b) is based on relatively high specific impulse values observed with pulsed SF thrusters with argon and on the fact that little or no second ions were supposedly observed in the jet. The writer feels that in view of the very high enthalpies there should have been second ionization near the cathode, possibly followed by partial recombination (which reverts back to case (a)). In fact, there is some uncertainty in all these pulsed thruster data, just as there is in the theorem, as stated. The writer does not agree with objection (c) that conclusive proof of a contradiction from purely theoretical solutions has been brought forth as yet, but feels that every effort must be made to advance with the theoretical solutions, at least with improved models taking into account real gas, magnetic Reynolds number and Hall effects.

The writer believes that the basic premise of a minimum voltage arc discharge very probably applies to MPD thrusters also, and that the agreement of absolute results or of trends between experiments and the theorem is more than pure coincidence. The self-field thruster data cited above and the many correlations of Cann and others also for applied field thrusters show that there is much experimental and some theoretical evidence connecting the arc behavior and E/B for different propellants to the ionization potentials in the same manner as is derivable from the minimum voltage hypothesis. Precise comparisons are in most cases not possible, since the levels and degrees of ionization (required to calculate U_C) are generally not known, because absolute spectroscopic determinations are exceedingly difficult in a thruster discharge. Also, the experimental values of $(J^2/\dot{m})_{cr}$ are uncertain because of ablation and other sources of error. Furthermore, the conclusions from the minimum voltage hypothesis are "broad brush" over all statements which could not possibly account for details like differences in geometry, local mass starvation regions in the discharge, etc. Thus, agreement or disagreement between experiments and the predictions derived from the minimum voltage hypothesis in any one case can be fortuitous and would not, in any case, prove (or disprove) all of the conclusions of the theorem stated above. It must also be admitted that the ionization energies (or the Alfvén velocities) as useful correlation parameters are obtained from the energy equation or the frozen flow efficiency (Eq. 1) alone, without the need for the minimum voltage hypothesis.

However, because of the insights which have been and can be gained, the further consideration of the minimum voltage idea appears fruitful, independently of the work toward more accurate approximate solutions of the flow discharge boundary value problem for specific thruster cases. The conversion of ion and electron thermal enthalpies (roughly as in Ref. 50) and the possible acceleration of neutrals by collisions should, if possible, be included in the minimum voltage theorem. Additional careful experiments with hydrogen (having only one level of ionization), as suggested by Cann, appear most important for comparison. Theoretically, the basic relations between the ionization potentials per unit of mass (i.e. U_C) and the attainable E/B values and EM velocities must be further clarified, as well as the conditions under which diffuse arcs become improbable and filaments and anode spots must appear.

The Axial Applied-Field (AAF-MPD) or Hall Arcjet Thrusters

The designation Axial Applied-Field thrusters was chosen because of the predominantly axial magnetic field direction in the region between the electrodes of the thrusters discussed in this paragraph, to distinguish them from the Hall-ion applied field thruster group, which uses a radial magnetic field in the discharge channel.

The AAF thrusters have in most cases very similar geometry to the axisymmetric self-field thrusters just discussed, but with the addition of either a field coil or a permanent magnet as a means of applying a coaxial fringing magnetic field. These thrusters have their origin partly in the purely thermal arc-jets (as developed also for heat shield materials testing) with the field coils applied to protect the anodes from a stationary arc attachment; and partly they evolved via the Hall-ion thrusters (which initially gave a rather poor performance) from the Hall current generator idea patented in the early 1930's. In most models tested, there was a central cathode in the rear and a ring anode further forward, with the cathode tip in a relatively small cavity (or built as a gas-fed hollow cathode) in order to have relatively high plasma densities there. A few models with the cathode forward of the anode have also been tested (Burkhart[54])-- a logical geometry in view of the electric field required to accelerate the ions. These latter designs represent a partial transition to the Hall-ion thrusters discussed in the next paragraph, but still with a predominantly axial B field in the space between the electrodes.

The thrusters discussed here are usually operated at propellant and power density levels substantially higher than the Hall-ion thrusters and therefore appear more promising for high power applications than the latter on the basis of thruster size and weight. Many investigators consider the AAF-MPD thrusters as still the most promising type for the long range, for all power levels from a few kilowatts upward. Prior to about 1975, most of the most capable laboratories concentrated much of their efforts on these thrusters, including elaborate and difficult diagnostic measurements of the jet velocities, plasma state, current and B field distributions.

The additional variables of magnetic field strength and (to some extent) geometry, increase considerably the quantity of data necessary to characterize these thrusters. There is thus a vast literature on the performance and diagnostic tests of these thrusters, with a large number of propellants which this brief review can not even attempt to adequately summarize or evaluate. But since the bulk of this work dates back to before 1975, it is adequately summarized in the comprehensive older summary papers of that time, e.g. Refs. 5, 6, 8 to 12, 39 and 55 to 57.

The axial-field (AAF-MPD) thrusters, compared with self-field thrusters, have the advantages of considerably higher arc voltages and hence lower currents for the same power level (see below), therefore potentially lower electrode loss fractions, erosion and cooling problems. They offer higher thermal and overall (thrust) efficiencies for a given power level and propellant, so that pulsed operation, though possible, does not appear necessary above ≈10 kW average power. This implies substantial reductions of the power conditioning and propellant feed system weight, complexity and potential life and reliability problems for the final system, and elimination of the uncertainties of pulsed testing. Since their performance does not deteriorate as rapidly with size and power level as that of the SF thruster group, much exploratory testing on the AAF thruster types can be done with scaled-down models in continuous operation. This alleviates somewhat the severe vacuum pumping and tank size requirements, at least for exploratory work, and the power supply problems for early space testing.

The AAF thrusters also have, however, in comparison with the SF-MPD group, a number of disadvantages or additional difficulties which must be overcome, starting with the present state of the art. These thrusters are more difficult to construct (magnet cooling), more expensive to test reliably (large vacuum facilities) and even more difficult to analyze and understand theoretically than the self-field thrusters, and the understanding of the flow discharge processes out in the plume appears still more vital here for full reliance on the test results.

The design difficulties of the AAF thrusters come from the need for a relatively strong magnetic field (on the order of

0.1 Wb/m^2, or 1,000 G); that is, a field coil with several hundred ampere turns. These coils are difficult to cool in the vicinity of a white hot radiation-cooled thruster, and they consume a substantial amount of power-- the more power, the higher the temperature of the coil metal. The quoted efficiencies in some of the earlier papers did not account for the power required for the coil. Seikel[55,57] suggested the use of cryogenically cooled (superconducting) coils as a possible solution, or radiation shielded permanent magnets up to limited field strengths. However, the testing of radiation shielded thruster designs also requires very good vacuum.

An additional problem with the AAF thrusters is that the magnetic field may interfere with parts of the spacecraft, though all MPD thrusters, especially pulsed ones, must be expected to produce some electromagnetic interference and noise (EMI). The dipole fields of the AAF thrusters, interacting with the earth's magnetic field, will also exert an appreciable disturbing moment on the spacecraft (on the order of 10^{-2} Nm or larger) which has to be compensated in some manner.

Both experimentally and theoretically, AAF thrusters are the least well understood and most open to question of the group. Because the energy conversion and plasma acceleration processes take place primarily out in the open plume (cf. Fig. 3), the surrounding gas in the vacuum tank always gets involved, even with very good vacuum (10^{-4} to 10^{-3} mbar). Krülle[39] cites spectroscopically measured jet plume velocities which were one half or less of those calculated from thrust and input mass flow. He ascribes this to ingestion of tank gas into the plume, which was also predictable from his theoretical model. In most experiments, including Krülle's, the measured thrust <u>decreased</u> with increasing vacuum tank pressures which would make the tank performance conservative, as long as the tank pressure was below 4 x 10^{-2} mbar. It is now generally believed (cf Refs. 57, 58, 60) that tank tests at least at 10^{-3} mbar will give reasonably reliable test data with possibly a little lower performance than in near perfect vacuum-- this in spite of the fact that with improving vacuum the plasma plumes still become substantially larger (see also subsection testing problems). In space, not only would the discharge extend much further out from the thrusters, but there is at least in principle the possibility that some of the accelerated plasma could follow along the field lines of the applied magnetic field radially outward or even back to the thruster, thus reducing the net thrust below the tank value. This effect, which in the tank is prevented by the tank gas and walls, represents an additional uncertainty of ground testing for this type of thruster.

Theoretically, the AAF-MPD thrusters, in the writer's opinion, are also the most difficult group of the three steady state thrusters discussed here. Numerous very different models for the theoretical or semi-empirical analysis of AAF-MPD thrusters have been formulated by several investigators, including Patrick and Schneidermann[59], Cann et al.[60], Harder[61], Jahn[62], Hassan[63], Bennett[64], Seikel et al.[55], Fradkin[65], Krülle[39,8] and their coworkers (where applicable). The older analyses (before 1968) have been summarized and partially evaluated by Nerheim and Kelly[5], but the best evaluations are those of Seikel et al.[55] and of Krülle[39].

Very briefly, four mechanisms for the magnetic gas acceleration and thrust production are being considered in these models (in addition to the expansion in the physical nozzle) as outlined in Ref. 55:

1. the self-magnetic forces (B_θ times meridional current) as previously described. This force can also contribute to the conversion of thermal into kinetic energy.

2. the actual Hall acceleration mechanism: an electrostatic ion acceleration axially and inward across the fringing field lines, carrying much of the current from the anode to the cathode jet. This may also be regarded[55] as a magnetic nozzle process ($j_\theta B_r$), where the azimuthal drift or Hall current $j_\theta \sim \bar{E} \times \bar{B}$.

Then there are the further magnetic nozzle ($j_\theta B_r$) expansion processes:

3. the conversion of the rotational kinetic energy of the plasma, resulting from the applied (meridional) current crossing the applied magnetic field, $v_\theta \sim \bar{j}_m \times \bar{B}_0$.

4. the conversion of thermal energy (more precisely enthalpy) into axial jet kinetic energy through the fringing magnetic field.

According to Seikel et al.[55], the free variables (mass flow, applied current and magnetic field) in any MPD thruster experiment can be chosen such as to make any one of these mechanisms important. However, both from their calculations and from those of Krülle[39] for typical cases of experimentally well-performing AAF thrusters, it turns out (as had previously been found experimentally) that mechanism (2) brings only a relatively small contribution to the thrust, so that the designation Hall Arcjet is no longer appropriate.

The mechanisms (2), (3) and (4) all require substantial azimuthal currents (j_θ) to flow in the jet plume, which in turn requires an azimuthally fairly uniform conductivity distribution. Mechanism (2) requires, in addition, an effective electron Hall parameter $(\omega_e \tau_e)_{eff}$ value much larger than unity in the anode jet in order to prevent the electrons from crossing the field lines from the cathode jet back to the anode, since for a major contribution by this mechanism, the applied (i.e. meridional) current in the outer anode jet must be predominantly ion current, and the arc voltage drop in that region must be correspondingly high.

From experiments, it had been found by many investigators that above certain critical values of the magnetic field strength to mass flow (or $B/\dot{m}$) ratios the discharges in these MPD thrusters are not azimuthally uniform but contain one or several rotating arc "spokes" or segments. Rotational frequencies in the range of about 30 to 600 kHz have been measured in many laboratories with magnetic probes and photocells, increasing with increasing magnetic field and decreasing with increasing mass flow. Comparable and sometimes higher rotational gas velocities were measured spectroscopically (with the Doppler shift method) in a few cases.

Under these conditions the discharge is believed to be highly turbulent[66], presumably due to the inherent instability of the arc itself and perhaps to the motion (cross flow). As a result, the effective transport properties of the plasma and thus the conductivity equation are strongly changed. As in MHD generator analysis[67], the effects of this turbulence are treated analytically with the simplest possible model, a postulated single frequency plasma oscillation, producing the so-called anomalous or Bohm diffusion of electrons across the magnetic field lines. Substituted into the Maxwell equations, the plasma oscillations result in a modification of the effective conductivity tensor, notably a strongly increased effective conductivity normal to the magnetic field (decreased effective Hall parameter) and decreased conductivity along the field[55,39,68]. With one constant in the theory, which can be adjusted to fit experimental data, it is shown that the effective Hall paramter $(\omega_e\tau_e)_{eff}$ will approach a value of 2 to 3 as the laminar Hall parameter $\omega_e\tau_e$ would become very large (of the order of 100). Seikel et al.[55], Krülle[39] and several others found that with these effective transport properties substituted into their analytical models, the important measured variables of their axial applied-field thruster experiments, such as Hall currents and potential gradients in the anode jet, could be satisfactorily matched. This is in spite of the use of axisymmetric models and clearly not rotationally symmetric discharges. Apparently the low effective Hall parameter values account roughly for the lack of discharge symmetry.

Extensive and difficult diagnostic programs[69,70] have been carried out to determine the variables in the plume which quantitatively describe each of the accelerating mechanisms (1 to 4 above). Among these variables are the azimuthal and meridional current distributions, the ion current and potential gradient in the anode jet, azimuthal and meridional velocity distributions of all ion and neutral particles, including tracer substances in the tank gas, degrees of ionization, electron densities, electron and heavy particle temperatures and the magnetic field distribution.

The comparisons of these measurements with the analytic model calculations indicated that the expected large Hall currents and large ion current fractions in the anode jets were not achieved in the best axial applied-field thrusters tested. Instead, the magnetic nozzle conversions of rotational kinetic and thermal energy modes dominated. To the writer's knowledge, the extent to which turbulence (anomalous diffusion) alone or the non-symmetric discharge reduced the contribution of the Hall current acceleration mechanism could not be determined.

In summary, the axial applied-field thrusters have achieved reliable thrust efficiencies of about 35 percent with ammonia at ≈2500-3000 seconds specific impulse and power levels for 30 kW class thrusters[55]. Even much smaller thrusters have achieved fair to good performance, especially with lithium. Much higher but less well substantiated older performance values are in the literature. Thruster life times on the order of 500 hours have been reported, in one case terminated by a cathode problem, possibly due in the author's opinion to thoria depletion of the tungsten. Improved performance is expected at higher power levels, possibly with pulsed operation[57]. For the applied-field thrusters, permanent magnets or superconducting coils (both thermal radiation shielded) were suggested, but the precise design has not yet been worked out. The coil location(s) and precise field shape in the electrode space are critical for good performance[55]. For performance testing of these thrusters, tank pressures of 10^{-3} mbar (10^{-1} Pa) appear adequate[58], though minor effects due to some entrainment of tank gas into the plume cannot be ruled out. Early space tests for both performance and electrode erosion verification appear essential for these thrusters.

The Low Density Hall-Ion Thrusters

These thrusters have been studied mostly at Avco Everett (Patrick and Janes[56]), at the DFVLR Stuttgart (Schreitmüller[72], Lindner[73], and also Krülle and Zeyfang[8]), in the Soviet Union and early on at EOS-Xerox. Mercury and argon have mainly been used as propellants. According to the references just given, the performance of these thrusters appears comparable with that of the higher density AAF thrusters, even in relatively small sizes. Krülle and Zeyfang quote c_e values of 20 to 30 km/s and efficiencies of 30 to 40% with noble gases at power levels of 1.4 kW, i.e. performance at least comparable with the best AAF thrusters. They appear to have no particular cooling or life problems (up to 100 hours) in the low kW sizes. Like ion engines, they have a plasma generation chamber separate from the accelerator, but a partially ionized plasma is acceptable. The accelerating mechanism appears to be somewhat better understood than that of the AAF thrusters, except for the previously mentioned discrepancy (by factors of up to 100) between the actual effective and the predicted Hall parameters. The plasma fluctuation or anomalous diffusion analysis model mentioned under the AAF thrusters is used here also to match experiments[57, 59].

These thrusters appear somewhat less promising than the AAF or SF thrusters for large power applications for three reasons:

1. They have substantially lower power densities, hence would be bulkier and heavier, especially in view of the magnets, i.e. in size and weight one step closer to the ion engines.

2. They require much better vacuum for adequate testing than the other two contenders, typically 10^{-5} mbar, which in practice requires cryo-pumping or very large diffusion pumps.

3. In the size range where these thrusters might be likely contenders, good ion engines already exist.

Research work on Hall-ion thrusters is facilitated by the fact that acceptable performance has been achieved already at relatively low power levels. However, performance improvements must be expected with increasing size, since wall losses appear to be a dominant factor. The design principles which have achieved such spectacular advances in the performance of ion engines during the past decade could be transferable to the Hall-ion thrusters.

Electrodes and Thruster Cooling

Present Level of Understanding of Electrode Attachments

An overwhelming amount of literature from the past 50 to 60 years exists on arc electrodes, as well as many excellent critical summaries in perhaps fifty textbooks and monographs (e.g. Refs. 74 to 78). However, the vast majority of past and present electrode and plasma/wall interaction studies pertain to conditions and regimes very different from those in MPD thrusters. Also, much of the information in the literature is either predominantly empirical or treats theoretical models for specific situations (e.g. macro- and microspots of vacuum arcs on cold copper electrodes), and is thus not directly transferable to MPD thruster electrodes. Thus, in spite of the qualitative insights into electrode physics gained from this copious amount of previous work, the present quantitative understanding of the electrode attachments for MPD propulsion is still rather limited-- certainly inadequate for thruster design and optimization purposes. Even those empirical or theoretical criteria which do exist for predicting precisely under which conditions the transition from diffuse to spot attachment on anodes or cathodes will occur cannot yet be utilized for design until adequate methods for calculating the plasma conditions near the electrode (from the midstream flow discharge solutions) become available. As far as erosion and additive (e.g. thoria) depletion are concerned, the possibility of achieving required life times of several months to years is still in question, since there is insufficient experience with such long life times; few electrodes of other applications have such long life requirements.

Importance of Electrode Design

Understanding of the electrode arc attachment regions is essential for successful thruster design and operation in view of the following problems:

- electrode erosion, which is critical both for thruster life and for spacecraft surface contamination

- transition from uniform to spot attachment on the anodes ("onset" effect) which limits the performance (I_s and η) and causes both excessive erosion and electromagnetic noise (EMI)

- electrode losses, which critically affect thruster efficiency and cooling requirements (i.e. thruster weight)

Erosion of cathodes is at present, in most cases, still excessive from the viewpoint of thruster life alone, while the limits imposed by spacecraft contamination cannot be stated generally. Anode erosion may be manageable, as long as uniform (spotless) attachment is achieved.

The formation of macrospots on the electrodes, i.e. the shift from diffuse and more or less even arc distribution (which for the cathode may include an even distribution of microspots) to localized concentrated arc attachments is, of course, akin to the arc filament formation in the midstream region but influenced by the specific boundary conditions at each of the electrodes. It must therefore be discussed separately for each electrode type. In MPD thrusters, macrospots cause excessive (destructive) erosion on each of the electrodes-- though this is sometimes tolerated with pulsed thrusters (as indicated in a following paragraph). Arc filamentation in the midstream tends to lower the thrust efficiency. It is not yet clear to this writer what relationship exists between electrode (e.g. anode) macrospot formation and uneven midstream discharges.

Electrode losses (notably on the anode) dominate the efficiency limits of small MPD thrusters, where the electrode losses can take up 50 to 60 percent of the overall voltage drop. For larger thrusters (MW regime) and with strong applied fields this loss is less dominant, but it does strictly limit the current densities on anodes if simple radiation cooling is required.

Electrode Voltage Drops and Loss Distribution

In most MPD thruster designs, except the Hall-ion thrusters, as in many other arc devices (like short-arc noble gas and carbon arc lamps, welding arcs, etc.) a major portion of the ionization takes place at or near the cathode. In these cases, there is therefore a relatively large potential drop in the arc near the cathode, much of which must however be charged to ionization (frozen flow loss) rather than as a cathode loss. The plasma thus formed is blown as a cathode jet toward the anode, where there is usually no mechanism or need for additional ion generation if the ion density there is sufficient to neutralize the imposed electron current. As long as this is satisfied, the anode potential drop can be quite small (even negative). In spite of this, the overall losses at the anode can be very large, as will be shown. Another energy loss which must be invested at the cathode is the electron work function Φ ($\approx$4 to 5 V for most metals), which appears as an additional heat load at the anode and is frequently accounted for as an anode loss, though actually it should be considered as a usually unrecoverable "cost" of running an arc.

MPD Thruster Cathode Phenomenology

This paragraph will deal only with refractory metal (W, Ta) cathodes, at present the only ones known to be potentially suitable. For larger MPD thrusters, the so-called high current regime from a few hundred ampere into the high kiloamperes and gas pressures at the cathode from a few hundred millibar on down are applicable. The cathode appears currently as the most critical component from the viewpoint of thruster life.

Cathodes of continuous thrusters (and those of some other arc devices) can operate, in their design range of conditions, with an apparently spot-free diffuse arc attachment covering a fairly large area. To reach this optimal (from the viewpoint of erosion) condition, the cathode must be of the right material (e.g. thoriated tungsten), be able to reach the required temperature (2,600 - 3,300 K) over a sufficient area (implying adequate but not excessive cooling) and be surrounded with sufficient gas density for the imposed current. This condition will be described more closely below. Under all other conditions, like continuous thrusters during warm-up, cathodes with inadequate gas pressure or design or "aged" material (thoria depletion) as well as cold cathodes of pulsed thrusters operate in some form of spot *mode*. This involves local melting and substantial vaporization of cathode material, as will be discussed.

The conditions for diffuse cathode attachments ("thermionic cathodes") in MPD thrusters are most comprehensively treated by Hügel and Krülle[79] together with Cann and Harder[80] and Bade and Yos[81], the last one being limited to atmospheric pressure experiments. Ref. 79 also examines the possible effects of the B_θ self-field on macrospots of cylindrical cathodes. Some effects of axial applied magnetic fields (for AAF thrusters) on the cathode tip pressure distribution are treated by Malliaris[82], but generally there are insufficient data to predict the effects of magnetic fields on the erosion due to cathode spots.

Without attempting to go into the complicated and only partly understood cathode processes, a few essentials will be discussed. All cathode attachments (diffuse or spots) have a space charge or fall zone of varying voltage (V_c) and thickness of roughly an ion mean free path, with voltage gradients at the surface (E_c) of the order of 10^5 to 10^6 V/cm, sufficient to produce some field emission (under certain conditions the effective local fields can be much stronger; see below). Ions from the adjacent arc column (neutral plasma) diffuse or are drawn into the fall zone where they are accelerated to bombard the cathode. This ion current density j_i, the dominant heating mechanism, produces a net heat flux to the surface of roughly

$$j_i(V_c + \frac{3}{2} kT_i + V_i - \Phi_0) \qquad (14)$$

where T_i is the ion temperature in the adjacent plasma, V_i an effective average ionization potential of the arriving ions and Φ_0 the cathode surface work function. The surface accomodation coefficients have all been set unity as is customary in view of other uncertainties (this may not be admissible for very light ions like H^+, He^+, N^+). The fraction (a) of the total current density $j = j_e + j_i$, which is supplied by the ion current, generally ranges between 8 and 15 percent but can be much higher. Its maximum value can be estimated from the ion diffusion rate in the adjacent plasma. Additionally, the ion current fraction is bounded by various approximate energy relations. First, the minimum value of j_i is set by the required heat input flux, expression (14), which must be larger than the electron emission energy $j_e\Phi_0$. Second, some analysts assume that all the ions drawn in have been produced by the electrons coming from the cathode into the plasma, implying the energy relation

$$j_e V_c \approx j_i V_I$$

so that

$$a = \frac{j_i}{j} \approx \frac{V_c}{V_c + V_I} < \frac{1}{2} \qquad (15)$$

Thus the ion bombardment heat, and to some extent the field of the fall space, supply the energy for the so-called field-enhanced thermionic emission of the electrons. While there is no complete theoretical solution for combined thermionic and field emission, a first order approximation for predominantly thermionic emission is the Richardson-Schottky emission equation. In a simplified form applicable for MPD arcs (following Refs. 79 and 81, retaining the dimensional constants of those authors), this is:

$$j_e = AT_s^2 \exp\left(-(11{,}609\ \Phi_0 - 332\ V_c^{1/8}\ M^{1/8}\ j_i^{1/4})/T_s\right) \quad [\mathrm{A/cm^2}] \tag{16}$$

In this A, the Richardson "constant" should have a universal value (120.4 $\mathrm{A/cm^2K^2}$) for pure surfaces of simple crystals of all metals. T_s is the cathode surface temperature, M the dimensionless molecular weight of the propellant; j_i is here also in $\mathrm{A/cm^2}$. The second term in the bracket of the exponential represents the amount by which the work function term would be effectively reduced through the field emission (see below).

In view of the approximate nature of the equation and material property uncertainties, this is usually treated as a semi-empirical relation in which A and sometimes also Φ_0 are adjusted to fit representative experiments.

Hügel[79] derives values of A between 0.03 and 0.5 for argon, and 0.7 to 3.0 for hydrogen, in each case increasing values with increasing pressure. Bade and Yos[81] suggest A = 1 $\mathrm{A/cm^2K^2}$ and Φ_0 = 2.6 V as average values for thoriated tungsten, but they could not determine pressure dependencies.

It appears likely that Eq. (16) underestimates the field emission contribution and thereby the pressure effect, the second term in the exponent of Eq. (16), would increase with increasing pressure and thinner fall zones. Ecker[76] and others who put forth theories about thermionic + field (T-F) emission have pointed out that large local peaks in the field are likely to occur due to the inherant surface roughness (on a molecular scale) and plasma fluctuations. These effects could materially increase local field emission rates, as is probably the case also in spots.

In summary, Hügel and Krülle[79] obtained with diffuse cathode attachments current densities to 2,000 A/cm^2 with argon and to 5,000 A/cm^2 with hydrogen at 100 mm Hg and 1,000 A, at $T_S \approx$ 2,800 to 2,900 K and 3,200 to 3,300 K, respectively.

The attainable current densities should increase with the pressure and the current. Fall voltages V_C were 3.5 to 8.5 V for argon and 3.3 to 4.5 V for hydrogen (both surprisingly low), decreasing with increasing pressures and currents. The net heat fluxes into the cathode ranged from 0.6 to 2.5 kW/cm^2 for argon and 1.7 to 4.0 kW/cm^2 for hydrogen, the largest values always at the highest pressure ($\approx$ 100 mm Hg). Note that the limit for reradiation at the melting point is 1 kW/cm^2, so that generally a larger area than the arc attachment must radiate to cool the cathode. The net heat flux equivalent voltages were 5 to 1 V for argon and 2 to 0.7 V for hydrogen, decreasing with increasing current for both and pressure for argon.

These results agree substantially with those of other investigators, except that Bade and Yos experienced more of a tendency of spot formation, presumably due to their higher gas pressure (1 bar). Thus, thoriated tungsten cathodes can operate in the spot-free thermionic (i.e. T-F) mode, with current densities up to at least 2 kA/cm^2 with minimal losses, at $\approx$2,800 K, the highest still acceptable surface temperature from the viewpoint of vapor pressure.

Much higher current densities (to at least 5 kA/cm^2) are possible, but at excessive temperatures for long life (i.e. above 3,000 K). Aging of the material (presumably thoria depletion) was found to be a serious problem in some cases[82].

Microspots and/or macrospots appear on thruster cathodes whenever the metal work function, temperature and/or the ambient plasma density are insufficient to allow the imposed arc current density to be supplied with distributed T-F emission alone. Molten areas like macrospots can also occur due to inadequate cooling. However, also under conditions where spot-free thermionic cathode operation should be possible, the arc sometimes changes to a spot mode for as yet unknown reasons[81], but presumably as the lower voltage solution. The existence of two possible solutions in many cathode and anode attachment cases was already shown by Ecker[76].

The cold cathodes of typical vacuum arcs always operate with microspots. The cathodes of pulsed (quasi-steady) MPD thrusters presumably also are fairly evenly covered with a large number of microspots, as evidenced by the roughness after a few shots. The cathodes of continuous thrusters during warm-up and overcooled cathodes generally operate with macrospots[79,81], which may be or at least have initially been structured, that is, clusters of microspots as described below.

Cathode microspots are the subject of much academic speculation and vacuum arc experiments. In spite of the many congresses held on vacuum arcs in the past decade, no up-to-date summary and evaluation of all this work on cathodes appears to exist. The most frequently discussed microspots have extreme current densities (on the order of 10^8 A/cm^2 and even higher) which lead within nano-seconds[83] to local vaporization and the formation of a minute crater (on the order of 10^{-4} cm diameter). High pressures, on the order of 100 bar, and vaporization rates in these craters have been calculated. The emission in these craters must be predominantly field emission. The hypothesis of field emission preceeded by melting is supported by recent findings in the Soviet Union. Fursey[84] photographed microscopic needle sharp cones, similar to the Taylor cones observed on the knife edges of field-emission ion thrusters, around the edges of cathode spot craters.

These dense current filaments are very unstable, in spite of the emerging vapor jets (cf. Schrade[85]). This accounts for the relatively small remaining craters and high frequency (ca. 10 megacycle and even some gigacycle) EM noise from such arcs. In spite of the short residence times, the overall erosion rates cited are very high, ≈40 to 100 μg/C, consisting of both vapor and so-called macroparticle ejection.

For thruster application, some older papers on both micro- and macrospots are more applicable. Zykova et al.[86] obtained from pulsed discharges in a 2.5 mm arc gap and cold electrodes (very different from a typical thruster situation) very detailed time-resolved optical and erosion trace observations of cathode and anode spots on tungsten, copper and other metals. Test gases were xenon, argon and helium, at pressures from 400 down to 10^{-4} mm Hg, with pulse durations up to 2.6 ms, and currents from 5 to 4,000 A. Three kinds of microspots were observed, including the fast moving ones just mentioned (spots of the first kind in the Russian literature). These 10^{-3} cm diameter spots appeared "instantly" at all currents and pressures covered, and left only slight damage, i.e. relatively small and shallow traces, on the higher melting metals. The ≈5 times larger and much slower moving (10 to 100 cm/s) spots of the second kind appeared 100 to 200 μs later, left "noticeable damage in the form of fused craters on all metals except tungsten," where they appeared only at the highest pressure (400 mm Hg). The equally large but slow moving or stationary spots of the third kind appeared only at 10 mm Hg and below, and only in clusters of 10 to 30 spots which were about a spot diameter apart. These low mobility clusters of spots carrying ≈200 A on copper and 200 to 300 A on tungsten (with 10 to 20 A per single spot), left "on all metals" relatively wide and deep fused craters. Thus the kinds of spots observed depended on all the variables, including the time during the pulse. Significantly, some of the microspots and clusters (the most damaging ones) were stable and nearly stationary. The area covered by spots increased with time as a function of current, metal and pressure, leading in some cases to the macrospots observed by other investigators with continuous arcs. Spot areas on copper at 1,300 A after 700 μs correspond to current densities of 6×10^4 to 3×10^5 A/cm^2 at 10 mm Hg and 10^{-4} mm Hg respectively, similar to the macrospot observations of Ref. 79.

A most important finding for pulsed thrusters (and possibly for warm-up of cathodes generally) was the strong dependence of the erosion trace volumes on the pressure (for a given gas and metal) with a pronounced minimum. The data given for copper correspond to 100, 2.5 and 25 μg/C at pressures of 10^{-4}, 100 and 400 mm Hg, respectively. No erosion volume data were given for tungsten, except that they were generally smaller. Such numbers (notably the larger ones) are typical of other vacuum arc erosion data in the literature for copper. The existence of such a pronounced minimum (by a factor of 10 to 100), if it should also exist for tungsten, could be most significant for thruster design. The conclusion would be that there may be an optimal pressure regime for pulsing (and possibly even for thruster warm-up) considerably above 10 mm Hg to avoid the clusters but below 400 mm Hg with tungsten to avoid the spots of the second kind. Some pulsed thrusters may (by design or by luck) operate in that regime.

It appears most essential that extensive experimental programs on thruster cathodes-- both with diffuse and with spot operation-- be carried out, to supplement the very few applicable studies to date. In particular, the effects of the gas flow and pressure and of the strong self-fields and possible applied fields on cathode operation and erosion must be systematically explored under typical thruster conditions, in addition to the material aging (additive depletion) effect on which only sparse information is available.

Thruster Anode Phenomenology

In a long arc without axial flow, there is a small ion drift (i.e. positive ion current) away from the anode. These ions must be replaced at the anode to provide plasma (space charge) neutralization and thus allow the main electron current to reach the anode. To produce these ions, either from ambient neutral atoms or from anode material, an anode arc contraction and fall zone sets itself up having, at the lower densities, a voltage close to the ionization potential of the substance to be ionized. Normally an anode spot develops, from which anode material is evaporated and ionized, because in practically all (except alkali vapor) arcs the anode metals are more easily ionized than other arc gases. These anode contractions, while less concentrated than those at cathode spots, produce self-field anode jets (similar to those from cathodes), which increase the ion flow and ion current contribution away from the anode above the normal ion drift of an arc column.

Spot-free anode functioning (at least without macrospots) is, however, possible in many MPD (except possibly Hall-ion) thrusters, in thermal arc jets and many other arc devices such as short-arc lamps. There the plasma is generated mostly near the cathode and in some cases also in the midstream discharge and is blown toward the anode surface. Under these conditions there is normally no need for ion generation near the anode and the anode potential drop becomes zero or even negative as predicted by the classical anode theory for this case[87].

For such MPD thrusters (with rear cathode), Hügel[7] has delineated three regimes of spot-free anode operation. The first regime is characterized by a sufficient supply of ions for neutralization and replacement of those drifting (or blowing) away, and a sufficiently large natural diffusion of electrons toward the anode to provide the current drawn. If the normal electron diffusion exceeds the electron current drawn by the external circuit, the anode surface, if it is "cold", becomes slightly negative relative to the plasma, similar to a cold floating probe in a hot plasma. (If it is hot, it will reemit some electrons.) Depending on the electron temperature this could be 3 to 5 V negative, though a cool gas wall boundary layer will also somewhat impede this electron flow.

As the current is increased at constant mass flow rate (or $J^2/\dot{m}$ is increased by any path), a second regime is reached through the increased magnetic pinch effect and increased relative current density, i.e. closer approach to "onset" conditions. This second regime described by Hügel is one in which the natural electron diffusion no longer supplies the demanded current density to the anode, though the ion supply to the anode is still sufficient. This anode regime exists for the thruster type and operating regime treated by Hügel and coworkers, where the ion supply carried by the jet to the anode exceeds the demand there longer (with increasing $J^2/\dot{m}$) than the natural electron diffusion satisfies the required electron current. In this regime, a positive voltage drop develops near the anode, increasing linearly with increasing J, from 0 to ≈6 to 8.5 V for argon.

This results primarily in increased electron drift current through potential gradient plus small contributions through ion generation (δn_e) and temperature increase (δT_e). Since in this regime the increase in anode voltage is proportional to the increase in current density (δj_e) imposed, this voltage (loss) rise should be reducible by increasing the anode area exposed to the plasma flow.

With further increase in the current/mass flow parameter ($J^2/\dot{m}$), a third regime of theoretically possible spot-free attachments is reached in these thrusters, where now also the rate of ion flow toward the anode becomes insufficient. In this regime, which Hügel calls that of the genuine or classical anode fall, the anode potential must instantly increase to the potential drop required for ion generation.

Using classical anode theories[88,89] and the "field" ionization process appropriate for the low density regime, Hügel calculated an anode fall voltage of 12.4 V for singly ionized argon and slightly lower values (10.7 V and 8.8 V) for xenon and krypton. Double ionization requires roughly twice those values.

Since all practical electrode materials have lower ionization potentials (below 8 V), a diffuse anode attachment of this last type is unstable and will break down to anode spots with anode vaporization, the normal operating mode of arc anodes described initially which results in a lower anode potential. Also the third regime just described, where the plasma density near the anode approaches zero, is in fact close to or at the "onset" or critical $J^2/\dot{m}$ point where probably various other arc instabilities appear in addition to the anode spot formation. The beginning of this third (i.e. the classical) anode operating regime is therefore (also in Hügel's designation) the onset point, at least for those thrusters where "anode starvation" due to the pinch pressure gradient is dominant.

On the formation and nature of anode spots, the following will give some insights. For the much more frequently investigated vacuum arcs, Miller[90] reports from different sources evidence of five different anode operating modes. They are, first, the diffuse, fully passive attachments (similar to those described above but at lower pressures); second, a diffuse attachment with some anode material backflow presumed to be sputtered by ions from the cathode; third, an attachment regime he called "luminous footpoints" with occasional point melting at sharp points or edges, with temperatures near melting (≈1370 K on copper) but acquiring still relatively modest damage; this regime of the vacuum arcs, which is accompanied by an increase in arc voltage and arc noise (voltage fluctuations) appears to be closest to the onset point phenomena of self-field thrusters; fourth and fifth, the regimes of "true anode spots" which come in a wide range of sizes and current densities depending on all the variables of the experiment (gap geometry, metals, current, time, background gas) as described by many investigators. One of these modes, the short pulse or initial one (appearing already in microsecond pulses) shows many small spots of high current densities, individually or in clusters. The number of individual spots depends on the overall current (among other things). The other or large spot mode appears to be the result of longer time and high current. These large spots may be internally structured, but they produce a common anode jet. Common to the true anode spots are surface temperatures near boiling ($\approx$2,500 to 3,300 K on copper) with severe erosion of the anodes even during relatively short pulses (of the order of milliseconds as used by the switch-gear people), except perhaps on tungsten.

Zykova et al.[86] report on two kinds of anode spots. The second kind, applicable to the MPD regime (p$\approx$10 mm Hg and below) of 10^{-3} cm diameter and 10 to 20 A per spot ($\approx 10^{7}$ A/cm^{2}) appeared in a cluster or region of the order of a square millimeter. This cluster moved at 1 to 10 m/s by formation of new and extinction of old microspots which were individually immobile. Typical average current densities of the clusters were $\approx$2 to 8 x 10^{4} A/cm^{2}. Traces left on all metals (except tungsten), after millisecond pulses, were individual round fused craters, $\approx 10^{-5}$ cm^{2} each, spread over 10^{-2} to 10^{-1} cm^{2} total area. On tungsten the traces were "barely noticeable".

True anode spots are generally still more damaging than cathode macrospots, for two reasons: first, they produce for a given current density roughly five to ten times higher net heat loads; i.e. $\approx$20 V times the spot current density for argon compared with $\approx$2 to 4 V for cathode spots; second, unlike cathode spots, the anode spots do not go away with heating of the electrode. In continuous self-field thrusters, anode spots are totally destructive and must absolutely be avoided because they remain stationary. With sufficiently short pulses and/or magnetic fields strong enough to move the spots, anode spots can possibly be tolerated in any pulsed and in continuous applied-field thrusters, thus particularly on high melting (tungsten, molybdenum) electrodes, but still at the penalty of some erosion. Strong magnetic fields, which through arc movement prevent the almost instant destruction of anodes, are known to promote in arcjets the formation of anode spots where otherwise diffuse attachment may have been possible (cf. Schall[91]).

Electrode Erosion

The limits of allowable electrode erosion will first of all be set by thruster life requirements. Much lower erosion limits could be imposed by spacecraft surface contamination if the eroded material vapor can end up in the form of slow ions (e.g. by charge exchange) in the spacecraft vicinity and deposit itself on charged insulator surfaces of the spacecraft. This latter erosion limit is difficult to estimate in a general way, but all condensible vapors near a spacecraft constitute a surface contamination hazard.

The allowable erosion, from the viewpoint of thruster life, depends on the life requirements-- typically 1,000 to 10,000 hours for near-earth missions-- and the thruster design, type and size. Some typical numbers for a 10 N continuous self-field thruster (4kA, 1 g/s argon) for 3,000 hours life (consuming 4.3×10^{10} C, 11,000 kg propellant) would be:

	cathode (W)	anode (Mo)
allowable erosion weight loss (W_{er}), g	50	250
W_{er}/Coulomb, µg/C	1.2×10^{-3}	5.8×10^{-3}
W_{er}/propellant weight	4.6×10^{-6}	2.3×10^{-5}

This corresponds to a few millimeters per year (or the order of 10^{-7} mm/sec) material loss from the faces of the electrodes exposed to the arc for that size and type of thruster. Note that this is less than the normal vaporization rate of tungsten at 3,000 K, and 1/2,000th of that rate at the melting point (see Fig. 9). Thus, allowable current specific erosion rates would be estimated at 1 and 6 x 10^{-3} μg/C for cathodes and anodes, respectively. Continuous applied-field thrusters of the same size and thrust require ≈2 to 5 times less current and could therefore be allowed slightly larger erosion rates per coulomb. Pulsed self-field thrusters tend to be larger for the same thrust (factor ≈5) and require roughly 10 times smaller average currents (or coulombs); therefore, they could tolerate roughly 50 times larger erosion rates per coulomb (say on the order of 6 x 10^{-2} μg/C). The allowable erosion to propellant consumption rates could be roughly 5 times larger for the pulsed self-field thrusters than for the continuous types (because of size difference).

The possible erosion mechanisms for both cathodes and anodes are

- macro particle ejection (spitting) and massive vaporization (boiling) from molten pools under macrospots

- localized vaporization (boiling) at microspots

- vaporization from larger molten surface areas (without boiling or spots)

- sputtering due to ion impact

- continuous surface vapor loss due to excessive average surface temperature without melting.

With brittle metals (e.g. tungsten) there is, in addition, the possibility of cracking due to thermal fatigue.

To prevent substantial overall vaporization at excessive average surface temperatures, the average electrode tempera-

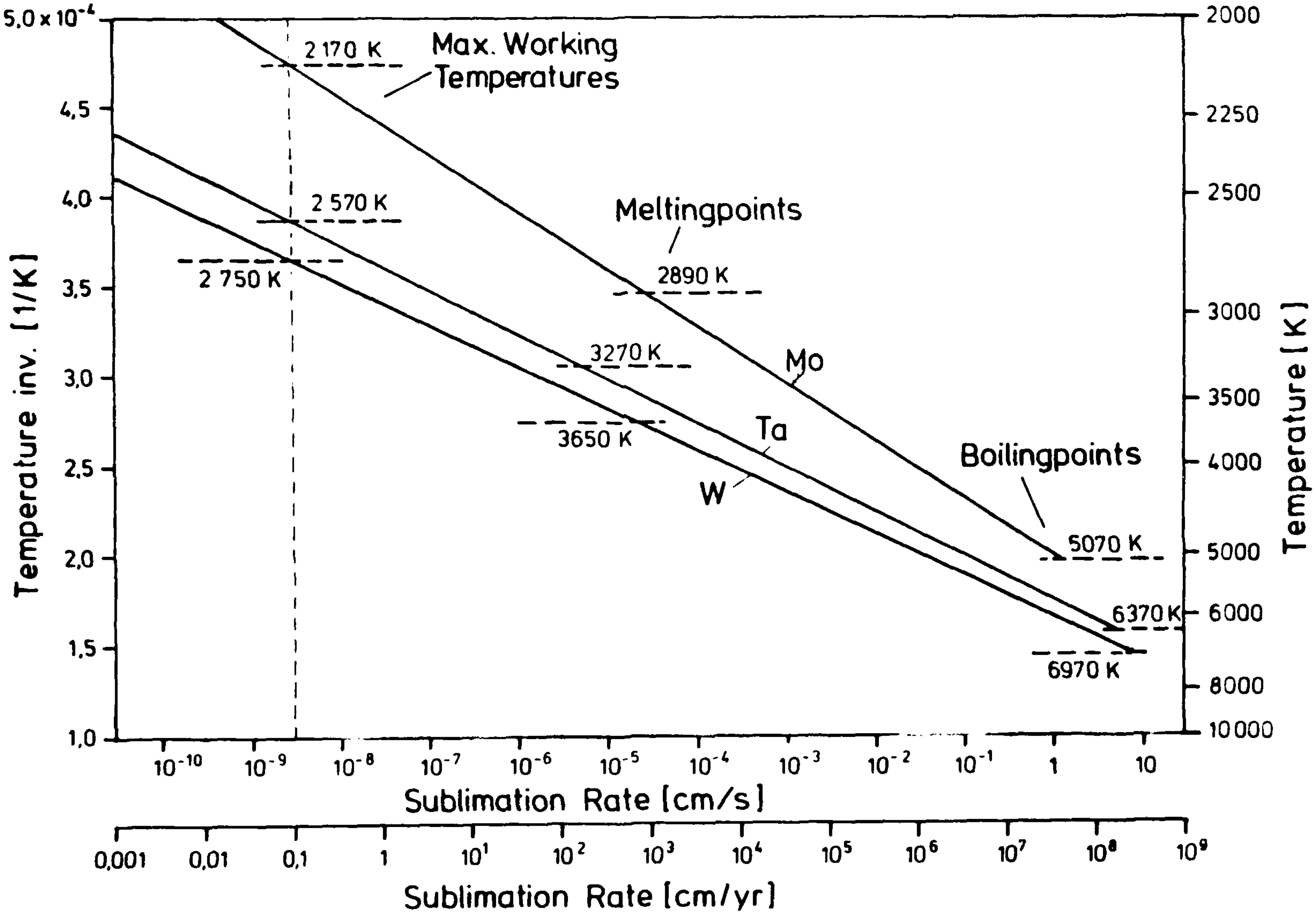


Fig. 9: Sublimation Rates vs. Temperature for Typical Electrode Materials.

tures must be controlled by adequate cooling. Typical average temperature limits for three electrode metals, as dictated by their vapor pressures, are given in Fig. 10. Generally, the average electrode temperatures should be held below those given for, say 1 mm surface loss per year to allow for local overheating. This should be readily achievable with anodes, but with cathodes the temperature given is too low for effective thermionic emission, so that the arc may concentrate itself and produce higher surface temperatures in spite of radiation cooling.

Sputtering erosion of the cathode surface is at present expected to be a small effect compared with the dominant erosion rates due to various types of spots. On the thruster anodes (unlike those of vacuum arcs), sputtering, if it occurs at all, is expected to be of negligible magnitude, since impact of high energy ions on the anodes is unlikely here.

This leaves the various kinds of spots (and possibly footpoints) as the major sources of erosion. Because of the scarcity of applicable data from thrusters, numerous other arc spot investigations have been cited above to indicate the types of spots observed, the wide ranges of erosion rates measured and some of the dependencies. However, it must be remembered that none of the experiments cited, other than those with thrusters, are really applicable in terms of the magnetic field, the electrode temperature, the gas flow and the geometry, all of which can strongly affect the spots and the erosion they produce. Only very few of those other data cover tungsten and molybdenum electrodes.

<u>Cathode erosion</u> due to spot clusters was quoted (from Ref. 86) at 1 to 100 μg/C on copper, and was strongly pressure dependent. For tungsten cathodes, spot erosion was indicated to be less, with no numbers given. In other vacuum arc literature, erosion rates of 40 to 100 μg/C are typical for microspot studies, mostly on copper.

The few existing data on pulsed self-field MPD thrusters[41] give values of ≈3 to 11 μg/C-- the lowest for argon, the highest for hydrogen-- all with standard thoriated tungsten electrode material. These last rates are roughly 100 times larger than those estimated to be allowable for 3,000 hours of thrusting life. Much lower erosion values (0.3 μg/C) have been obtained recently[92] with barium oxide doped tungsten, even with hydrogen, but the life expectancy of this relatively porous material at elevated temperatures remains to be established.

For continuous self-field and applied field thrusters, there are few accurate measurements, but it is known from various life tests that the cathode erosion rates in well functioning self-field and applied-field thrusters were substantially smaller than the values given above. Indications from electrodes with 50 and more hours (or about 4×10^8 Coulomb) of life without failure are that cathode erosion in continuous operation must be substantially below 0.1 μg/C or 1 g/hour at 3,000 A. The few past life tests of such devices confirm this. Vacuum arc-like microspots could hardly play a major role at these low erosion rates.

The major cause of tungsten cathode erosion in continuous thrusters is the droplet expulsion ("cathode spitting") during warm-up due to macrospots, for instance ≈10 to 30 mg average per cold start for a 10 N laboratory thruster. Since this is a statistically varying quantity with some uncertainty and the steady erosion rates are expected to be on the order of 10^{-2} μg/C (or 130 mg/hr at 4 kA), runs of many hours are required to determine the steady state erosion rates reliably.

The extremely low cathode erosion rates achieved with some arc devices, e.g. the 760 hour life test[19], are astonishing when one considers the normal vaporization losses (Fig. 11) and the fact that some cathode vapor, including that from sputtering, is always involved in the cathode region processes. It has been speculated[80] that the same particles participate in this process several times by being ionized and then reattracted to the surface.

A serious life limiting process in cathodes could be the depletion of thoria or other additives, which appears to have terminated at least one life test after 560 hours[93]. This is difficult to detect in cathode sections after, say 50 hours of life and may require long life tests for accurate determination and special provisions to counteract, if possible.

Anode erosion in MPD thrusters varies over wide ranges, but there are only few accurate data available. Continuous self-field thrusters must be operated at currents some safe margin below the onset or anode voltage rise regime, to avoid rapid destruction. At the first sign of any onset phenomena like spots, foot points and/or fluctuations, the current is instantly turned off or reduced. With diffuse anode attachment the anode erosion rates, if any, are so small as to be difficult to detect. On the contrary, on cooled copper anodes some grey deposits (presumably of tungsten cathode vapor) are observed, which would tend to build up rather than erode the anodes.

Continuous axial applied-field thrusters generally have rotating current spokes, and these will in some cases contain spots moved by the magnetic field. This leaves traces, but no accurate erosion rates are known to the author.

In summary, the erosion of tungsten cathodes appears acceptable in the thermionic regime, as long as the additive (thoria, etc.) remains present at the surface. For cooler cathodes (typical of present pulsed thrusters) and for frequent cold starts, cathode erosion still appears excessive.

Anode erosion appears manageable with diffuse attachment (i.e. far from onset conditions) and possibly also with spots in pulsed and/or axial applied-field thrusters, using high melting metals.

Electrode Losses and Cooling

The electrode losses are hard to separate out precisely because of the complicated and interlacing energy transfer mechanisms which occur at the electrodes, as was already briefly discussed. However, without attempting to resolve all these still open questions, we will attempt to put down a few orders of magnitude on the net heat loads and draw conclusions on limiting permissible current densities and cooling capabilities.

For typical thermionic cathodes of continuous thrusters, the net heat loads have been given above (from Ref. 79) as 1 to 5 W/A and 0.7 to 2 W/A for argon and hydrogen, respectively, with the lower values applying to the higher current densities. Many other propellants fall somewhere in between. The net heat flux values ranged from 0.6 to 2.5 kW/cm^2 for argon and 1.7 to 4.0 kW/cm^2 for hydrogen, depending on pressure and current density.

Radiation cooling at the allowable temperature for long life (2,800 K, with emissivity 0.35 for tungsten) removes only about 120 W/cm^2. Therefore, purely radiation-cooled cathodes in continuous operation can be loaded only to about 50 to 100 A/cm^2, a factor of 10 to 100 lower than the possible current densities, according to Refs. 79 to 81. As was the case in those experiments, the radiating area must be very much larger than the area of arc attachment. In a radiation cooled thruster, the cathode will receive some additional radiative heat load from the anode and other hot parts of the thruster. If higher current densities than those given above are required (or unavoidable), additional cooling must be provided. With a few propellants (e.g. H_2, NH_3, Li) this can be done regeneratively. Otherwise, heat removal by simple conduction or (for larger thrusters) by a liquid metal heat pipe must be provided.

For pulsed thrusters, the full available current densities of up to 2 kA/cm^2 may be utilized with the duty cycle adjusted to allow radiation cooling. In pulsed thrusters at present, the measured values of the net heat going into the cathodes are much higher than for continuous thrusters, amounting to about 15 to 20 V times the current, or about 15 percent of the total power for ammonia, and similar values for hydrogen. This must be due to the high current densities and low cathode temperatures, which lead to high cathode fall voltages.

In any case, cathode cooling does not appear to present insurmountable problems if macrospots can be avoided.

The anode net energy input equation in simplified form as given by Hügel (assuming the total current being carried by the electrons) is:

$$QA = J\ (V_a + 5/2\ k\ T_e + \Phi_0 + V_{conv}) \tag{17}$$

which must be removed by cooling. The first three terms on the right are energies brought in by the electrons, while V_{conv} is the normal (heavy particle) convective heat transfer, and radiation from gas and cathode is neglected.

For liquid cooled copper anodes of continuous thrusters, V_a is ≈ 0 to 10 V, the upper value being applicable near onset. With $(5/2)kT_e \approx 3.5$ V, $\Phi_0 = 4.4$ V and $V_{conv} \approx$ 2 V, the net anode heat loads run from about 9 to 20 V times the current, where the higher value applies to the interesting performance points. With optimized high performance thrusters, the values can run considerably higher.

If the anode is to be radiation-cooled, then the average heat flux values and thus the average current densities are quite limited. At the maximum allowable temperature for long life, tungsten can radiate only about 120 W/cm^2 and molybdenum less than half of that. The effective emissivity of the anode can be increased by a graphite radiator fin[93], by coatings or by fins on the metal which are more effective the smaller the thruster. Still, except for very small thrusters, the current densities on radiation-cooled anodes will be limited to the order of ≈30 A/cm^2 of inside surface, assuming 20 V anode loss.

Regenerative cooling of anodes (at specific impulse values in the MPD regime) is possible only with hydrogen or with alkali metal propellants, notably lithium. In the latter case, the ideal anode operating mode would be evaporation and ionization of the propellant on the porous or wetted anode surface, resulting in increased ion current fraction, reduced anode fall and utilization of part of the anode loss energy. Otherwise, where higher current densities are required or desirable, additional anode cooling on larger thrusters can be achieved with built-in heat pipes or with a liquid metal convective cooling loop.

MPD Propulsion Systems Choices and Problems

The choices and problems discussed here affect the performance and weight as well as the development and operating cost of the whole system. The decisions require full developmental information on all the essential systems conponents and on the mission of application, neither of which is available now. Affecting these choices are, in addition, currently unresolved essential questions, e.g. concerning tank test validity and environmental acceptability.

Propellant Choice

This has been aptly called "a long chapter on a question to which there is as yet no answer", for reasons given above. Therefore, since several recent papers discuss the choices[38,41,92] it will be treated briefly here.

The propellant choice affects the thruster, the system, the spacecraft, the atmosphere environment and the operating logistics. The thrust to power ratio seemed to favor heavy propellants like argon, but now lighter mixtures like N_2H_4 and NH_3 gain preference, which also are storable and may be environmentally and logistically (N_2H_4) the most desirable with still fair thrust to power ratios. Further thruster tests and system/mission studies are needed for final choices.

The regenerative cooling capability is minimal for most propellants in the MPD regime, except possibly for lithium and hydrogen. Lithium could remove $\Delta h_{Rc} \approx 2.5 \times 10^7$ J/kg during vaporization and heating, and hydrogen about 2.9×10^7 J/kg during heating (by $\Delta T \approx 2000$ K) alone. Assuming for example $\eta = 0.30$ and $I_s = 30$ km/s, the ratio

$$\frac{\text{Regen. cooling power}}{\text{Input power}} = \frac{2\eta \cdot \Delta h_{Rc}}{c_e^2}$$

would approach ≈2 to 3% for these two propellants. This is still very small but could suffice to cool some component of a large pulsed thruster. For continuous MPD thrusters of the 250 kW class, this regenerative cooling capability would be too small to be significant. For an arcjet in the 10 km/s I_s regime with > 60% efficiency, the above ratio becomes > 36%, which with proper design can be adequate for full regenerative cooling.

Most other propellants have small to negligible regenerative cooling capability for the MPD regime.

For storage weight and volume, the highly cryogenic, low density propellants hydrogen and helium are of course the worst. Helium has until recently not been considered at all. Hydrogen may be acceptable for missions requiring more or less continuous propellant use from the beginning, sufficient that the boiloff keeps the rest cold enough. In the longer future, hydrogen may be produced in space from water (where the oxygen is needed also) or from some other compound.

The acceptability of cryogenic propellants generally will be very dependent on missions and on future space storage facilities.

For possible propellant sharing with high thrust systems, and of course for emergency thrust with insufficient or no electric power available, hydrazene stands out. Hydrogen also can give emergency thrust with still reasonable specific impulse.

Pulsed (Quasi-Steady) vs. Continuous Thruster Systems

We will discuss this choice first for the case of self-field thrusters, since pulsed (quasi-steady) operation is now used primarily with these thrusters, both by the University of

Tokyo for the space tests and by Princeton University and JPL for ground testing. Subsequently, the differences in the test and system requirements for AAF thrusters will be mentioned, although with these there is less need for pulsed systems. It is clear that the relative advantages of the pulsed systems are disadvantages of the continuous, and vice versa. Therefore, the presumed advantages and disadvantages of the pulsed systems only will be discussed in detail, with subsequent discussion of the continuous systems as necessary.

Advantages of pulsed (quasi-steady) SF-MPD thrusters and systems are:

- They have potentially higher thruster efficiencies, especially for average power levels below 1 MW, due to lower electrode loss fractions and higher Reynolds numbers. However, some additional losses at the start and end of each pulse are to be expected.

- Higher specific impulse levels for the same propellants at "onset" are also claimed but not yet explained.

- Much easier, faster and less expensive preliminary exploratory ground testing at low average power (low duty cycle) is possible for exploring the effects of sizes, power levels, geometries, propellants etc. Some reasons for this are that heatsink models can be made from simple materials, vacuum pumps are relatively modest in size and the fiberglass vacuum tank is uncooled and nonconducting.

Note however that these advantages no longer apply for final performance, cooling and life testing at full operating temperature and average power level.

- Large area melting on cathode during warm-up ("cathode spitting") is less likely due to shortness of pulse (cf. Fig. 10).

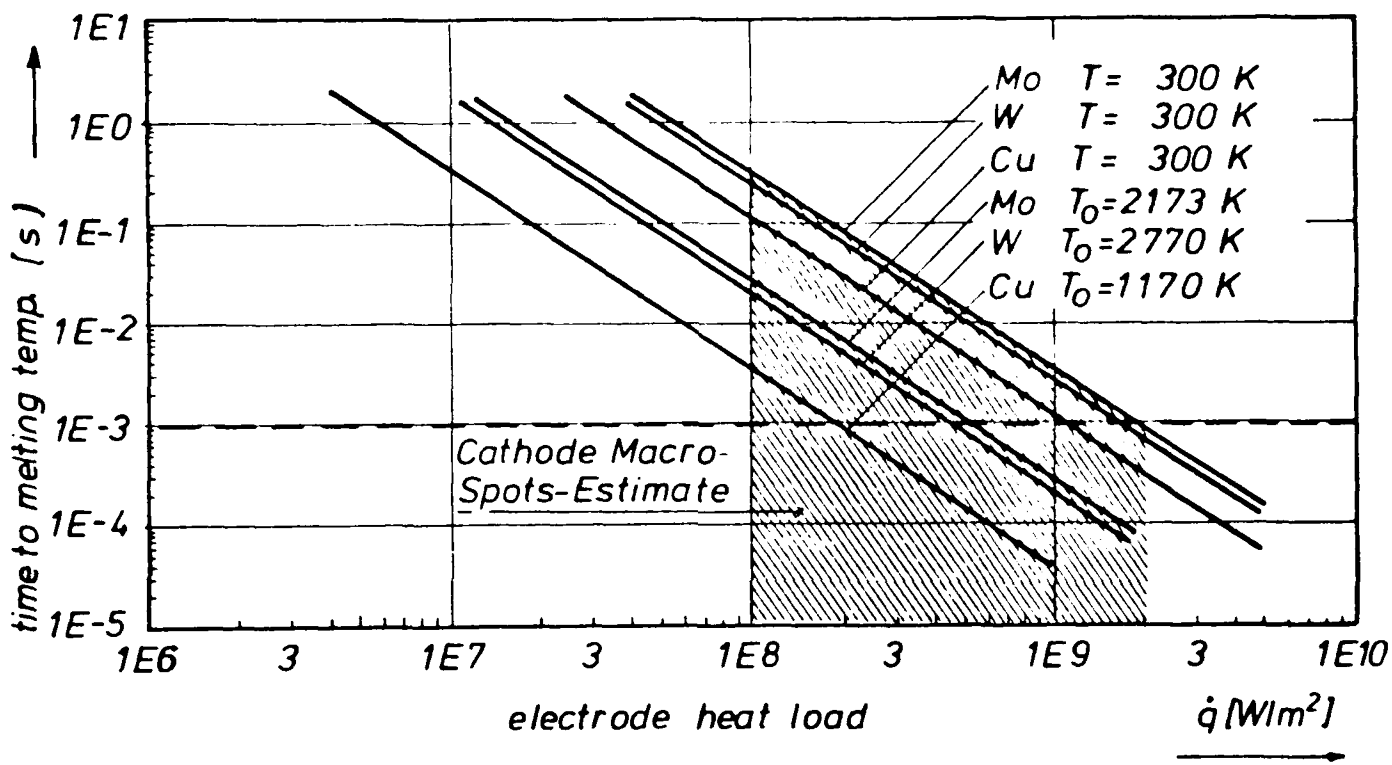


Fig. 10: Time to Reach Melting vs. Surface Heat Load (One-Dimensional, Large Area Approx.) for Different Electrode Materials and Initial Temperatures.

- Early space testing with limited power and adaptation of the system for applications with different thrust and power levels become easier through changes of duty cycle (pulse frequency). However, special provisions (heat shielding) are necessary for bringing the thruster to a representative temperature.

- The pulsed system is more suitable for attitude control of moderate-size structures.

Disadvantages of pulsed systems are:

- The power processor is much heavier and more complicated.

- With the propellant feed system, precise valve timing is difficult. The valve presents potential life/reliability problems (10^8 to 10^9 pulses are required for a typical application).

- Cathodes, after heat-up, could be more problematical due to higher current densities.

- Vibration, thermal and mechanical fatigue problems are to be expected in various system components.

- EMI effects on space craft are potentially increased, as well as communication interference.

- In a ground test, the precise determination of performance, thrust measurement, tank pressure effects, massflow and erosion is more difficult.

- Electrode effects are more difficult to study due to unknown temperature variation during the pulse. Electrode and thruster performance and life evaluation are hardly possible with low duty cycle facilities and models, due to unrealistic thruster temperatures and insufficient total operating time.

Thus, in summary, the pulsed self-field thrusters may offer performance advantages and are ideal for the preliminary evaluation of geometry, scale and propellant effects on performance. They are more difficult for precise performance testing, offer no substantial advantage in final development and life testing, and have definite systems liabilities.

By contrast, the continuous SF thrusters require more expensive refractory or, for research, liquid-cooled models, hence more time for exploratory work and very large vacuum facilities. They permit simple, precise experimental determination of all performance parameters and life testing over representative operating times. For early space testing, they will require scaling down or the use of APU's or large rechargable battery banks.

The applied-field (AAF) thrusters require in addition still better vacuum (10^{-3} mbar or below) and relatively larger vacuum tanks to prevent wall interference. For pulsed testing of AF thrusters, all metal tanks may present a slight distortion of the fields, unless the applied-field is operated continuously. On the other hand, AAF thrusters can be explored more fruitfully in smaller sizes and have less need to be pulsed.

The final choice between pulsed and continuous systems must, in the writer's opinion, await both further performance tests with SF and AAF thrusters and system application studies giving the weight and cost differences between the two systems and the dry weight vs. performance trading factors. Quite possibly, thrust and total impulse regimes (with corresponding applications) will emerge, for which pulsed systems may be preferable, while for much larger power levels the continuous systems may predominate. The many years of satisfactory service of the pulsed ablation thrusters indicate their usefulness for certain applications. These thrusters, however, avoid the difficult propellant valve problem of the pulsed (quasi-steady) MPD thrusters using gaseous propellants.

Ground Testing Problems

For the thruster sizes of probable interest here, that is, with thrust of several Newton and power above 100 kW, the huge vacuum pump capacity required is the dominating problem of ground testing, both for continuous thrusters and for pulsed ones at the full duty cycle. The vacuum required for reliable performance testing is at least 10^{-3} mbar, according to some sources (Cann[39]), and ideally perhaps as low as 10^{-4} mbar, at least for Hall-ion thrusters. For self-field thrusters, 10^{-2} mbar may be sufficient, according to our own experience.

Table 3 shows some typical numbers of the pump capacities needed for a 4.45 N (1 lbf) thruster, with two typical gaseous propellants, argon and hydrogen. Roughly, this would amount to 10 to 100 one-meter diameter diffusion pumps with corresponding LN traps, valves and ducting, much larger than most laboratories have at their disposal, except NASA Lewis RC.

Table 3: Facility Requirements for 4.45 N (1 lbf) Thruster Test

Thruster exit velocity c_e (km/s) or I_s (kNs/kg)	10	20	30	40	50
Electric input power for 30% efficiency (kW)	74	148	223	297	371
Propellant mass flow (kg/s x 10^4)	4.45	2.22	1.48	1.11	0.89
Pump inlet volume flow at 0.1 Pa (10^{-3} mbar) 288 K for <u>argon</u>, (m^3/s)	263	132	87.8	65.8	52.7
ditto, for hydrogen (m^3/s)	5252	2636	1753	1314	1052

Possible ways of dealing with this problem are:

1. pulsed (quasi-steady) testing at low duty cycle

2. working with small (1/100th to 1/10th scale) thruster models, especially with AAF thrusters

3. a combination of pulsed, small scale and full power tests, the latter with inadequate vacuum, to include all effects

4. use of condensible propellants and cryo-pumping.

As condensible propellants, the alcali metals (here Na, K or Li) are ideal, since they even remove other gases present by gettering. For argon, xenon, neon and nitrogen, cryopumping is possible also at considerable expense; but helium, hydrogen and therefore also ammonia, hydrazene and similar hydrogen-based propellants would be excluded. For the long range, a safe method of chemically binding the hydrogen should be developed, since N_2H_4 or NH_3 may still be the preferred propellants.

For the near term it appears that the alternative (3), the use of a combination of test procedures, will be the best approach, at least until it is more clear than now what propellants, thruster types and sizes will be candidates for full development. After some space testing experience and comparisons, it may be possible to relax the vacuum requirements for many tests, particularly the expensive electrode life tests.

Other ground testing problems arise in connection with the test tanks. For pulsed thruster testing at low duty cycles and for very small continuous thrusters, non-magnetic, nonconducting (e.g. fiberglass vacuum tanks) are ideal.

For the higher average power levels which are of interest here, the tank walls have to be cooled and should be non-magnetic (stainless steel or aluminum), especially for the AAF thrusters. But with metal walls, some current flow through the tank walls can occur, affecting their test results. Representative average power levels (duty cycles) and longer run duration are necessary to get the thruster to equilibrium temperature, both for performance and electrode life reasons and to accumulate expectancy. This means that for the AAF thrusters the metal tanks have to be very large if wall interference due to discharge attachments are to be avoided.

Space Testing Problems

Early space testing of one or several thruster types will be needed as a final proof of performance to select one type for development. However, in the next ca. 5 to 7 years, the continuous power levels available for such tests are likely to be on the order of 25 kW or less, that is, lower than the power levels of the thrusters to be developed for the large structures of the 90's and beyond.

Pulsed quasi-steady testing in space with the available power, that is, with a low duty cycle, is the obvious first choice, which was made by the University of Tokyo for their Shuttle-based tests. The disadvantage is that a thruster designed for higher average power levels will not reach its full operating temperature and therefore will give only limited information on its full power performance and electrode life. There may be ways to circumvent this by artificially reducing the radiation cooling of the thruster (e.g. with radiation shielding). Full power testing for a few minutes (sufficient to reach equilibrium thruster temperature) using an APU or a bank of rechargable batteries may be considered with the Shuttle.

Electromagnetic interference (EMI), especially with pulsed operation and with some propellants, and spacecraft (solar-cell) contamination problems will of course have to be dealt with.

Other Thruster Design Problems

Some of the other thruster design problems include:

1. seals and allowance for differential expansion

2. insulators, including their cooling

3. general materials compatibility of the components with the propellant and with each other

4. physical integrity under repeated heating and cooling, thermal shock resistance, creep

5. possibly high temperature ($\approx$2000 K) heat pipes, if needed, e.g. for cathodes

For the insulators, there are very few choices considering physical stability, compatibility with the refractory metals at temperatures above 2000 K, and electrical resistivity. Since the insulators are difficult to cool, they should be protected from direct contact with the plasma.

Some experience can be drawn from nuclear, notably incore and out-of-core thermionic and high temperature gas-cooled reactor projects where similar materials and design problems arise.

Key Problem Areas to be Resolved for Successful MPD Propulsion Development

The Plasma Acceleration Efficiency Problem

It was mentioned that electrostatic ion acceleration is basically most efficient, potentially 100% for the acceleration process alone, since no energy is expended until the particle moves, and then only charge times potential drop. This is somewhat equivalent to the idealized piston engine (or Catapult piston) which is ideal for accelerating something from rest.

By contrast, a single Faraday J x B channel has a relatively poor efficiency accelerating a gas from rest to the final (E/B) velocity, similar to a DC shunt motor or a turbine, which are poor for acceleration from rest. Demetriades and Ziemer in Ref. 94 quote 40% efficiency limit for a channel. The DC motor can be improved by applying a variable voltage during acceleration (always close to the "back-emf"), and the channel can be similarly improved by subdividing into segments with different E/B values. The Hall-accelerator or Hall channel should be one step closer to the ion engines if one could succeed in achieving really high values of the effective Hall parameter ($\omega_e\tau_e$), i.e. if one could really suppress the electron current. Sutton and Sherman[95] quote 60% for $\omega_e\tau_e = 4$. The theoretical limiting efficiencies of accelerating a plasma from rest with a moving (expanding) magnetic field (like the pulsed inductive thrusters or travelling wave channels) should basically be higher than that of the single Faraday channel.

The self-field thruster is basically a Faraday channel with a magnetic field which drops to zero at the last current line and has variable cross sectional area, particularly for the external accelerating region. King[34] calculates streamtube efficiencies up to 65% (at 1500 sec) with a one-dimensional ideal gas model (constant cross section, constant conductivity) for equilibrium argon, and can show efficiency improvements with a variable cross-section geometry. Hall effect or electrode losses are not included, and the upper efficiency limit would lie in the "onset" region, hence would not be experimentally realizable.

These simple model calculations generally postulate constant transport properties and neglect electrode and frozen flow losses (as well as limits imposed by Hall parameter, anode starvation, etc.), so that the performance attainable in practice must be much lower.

To this writer, it appears very important that the basic possibilities for improving plasma acceleration efficiencies be investigated. Specifically, investigations should be conducted to find out what would theoretically be possible, e.g. with staged acceleration (several different voltages) and/or devices designed specifically for staged thermal followed by electromagnetic acceleration; this to find the ideal fluid MGD acceleration efficiency limits and the corresponding geometric field and discharge configurations, even if these are not realizable in practice.

Next, the three-dimensional and Hall effects must be investigated, as well as discharge limitations and instabilities (see below) which may prevent the reaching of high performance regimes. Concerning these latter problem areas, the open questions posed in the following paragraphs need to be answered.

Midstream Flow-Discharge and Acceleration Processes

Is a uniformly distributed (rotationally symmetric) discharge possible-- provided the electrodes permit this? Under what conditions does the arc in midstream contract itself into spokes or filaments (e.g. due to excess mass flow, insufficient mass flow or only due to electrode spots and jets)? (This can probably not be experimentally investigated for the midstream alone, since contraction may simultaneously occur at the anode at least.)

Is there an inherent maximum current to mass flow ratio or $(J^2/\dot{m})_{max}$ for the midstream (a) for hydrogen and (b) for multiply ionizable substances for each level of ionization?

Under what conditions is a flow-discharge stable (a) as a filament and (b) as a distributed discharge, with and without external magnetic field? What effect does instability and resulting turbulence have on the transport properties in case (b)? Specifically, what is the reason for the generally observed unexpectedly low Hall currents-- filaments or turbulence?

What are the possible and, under each set of conditions, prevalent mechanisms of plasma acceleration? In particular, is the acceleration of neutrals, i.e. the use of a partially ionized plasma, desirable (from an efficiency viewpoint) and under what conditions is acceleration of neutrals (e.g. through electron-neutrals collisions) appreciable?

What are the limiting relationships between the applied voltage and the attainable exit velocity, if any?

The midstream acceleration and arc stability problems will have to be attacked primarily theoretically, probably still with simplified models as in the past, since complete numerical solutions of the conservation equations for the MPD flow-discharge appear not yet in sight. Key experiments to cross-check the results from the analytical models should then be carried out. Some of these experiments can be planned immediately, including the critical experiments with hydrogen suggested by Cann to test the minimum voltage theorem. The observation of the appearance and movement of current "spokes" and possibly resulting current distribution measurements could also begin immediately for all types of thrusters, as a continuation of such work previously done with AAF thrusters. The detection of spokes is easier with AAF thrusters-- by means of magnetic pick-ups-- since in the AAF thrusters the spokes rotate with clearly defined frequencies. This must be done early to determine whether and in which cases it is still meaningful to work with flow discharge models assuming rotationally symmetric (circumferentially uniform) current distributions.

Once suitable models have been formulated and flow cases calculated, measurements of critical variables to check the accuracy of these model calculations must be carried out. One example would be the continuation of the spectroscopic (Doppler shift) jet velocity measurements carried out earlier at the DFVLR Stuttgart and elsewhere. Particular emphasis should be placed on velocity as well as current distributions (including ringcurrents) along the outer discharge and jet of AAF MPD thrusters, to settle the unanswered questions of Hall currents or magnetic nozzle effect along the plume. The quantitative degree and effects of flow ingestion of tank gas must be evaluated, perhaps with tracer substances.

In formulating and planning such an extensive experimental research project, the costs and expected final results must be weighed against the same for earlier space tests which, while not clarifying all the detailed mechanisms of the tank effects on the thrust, will give the net effect on thrust I_s and efficiency by comparison with tank tests in various facilities.

In any case, however, the earlier experiments to determine the effect of the tank pressure on the measured thruster performance will have to be extended to all thruster types.

Electrode Phenomena

On hot (refractory metal) cathodes, is an ostensibly spot-free emission possible, and up to what current densities?

Under what conditions (quantitatively) does the arc contract itself to macro-spots, i.e. what determines the transition from distributed micro-spots (or diffuse emission) to macro-spots or the reverse process, e.g. during warm-up?

How can destructive macro-spots be avoided or minimized during warm-up? What are the current densities and local heat loads of macro-spots, as a function of propellant density and type and cathode material and temperature? Does pulsed (quasi-steady) operation have a basic advantage during cathode warm-up?

What are the quantitative erosion rates due to sputtering and microspots on hot refractory metal cathodes as a function of propellant type and density and current density? This is the basic cathode life limit question.

What are the cathode heat loads as a function of cathode and propellant materials and operating conditions?

On hot refractory metal anodes, does a truly continuous arc attachment (without micro-spots) exist for a technically smooth surface? What are the fall potentials and the total heat loads with ostensibly smooth attachment as a function of propellant type and anode materials and the operating conditions? What are the resulting current density limits for radiation-cooled anodes? Can these values be influenced by the anode construction and propellant transpiration?

Under what conditions (propellant to current density ratio) does transition to macro-spots inevitably occur on anodes, assuming adequate cooling?

The electrode studies will have to be predominantly empirical, supported by calculations with theoretical models. The models consist of the arc portion near the spot, which is examined for its stability and the heat penetration into the electrode, giving the size of the molten pool and the local vapor production.

These spot calculations, as well as the corresponding experimental measurements, must be carried out for materials, temperatures and ambient gas densities applicable to MPD thrusters, i.e. refractory metals both cold (for start-up) and near their allowable operating temperatures, at various applicable ambient gas densities and compositions. Currently, a multitude of papers are published and congresses held on cold copper electrodes under vacuum arc condition. For the MPD electrode studies it may be necessary to plan additional measurements of basic materials and properties which are not readily available, e.g. of thoriated tungsten and other refractories, at high temperatures, including the effective work functions.

An important unknown currently is the diffusion or other migration of the thoria through current-carrying hot tungsten cathodes and the thoria deprivation near the surface after extended operating periods. The quantitative effect of the thoria and the necessary concentrations for this effect (i.e. the reduction of the effective work function) are required for theoretical calculations and meaningful interpretation of experiments.

A new anode theory has to be developed, which describes the anode phenomena adequately over the whole operating regime and allows minimization of the anode losses.

Complete Thrusters

1. Do any applied-field configurations have a basic advantage in the potential efficiency over steady self-field thrusters, considering both midstream and electrode losses?

2. Do the minimum propellant density limits at the electrodes together with midstream limits impose basic limits in $J^2/\dot{m}$ or $JB/\dot{m}$ respectively, and thus on the attainable specific impulse values?

3. Are the Cann minimum voltage criterion and the thereby implied massflow, I_s, and efficiency limits unconditionally valid or can these limits be exceeded or violated and, if so, under what conditions and which of the underlying assumptions is not applicable?

4. Does pulsed (quasi-steady) thruster operation have any basic advantages for anode and cathode cooling and with respect to erosion during warm-up, apart from the known efficiency advantage due to high power operation?

5. Based on (a) electrode limitations and (b) plasma acceleration mechanisms, what are the optimal geometries and propellant injection points for each thruster type (SF and various AAF types?) For instance, for which cases should the anodes be in the rear and have most of the ion generation take place there?

6. What propellants appear optimal for each thruster type and I_s range, independent of systems considerations but including the thrust to power ratio influence in transport costs?

7. Are there anticipated size/power limits or optimal sizes for purely radiation-cooled MPD thrusters on the basis of efficiency, cooling and weight considerations?

8. What are the best design configurations for resolving the coil or permanent magnet cooling problem in AAF-MPD and Hall-ion thrusters?

9. What are compatible material combinations for insulators, electrodes, seals and other components for temperatures around or above 2000 K and various promising propellants?

System Considerations

1. Optimal specific impulse values for typical model missions and MPD-typical efficiency vs. I_s curves, as well as other parameters (engine, power and propellant system weights, etc.) are necessary. Also requisite are acceptable minimum performance/maximum system weight and cost combinations which will still make MPD propulsion preferable over ion or chemical systems.

2. Rough trading factors between system weights and costs and different efficiency vs. I_s curves are needed.

3. The weight, cost and reliability penalties of pulsed systems, of cryogenic propellants or other features which would improve performance or thruster life must be estimated.

4. The number of cold starts and warm-up cycles and the propellant use distributions for typical missions are needed for thruster and system life and propellant storage consideration (if cryogenic).

5. The potentially acceptable propellants from the viewpoints of spacecraft contamination, environmental impact, logistics and availability must be selected to minimize the necessary thruster performance and life testing and system calculations and design studies.

Summary and Conclusions

The history of earlier research and technology development work on plasma propulsion was compared with the parallel larger effort on ion propulsion. Apart from the extreme theoretical and experimental difficulties inherent in all arc-like discharges and plasma dynamics, it appears that the advance of plasma propulsion was held back by:

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UNCLASSIFIED IRA-85-P6 AFRPL-TR-86-013 F/G 21/3 NL

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

- an initial lack of comprehension of these difficulties

- the large variety of possible thruster types and propellants, leading to diffusion of effort

- the lack of a clear-cut development goal in terms of missions or available power supplies

- the rather limited funding considering the basic difficulties

While at least the difficulties are much better understood today, future research and technology efforts in preparation for a development should benefit from the past experience. Needed are a realistic appraisal of the difficulty and the time and resources required for success, and careful planning and coordination of all efforts by someone with superior overall understanding of the whole problem complex.

The most likely MPD thruster types for possible future development were compared, together with general performance trends, as well as the present uncertainties and lack of understanding of performance values and limits. No clear-cut performance leader or other sufficiently certain criterion for selecting any one thruster type at present was found. The Cann-Bennet minimum voltage theorem was briefly reviewed. The highest specific impulse values at "onset", achieved with pulsed and continuous self-field thrusters and different propellants, appeared (again) to agree in their trend remarkably well with the limits (i.e. the Alfvén velocities at different ionization levels) predicted by the theory.

Various system considerations for choosing between pulsed and continuously operated thrusters, as well as various thruster types and propellants, were reviewed. Pulsed SF thrusters show definite advantages for preliminary development (geometry, size and propellant selection), for performance at lower average power levels, in the required areas for radiation cooling and possibly in the electrode damage during warm-up. These somewhat predictable advantages must be weighed against as yet not fully known life (fatigue) and system liabilities-- energy storage and power processing, propellant valve, increased EMI-- and current performance uncertainties.

The costs and other difficulties of reliable ground testing for the final development of large MPD thrusters were reviewed and related to the propellant selection as an additional criterion besides the thruster performance effect cited earlier. A preliminary suggestion for holding down ground testing expenses was made together with a plea for early space tests to remove ground test uncertainties, especially for the applied-field thrusters, prior to a final selection of the thruster type to be fully developed.

Electrode design, erosion and cooling was reviewed, with the conclusion that despite the vast accumulated empirical and theoretical detail knowledge, the basic understanding of certain essential stability problems (the spot formation) was still insufficient for thruster design-- analogous to the spoke formation and instabilities problem of the midstream discharge. For large, simple, radiation-cooled thrusters (without convective or heat-pipe heat transport) the radiation cooling was found to impose very definite average current density limits. For tungsten cathodes, a maximum of about 100 A/cm^2 is allowed, a factor 10 or more less than the T-F emission limit (for thoriated tungsten), thus allowing this factor-of-ten increase in the current density for pulsed operation. On molybdenum anodes, $\approx$5-7 A/cm^2 inside surface is believed to be a limit for radiation cooling, much lower than values used on liquid-cooled copper anodes for research thrusters. Pulsed thrusters have somewhat more margin for cooling, since (for the same average thrust) their average currents are lower.

Present-day vacuum arc experiments and theory predict excessively high cathode erosion rates at microspots far beyond previous AAF-MPD and arcjet life test experience. Cathode erosion due to microspots and sputtering and its accurate determination are considered major development problems for MPD, especially self-field thrusters. With continuous thrusters at least, this is aggravated by macrospot erosion during warm-up.

In the last chapter, the presently unresolved questions are reviewed, the answers to which appear necessary for a rational, economical and successful MPD propulsion system development. Some brief suggestions are given as to how these could be attacked in the author's opinion.

Overall, plasma propulsion still appears to be an approach worth pursuing for large space structures, though the current reliable information on its potential performance, life, system weight and cost appears insufficient to evaluate its competitive position relative to chemical, ion and other possible systems. With the recent advance in ion engine performance in the 20 to 35 kNs/kg I_s range (or in thrust-to-power ratio) the major advantages of plasma propulsion will have to come from system weight and cost, compactness and reliability.

Prior to deciding on a development program or selecting one thruster type for development, a major coordinated technology program appears appropriate to determine reliably the minimum performance attainable with each thruster type and one or two "acceptable" propellants, the probable electrode life potential achievable with development, representative thruster and system weights and costs.

Simultaneously, system application studies should determine parametrically system performance requirements, weights and costs which would make plasma propulsion performance and cost competitive, including the various trading factors.

In parallel, a concerted effort to resolve as far as possible the basic problems of MPD arc discharges and plasma acceleration, including electrode phenomena, should be made, utilizing the advances of plasma dynamic theory and the advanced computer codes developed in recent years. While the technology and preliminary development programs should not wait for the resolution of the very difficult plasma dynamic stability problems, any enhanced understanding gained could materially assist the rational thruster development process.

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