National Aeronautics and
Space Administration
Jet Propulsion Laboratory
NASA Management Office
FOIA Public Liaison Office
4800 Oak Grove Dr., M/S 180-200K
Pasadena, CA 91109-8001

Reply to Attn of: RA000/NMO

March 5, 2019

Mr. John Greenewald
john@greenewald.com
27305 W. Live Oak Rd., Ste. 1203
Castaic, CA 91384-4520

Re: FOIA Request 19-JPL-F-00180

Dear Mr. Greenewald:

Thank you for your Freedom of Information Act (FOIA) dated and received in our office on December 13, 2018. Your request was assigned Case File Number 19-JPL-F-00180. Due to lapse of federal funding, NASA was closed from December 26, 2018 through January 28, 2019, which caused the delay in processing your FOIA request. Your request was for:

> **Advanced Propulsion Options For The Mars Cargo Mission Document ADB186535 9/1/1989**

We found a record responsive to your request. Portions of this record were withheld pursuant to FOIA Exemption 5 U.S.C. § 552 (b)(3), relying on 22 U.S.C. § 2751, et seq., the Arms Export Control Act, which protects certain export-controlled information.

24 pages are being released in full (RIF);[1]
19 pages are being released in part (RIP);
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[1] All page counts are provided in approximate numbers.

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Freedom of Information Act Unit
NASA Headquarters
Attn: Nikki Gramian
Principal Agency FOIA Officer
300 E Street, S.W., 5P32
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National Archives and Records Administration
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College Park, Maryland 20740-6001
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You requested to be categorized as a *"representative of the news media"* but you did not provide sufficient information for placement in this requester category in accordance with NASA FOIA Regulations 14 CFR § 1206.507. Requester category shall be considered on a case-by-case basis using the criteria in this section. Therefore, we placed your request in the "*all others*" category for fee purposes. In this category, requesters are afforded two hours of search and 100 pages of copy/scan without charge. Fees for processing this request are less than $50.00 and are not being charged in accordance with 14 CFR §1206.504(f).

In accordance with § 1206.804 (c), after consultation with the NASA Management Office General Counsel Office, I am the official responsible for the initial determination of your request. If I can be of further assistance, please feel free to contact me in writing to this center at the address shown on the letterhead. You may also e-mail correspondence to jpl-foia@nasa.gov or reach me by telephone at 818-393-6779 and fax at 818-393-3160. Also, you may contact Ms. Nikki Gramian, Principal Agency FOIA Officer and Chief Public Liaison at the information provided above.

Thank you very much.

Sincerely,

Dennis B. Mahon
Freedom of Information Act
Public Liaison Officer

Enclosure:
JPL D-6620 Advanced Propulsion for the Mars Cargo Mission

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JPL D-6620

Advanced Propulsion Options for the Mars Cargo Mission

R. H. Frisbee
M. G. Sargent
J. J. Blandino
J. C. Sercel
N. Gowda

September 1989

National Aeronautics and
Space Administration

JPL

Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California

: 158

ABSTRACT

This report summarizes the results of an evaluation of a variety of advanced low-thrust propulsion options for the cargo-delivery portion of a split-mission piloted Mars exploration scenario. The propulsion options considered were solar sails, 100-MW_e class nuclear electric propulsion (NEP), 100-MW_e class solar electric propulsion (SEP), magnetic sails (magsails), mass drivers, rail guns, solar thermal rockets, beamed-energy (laser and microwave) propulsion systems, and tethers. The requirement was to transport 400 metric tons (MT) of cargo from a 500-km altitude low Earth orbit (LEO) to a 6000-km altitude Mars orbit (e.g., Phobos' orbit) for the 2014 opportunity. The primary figures of merit used in this study were total initial mass in low Earth orbit (IMLEO) and the Earth-to-Mars trip time.

The baseline propulsion system, against which the advanced propulsion concepts were compared, was an aerobraked chemical (O_2/H_2) propulsion system with a specific impulse (I_{sp}) of 470 lbf-s/lb_m. This system had an initial total mass in LEO of 1640 MT (including payload) and had an Earth-to-Mars trip time of 294 days. It was found that solar sails can provide the greatest mass savings over the baseline chemical system. However, solar sails suffer from having very long trip times. A good performance compromise between a low IMLEO and a short trip time can be obtained by using 100-MW_e class NEP systems; they can even be lighter and faster overall than the baseline chemical system. Such systems may be particularly suited to the piloted portion of the mission, where a premium is placed on trip time. A 100-MW_e SEP system is a close competitor to the NEP system, providing almost as good a performance, but without the technological, operational, or "political" constraints of space nuclear power.

Magsail, mass driver, beamed-energy, and tether concepts were found to have moderate benefits in mass or trip time, but their performance is contingent on several factors which could reduce their effectiveness. For example, the magsail concept, like the solar sail, has infinite specific impulse. However, magsails can only operate far from a planet; this imposes a large infrastructure overhead since a fleet of orbit transfer vehicles (OTV) are required to transport the magsails and their payloads from LEO to the magsail operational orbit. Mass drivers have a low I_{sp} for the Mars cargo mission but they do have a high efficiency (electric-to-jet power). They also can make use of any material as propellant. Thus, if copious amounts of "free" lunar O_2 propellant were available, a mass driver operating at modest power levels (10 MW_e or less) could show a mass savings over the baseline system, and do so for trip times on the order of 500 days. However, this is contingent on the availability of "free" lunar propellant; without this "free" propellant, the mass driver is not competitive. Beamed-energy concepts were found to provide some benefits in mass when used as OTVs to deploy the payload (with a chemical O_2/H_2 stage for Earth escape and aerocapture and Mars) at GEO altitudes. A laser-augmented SEP vehicle used for the round trip to Mars also provides significant trip time savings over an un-augmented SEP system, since the laser provides a rapid Earth escape/capture. However, all the beamed-energy concepts suffer from the limited range over which power can be beamed (e.g., microwaves to GEO or near-visible light to the Moon). Even the laser-augmented SEP system, which reverts to a normal solar powered SEP far from the Earth, requires very high-powered lasers (10-MW beam or more) to provide any significant trip time savings. Also, the space-based infrastructure (laser/microwave power stations, orbital relay mirrors) required to support beamed-energy transmission would need to be "amortized" over many users. Lastly, tether systems show only a small advantage in IMLEO over the baseline system. This is due primarily to the need to break up the 400 MT payload into twenty 20-MT segments, each with its own chemical O_2/H_2 stage for tether-assisted Earth escape and Mars capture. Also, there is a significant LEO, Deimos, and Phobos tether station set-up mass investment which must be "amortized"

over many missions. However, tethers may have greater benefits for the piloted portion of the mission. For example, tethers can be used to lower (de-orbit) landers and raise ascent vehicles. Also, a tether station on Deimos can provide a vehicle returning to Earth with Mars' escape velocity, thereby greatly reducing the trans-Earth injection propulsion requirements.

Two concepts were found to have very poor performance for the Mars cargo mission scenario assumed in this study. These were solar thermal propulsion and rail guns. Solar thermal propulsion suffers from having too low an I_{sp} (1200 lb_f-s/lb_m) for this mission. Rail guns suffer from both a low I_{sp} and a low efficiency (electric-to-jet power); they require high powers (50 MW_e) for optimum performance and can only show a mass savings over the baseline chemical system if copious amounts of "free" lunar oxygen are available as propellant in LEO.

Based on the results of this study, solar sails, 100-MW_e class NEP systems, and 100-MW_e class SEP systems should be considered in detail for application to the Mars cargo mission. Further, 100-MW_e class NEP and SEP systems should be evaluated in detail for the piloted portion of future Mars missions since they have the potential for significant savings in both IMLEO and trip time as compared to the baseline chemical systems. Similarly, tethers should be evaluated for the piloted portion of the Mars mission since they may provide major savings in mass for the Mars-to-Earth portion of the trip. Magsails, mass drivers, and beamed-energy concepts should also be considered for the Mars cargo mission, although their performance will depend on a number of factors (e.g., "amortization" of a space-based laser for laser propulsion vehicles).

Finally, it should be noted that the conclusions reached in this study are highly mission-scenario dependent. Thus, a concept that has no benefit for the Mars cargo mission scenario assumed in this study may show significant benefits for the piloted mission. Similarly, concepts that are not attractive for Mars missions may provide major benefits when used for cis-lunar missions (e.g., LEO-to-GEO OTVs or lunar base missions). Also, different thrusting or trajectory strategies (e.g., low-thrust spiral planetary escape or capture, as used in this study, versus multiple-impulse medium-thrust trajectories) may have a significant impact on performance. Furthermore, in this study, the concepts were used in a "pure" Mars cargo mission mode with a minimum of mixing of modes. For example, only the beamed-energy concepts were used in a LEO-to-GEO OTV mode due to the limitations in transmission distances. Future studies should consider the option of "mixed" mission modes of operation; such as, for example, the use of an advanced concept for a LEO-to-GEO OTV-type transfer followed by trans-Mars injection by a second system. This may be a particularly attractive approach, since a number of previous studies have shown that systems with I_{sp}s of 1000 to 1500 lb_f-s/lb_m (e.g., mass drivers, rail guns, solar thermal propulsion, laser/microwave thermal propulsion) can provide major savings in IMLEO as compared to chemical systems, and savings in trip time as compared to high-I_{sp} electric propulsion systems at comparable power levels. Finally, the same advanced propulsion concepts considered in this study for the Mars cargo mission should also be evaluated for the lunar base cargo mission, again with IMLEO and trip time as the primary figures of merit.

ACKNOWLEDGEMENTS

This work described in this report was performed by the Jet Propulsion Laboratory, California Institute of Technology, under contract to the NASA Lewis Research Center (LeRC), Contract NAS7-918, Task Order RE-65, Amendment No. 608. The NASA LeRC technical manager was Mr. Jim Gilland.

The authors wish to express their thanks to Carl G. Sauer Jr. (trajectory analyses), Bryan A. Palaszewski (propellant tankage and thermal analyses), Jack A. Jones (refrigeration systems), Robert W. Staehle (solar sails), Paul M. Stella (100-MW_e class solar photovoltaic arrays), Stanley Krauthamer (high-power electric propulsion power processing units), Dr. John L. Callas (magsail analyses), and Dr. Paul A. Penzo (tethers) for their technical assistance.

CONTENTS

SECTION 1

INTRODUCTION

There are a wide variety of advanced propulsion concepts which hold the potential for significantly reducing the initial mass in low Earth orbit (IMLEO) or reducing the trip time required for missions to support future NASA piloted missions to Mars. The overall objective of this study was to evaluate the benefits (in terms of reduced IMLEO and trip time) of the use of several advanced low-thrust propulsion concepts for the cargo mission portion of a split piloted Mars mission in the year 2014. The concepts evaluated in this study include those that derive their power from sunlight or laser light, as well as those that use electric power from a nuclear reactor or solar photovoltaic cells.

1.1 CONCEPTS EVALUATED

Concepts and mission scenarios evaluated in this study are summarized in Figures 1-1 and 1-2. Those concepts which use sunlight directly include the solar sail, which uses momentum exchange from solar photons to "push" a gossamer sail, and the solar thermal rocket, which focuses sunlight into a thrust chamber to heat a propellant working fluid like hydrogen, which is then expelled through a conventional nozzle. A concept related to the solar sail is the magnetic sail (mag sail), which uses a magnetic interaction with the charged particles in the solar wind to "push" the "sail" (actually a superconducting solenoid magnet ring).

Two concepts which directly use beamed energy (e.g., laser light) from a remote beam source are the laser thermal rocket and the microwave thermal rocket. The laser thermal rocket is similar to the solar thermal rocket except that near-visible laser light from a remote laser transmitter (ground or space-based) is used instead of sunlight. Two types of microwave thermal rocket concepts are possible. The first is the analog of the laser thermal rocket in that microwave radiation is absorbed by the propellant and used to heat the propellant. By contrast, the electron-cyclotron resonance (ECR) microwave thruster concept uses a microwave beam to directly excite a propellant and expel it; the propellant is in fact not just heated thermally but rather is excited electromagnetically by coupling to the energy in the microwave beam. The ECR thruster concept is the one selected in this study for use with the microwave "thermal" propulsion system

The laser or microwave radiation can also be used indirectly to power an electric thruster (e.g., ion thruster) by first converting the incoming photons to electricity by either "solar" photovoltaic cells (near-visible wavelength) or by a rectenna (microwave wavelength); these concepts represent electric propulsion vehicles with a potentially light-weight power supply (receiver) on the vehicle because the actual power supply (transmitter) is remotely located on the ground or in low Earth orbit (LEO).

A second general category of concepts are those which use a nuclear or solar electric power supply to operate electric propulsion thrusters. These include 100-MW class Nuclear Electric Propulsion (NEP) and Solar Electric Propulsion (SEP), as well as megawatt-class rail guns and mass drivers. In the rail gun and mass driver, the propellant is in the form of "pellets" which are accelerated electromagnetically in a "bucket" and shot out from the vehicle to provide thrust. Rail guns and mass drivers can use any material as the "pellet" mass and thus could use extraterrestrial materials as a propellant source, thus reducing the required IMLEO.

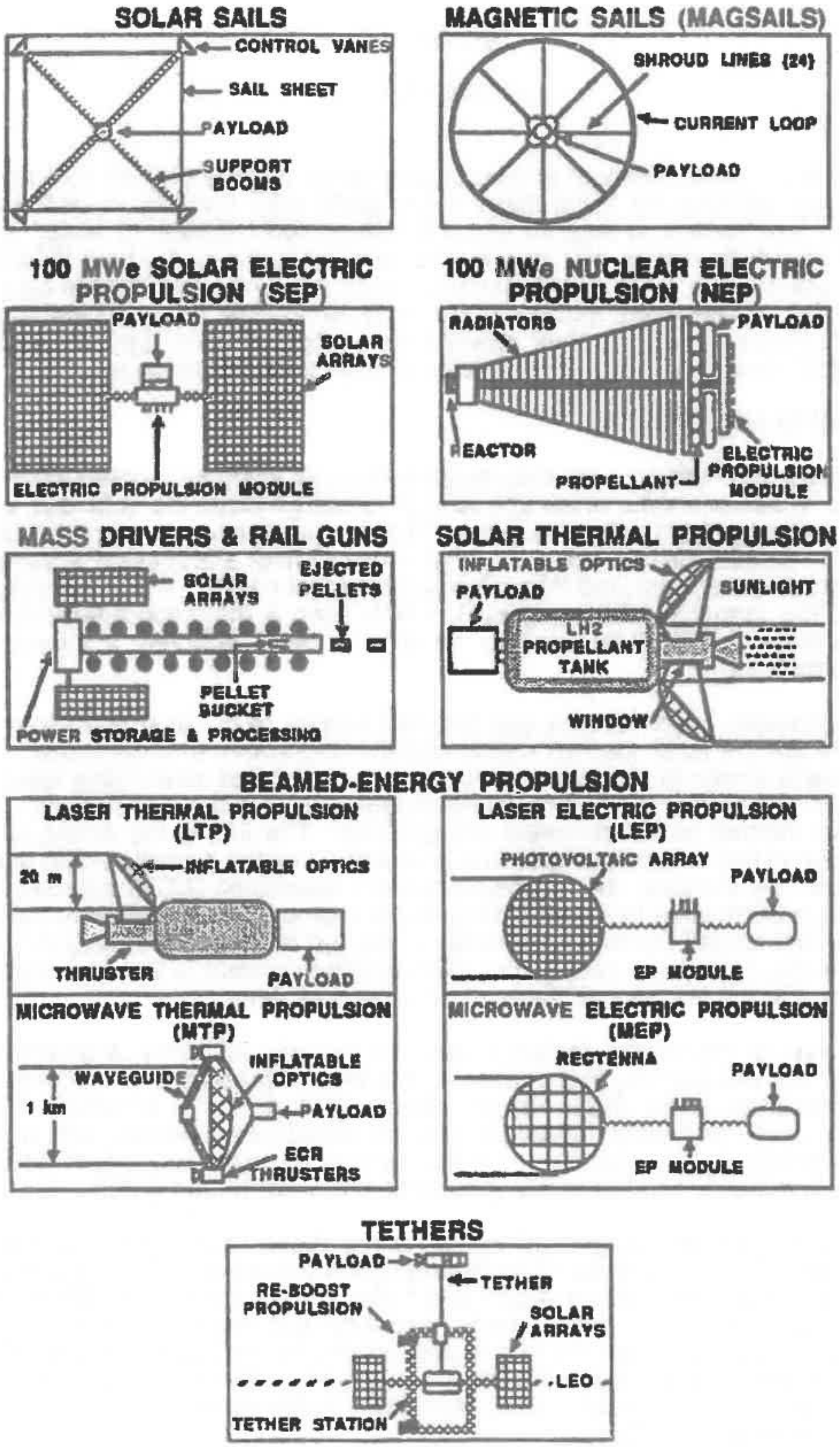

Figure 1-1. Advanced Propulsion Concepts Evaluated in This Study

CONCEPT	MIN/MAX ALTITUDE	TRANSFER TO MIN/MAX ALT FROM LEO	TRANSFER TO MARS & MARS ORBIT INSERTION	RETURN TO EARTH
• SOLAR SAIL • ADV. SAIL	2000 km (DRAG-FREE)	• CHEM • SEP	• SOLAR SAIL • ADV. SAIL	• SOLAR SAIL • ADV. SAIL
• MAGSAIL	1 AU (SOLAR WIND)	• SEP	• MAGSAIL	• MAGSAIL
• 100 MW SEP • SOLAR THERM • MASS DRIVER & RAIL GUN	(LEO) (LEO) (LEO)	(N/A)	• 100 MW SEP • STP • MD & RG	• 100 MW SEP • NONE • MD & RG
• 100 MW NEP	1000 km (NSO)	• SEP • 100 MW NEP (1st TIME)	• 100 MW NEP	• 100 MW NEP
• LASER THERM • μ-WAVE THERM • μ-WAVE ELECT • LASER ELECT	LEO>GEO LEO>GEO>LEO LEO>GEO>LEO LEO>GEO>LEO	• LTP • μTP • μEP • LEP	CHEM (O2/H2) AEROCAPTURE	NONE
• LASER ELECT	(LEO)	• LEP	SEP AFTER EARTH ESCAPE (LEP USED AS SEP)	SEP TO EARTH CAPTURE, THEN LEP TO LEO

Figure 1-2. Mission Scenarios

A final, non-propulsive concept is the use of tethers for orbit raising and lowering in Earth and Mars orbit, respectively. The use of tethers can significantly reduce the requirement of the spacecraft by using long cables to reel the spacecraft in or out of the deep gravity well of a planet and thus raise or lower orbits.

1.2 TRADE STUDIES

As mentioned above, the primary figures-of-merit used in evaluating concepts for this study were the initial mass in LEO and trip time required for the Mars cargo mission. The primary focus is on total system mass, including the empty or "dry" vehicle weights, propellant, and payload (400 MT total to Mars/Phobos orbit). Also included in the total mass is the weight of any supporting infrastructure. This infrastructure can take many forms, depending on the concept and mission scenario. For example, several of the concepts cannot operate directly from LEO, but instead have some minimum altitude at which they must operate. Thus, an added fleet of orbit transfer vehicles (OTVs) is required to boost the system from LEO to the minimum operating altitude; the dry weight and propellant required for the OTV fleet is included in the infrastructure mass requirement.

For trip times, the primary figure-of-merit is the Earth-to-Mars trip time, since the cargo mission is a one-way delivery. In most cases, however, the vehicles are re-usable, so a Mars-to-Earth trip time is also found. The round-trip time is important if the vehicles are to be phased properly with subsequent launch opportunities. For example, a system with a round-trip time of less than the Earth-Mars synodic period (2.2 years) could be used for the next launch opportunity; longer round-trip times would require skipping one or more opportunities, thus requiring a larger overall cargo vehicle fleet for continuous operations.

However, in this study, it is assumed that the full system must be deployed the "first" time, so all associated masses are included and only the Earth-to-Mars delivery time is considered in detail. Re-use and "amortization" of vehicles for multiple cargo delivery cycles should be considered in detail in future studies to identify benefits and penalties associated with re-use of vehicles for a continuous Mars base operation and growth.

[redacted] (b)(3)

1.4 ASSUMPTIONS COMMON TO ALL CONCEPTS

Several ground rules and assumptions were established which were common for all of the concepts. The first was that the time frame of the mission be the year 2014. The primary requirement is to transport 400 metric tons (MT) of cargo from a 500-km, 28.5° low Earth orbit (LEO). This initial starting node was chosen as typical of a space station orbit. All calculations of IMLEO use this LEO node as a reference point. The payload is delivered via a slow minimum-energy conjunction-class trajectory to a 6000-km Mars orbit. This orbit is taken as the delivery node; it is at the same altitude as Phobos, although the need to actually rendezvous and land on Phobos was not considered in detail.

Several of the concepts described below are large in size; it was assumed that it would be neither practical nor desirable to have these vehicles dock directly with a space station or base in LEO or Mars orbit. Instead, a separate chemical stage was added to the payload to provide a small Delta-V capability (50 m/s) for any required rendezvous and docking of payloads in Earth or Mars orbit. For this purpose, the Orbital Maneuvering Vehicle (OMV) was used. This vehicle has a "dry" weight (M_{Dry}) of 4035 kg and a useable propellant (M_p) capacity of 4286 kg with an I_{sp} of 300 lb_f-s/lbm.[1] The OMV can provide a 50-m/s Delta-V for payloads weighing up to 100 MT; for payloads in excess of 100 MT, a "stretched" OMV was used with the following scaling equation:

$$M_{Dry}\ OMV = 3136.1 + 0.20972 \cdot M_p \quad \text{[all masses in kg]}$$

Also, the OMV has a 463 W electric power system composed of solar cells and batteries (for shadow periods). Even though sunlight intensity at Mars is less than half that at Earth, the amount of time spent in sunlight and shadow in a 6000-km altitude Mars orbit is such that the OMV power system can provide about 66 % of its rated power at Mars.

In addition, structural or docking adapters were added to the payloads, thus increasing the "effective" payload weight. This is illustrated in Fig. 1-3 for the case of the OMV. Note that some of the concepts and mission scenarios require aerobraking of the payload into Mars orbit; this is performed by an O_2/H_2 stage with an I_{sp} of 470 lb_f-s/lbm and an aerobrake mass corresponding to 15 % of the vehicle (stage, propellant, and payload) mass at the start of the aeromaneuver.

Another study ground-rule was that the total 400 MT payload could be split into smaller units, such that the smallest unit was 20 MT. Thus, it is possible to see the effect on IMLEO and trip time by increasing the number of vehicles, but decreasing the payload per vehicle (and thus mass per vehicle), e.g. one vehicle (with a 400 MT payload), two vehicles flying in parallel (each with 200 MT payload), and so on to 20 vehicles (each with a 20 MT payload).

A final study assumption was that only one "new" or advanced concept be used at a time. For example, an aerobraked O_2/H_2 vehicle was used with the tether concept; a 100-kW class solar electric propulsion vehicle was used as the OTV for those concepts that cannot leave directly from LEO (e.g., solar sails). In the context assumed in this study, aerobraked chemical or 100-kW class SEP vehicles are considered to represent the baseline (non-advanced) propulsion technology available in the year 2014 time frame assumed for this study. Similarly, in the laser propulsion concepts, the beam power is limited to ranges of 1 to 10 MW since this would require electric power supplies for the lasers of 10 to 100 MW (electric) assuming a 10% electric-to-laser efficiency; in this case, beam powers in excess of 10 MW would require 100-MW class electric power supplies which would be considered a second "new" technology in addition to the laser. One area that should be considered in future studies are synergistic combinations of advanced propulsion concepts (e.g., tethers and high powered SEP vehicles).

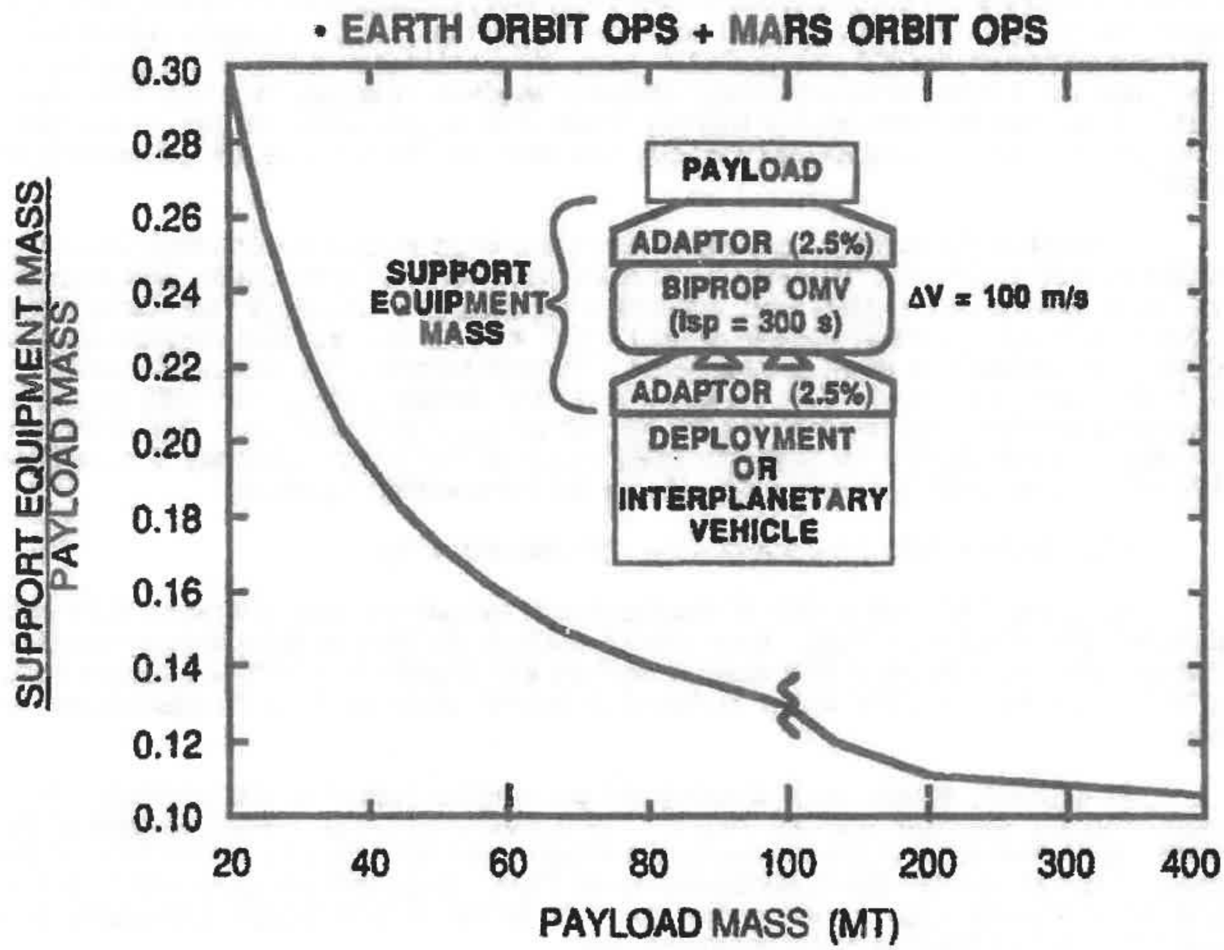

Figure 1-3. Effective Payload Weight Due to Adding an Orbital Maneuvering Vehicle (OMV) for Earth and Mars Orbital Operations

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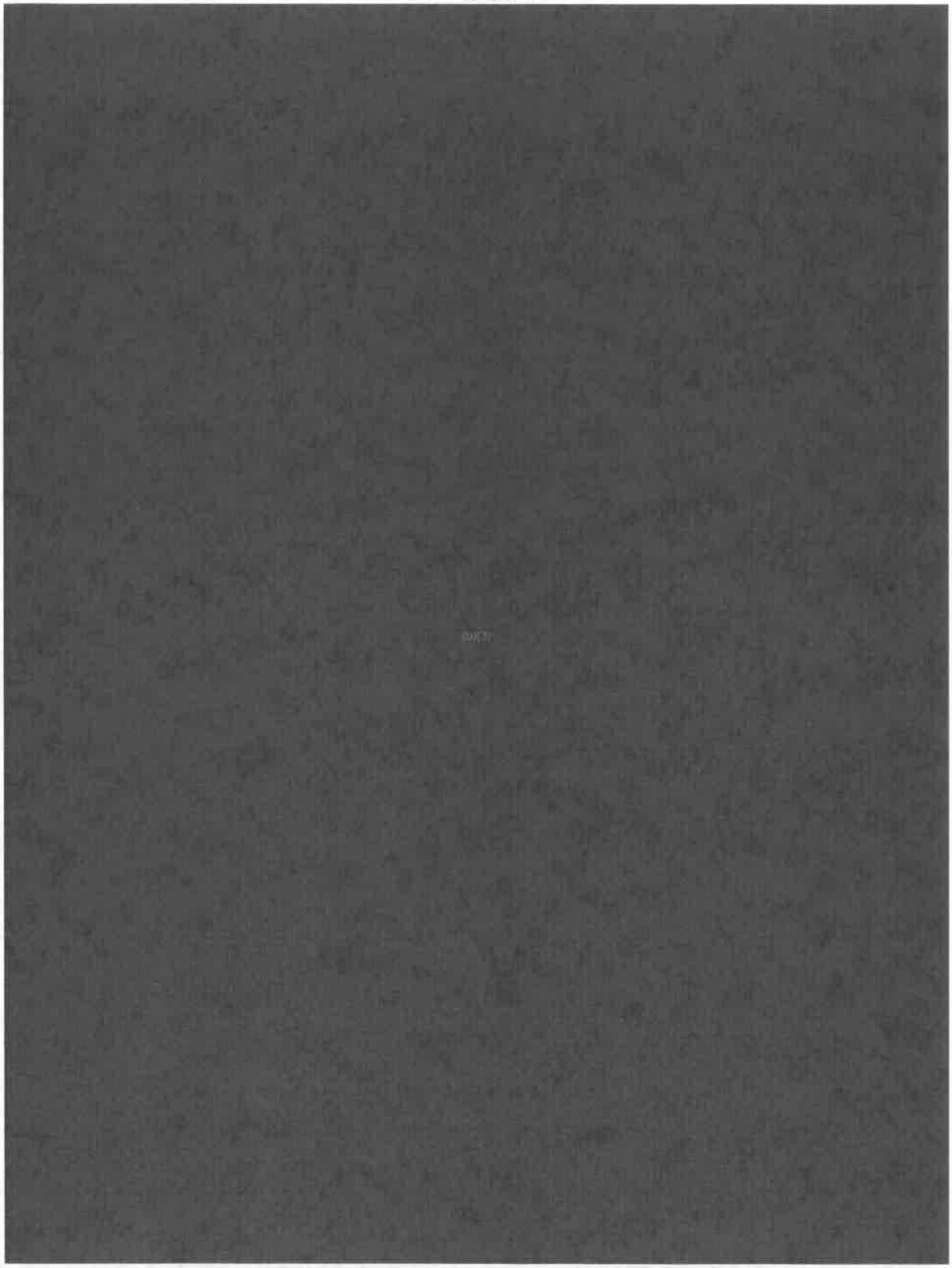

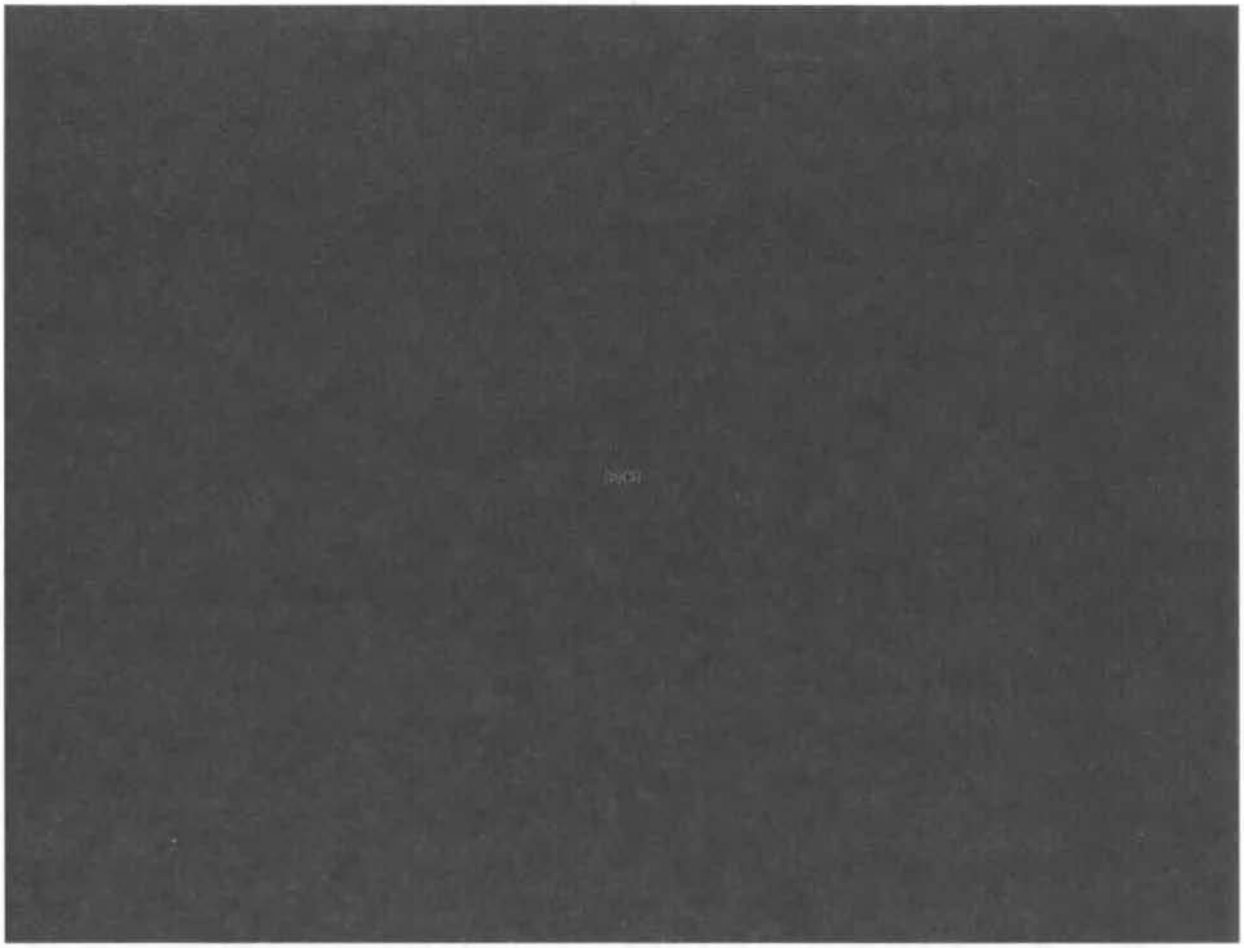

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SECTION 2

SOLAR SAILS

2.1 INTRODUCTION

Solar sails operate by using momentum exchange with solar photons; this amounts to a force of 9 Newtons/km^2 at 1 AU. As such, a solar sail has "infinite" specific impulse, because it requires no propellant, but it has a low acceleration resulting in long trip times. Also, solar sails are typically large, gossamer structures with dimensions of kilometers; for example, a typical solar sail has an area of 4 km^2.

Solar sails were first extensively studied in the late 1970's for the Halley Comet rendezvous mission.[1] At that time, there was an extensive analyses made of solar sail fabrication techniques (thin silvered sheets and light-weight booms), control and dynamics, and trajectory analysis. The study found that solar sails were eminently feasible from a technology and mission performance point of view, but the development risk was considered too high for the short time available before launch. Instead, Solar Electric Propulsion (SEP) was considered less risky given the mission's schedule constraints.

Although the Halley Comet mission was not pursued by the United States, interest in solar sails for a variety of lunar and Mars cargo missions, as well as planetary mission, has continued because solar sails represent the most fuel efficient possible inter-orbital "supertanker" in space. Solar sails have been extensively studied in the past for Mars cargo missions; much of the discussions below are derived from these studies.[2,3]

Figure 2-1 illustrates two solar sail concepts. The first is the classic square sail consisting of a thin (few mills) sheet of silvered or aluminized plastic stretched over a supporting light-weight boom. Small "fly swatter" vanes are located at the corners of the sail; they have a combined area of 0.5% of the total sail area and are rotated to produce differential light pressure for use in maneuvering the sail.[3] The sail can also be maneuvered by shifting the payload so that the center of mass is offset from the center of (light) pressure. The second type of sail illustrated in Fig. 2-1 is the heliogyro solar sail. In this concept, the sail is spun like a helicopter blade; the sail material is unrolled and stabilized by centrifugal force. Maneuvering is accomplished by changing the "pitch" of the blades. The heliogyro sail is easier to deploy than the square sail; has a greater stability from random disturbances (due to its rotational inertia), but has a slower maneuvering rate due to the rotational inertia.[1] Thus, the two types of sails have different strengths and weaknesses, although the square sail, with its faster maneuvering (turning) response, might be favored for missions involving extensive planetary escape and capture spiral orbits (because the sail has to re-orient itself relative to the sun on each orbit).

Currently, there is no NASA-funded work on solar sails, although a private organization, the World Space Foundation, has built a prototype sail (880 m^2 area) as a demonstration of the required on-orbit deployment and maneuvering capability. The group is awaiting a launch vehicle to place the sail in a high-altitude orbit, because a sail cannot operate below an altitude of about 2000 km due to air drag would exceeding photon pressure at a lower altitude.[4]

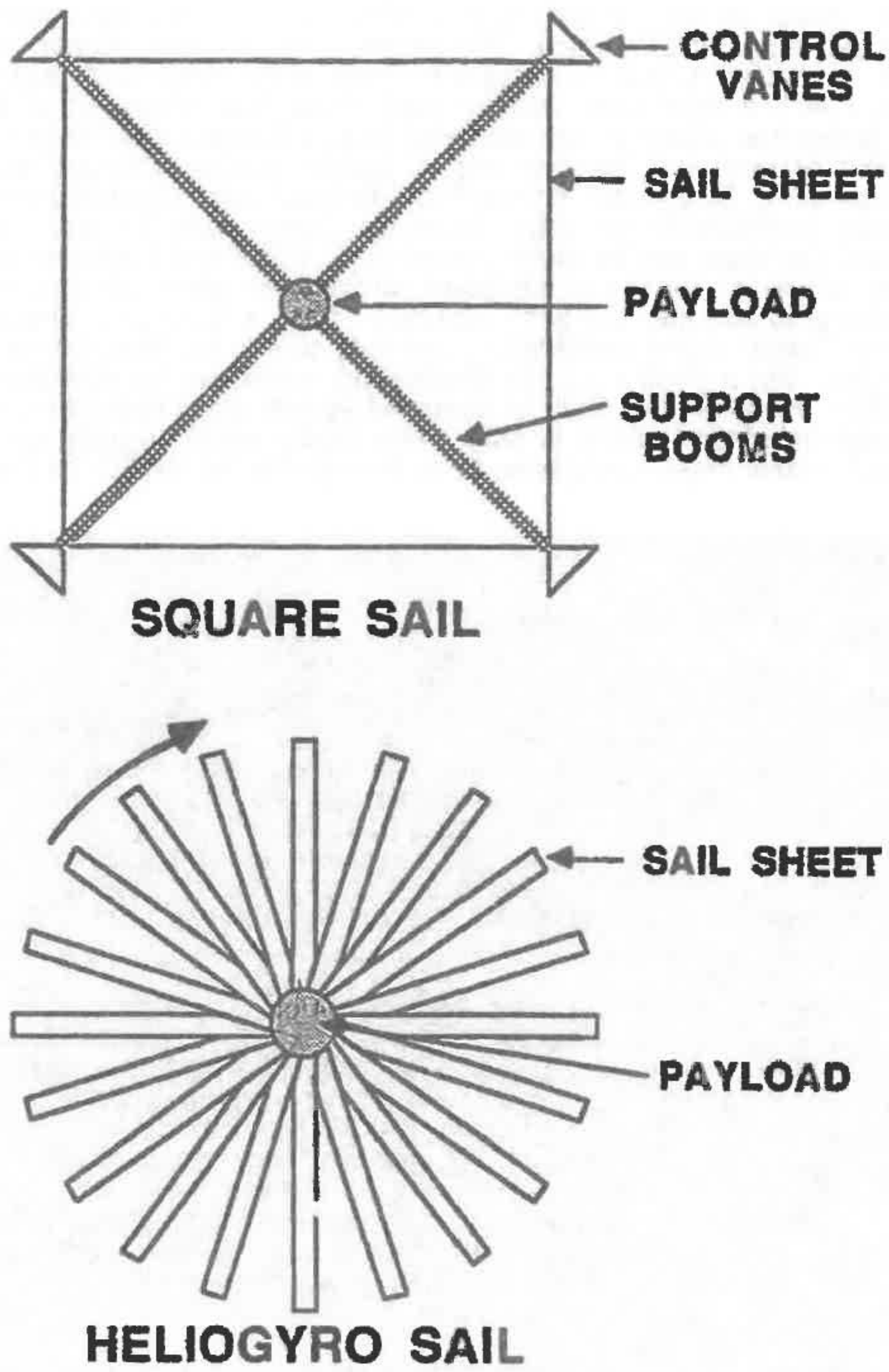

Figure 2-1. Solar Sail Concepts

In this study, no distinction is made between square and heliogyro sails. Instead, the primary performance parameter is the areal density (grams/m^2) of the sail. This parameter is an important measure of sail performance, because it determines the acceleration of the sail (i.e., solar pressure [N/km^2] divided by areal density [g/m^2] gives acceleration). Areal density, in turn, is determined by both the thickness of the sail sheeting and the supporting structure. For example, in the Halley Comet mission and more recently in studies by Staehle on sails for Mars cargo missions,[2] deployable sails were assumed with a total areal density (sail sheet plus structure) of 5 g/m^2. A deployable sail requires relatively thick sail sheet (e.g., 2.5-micron thick Kapton) to survive folding (on the ground) and packing into a launch vehicle, followed by unfolding (deployment) on orbit. By contrast, Garvey[3] and Drexler[5] have considered sails erected or constructed (fabricated) on orbit; because these sails do not need to be folded/unfolded, the sheet can be much thinner (e.g., 0.015 to 0.2-microns thick). This results in sails which are erected or fabricated on-orbit with areal densities ranging from 1.0 g/m^2 (Garvey) to less than 0.3 g/m^2 (Drexler). Thus, a Garvey- or Drexler-type sail could have significantly higher acceleration, and thus shorter trip time, than a deployable Staehle-type sail. For a given area, the Staehle sail would also be significantly heavier (greater IMLEO). However, this must be balanced against the infrastructure requirement of a sail erection/fabrication facility in orbit. This facility would basically be a separate space station,[3] whose mass would have to be included in the IMLEO for the advanced sails.

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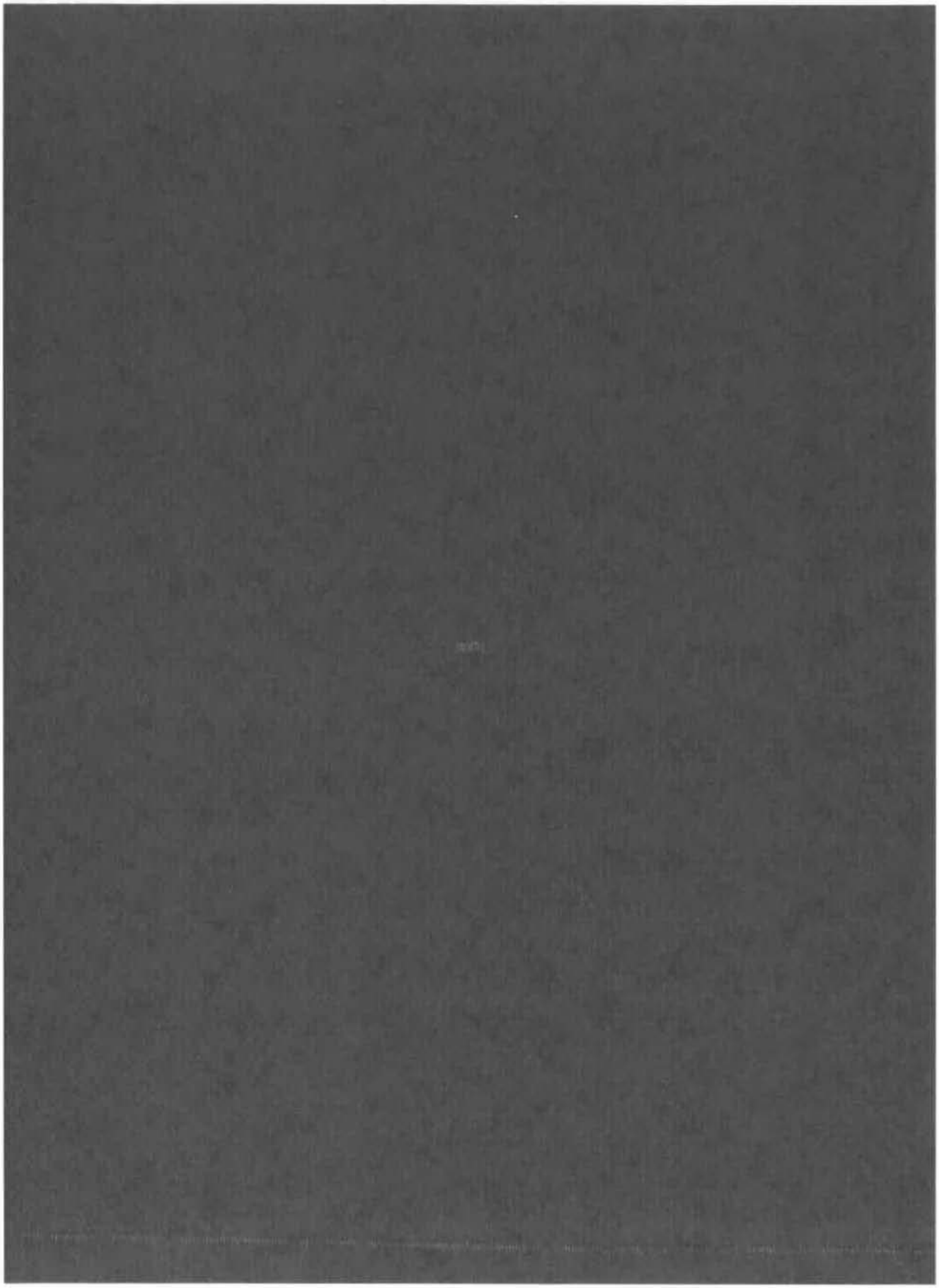

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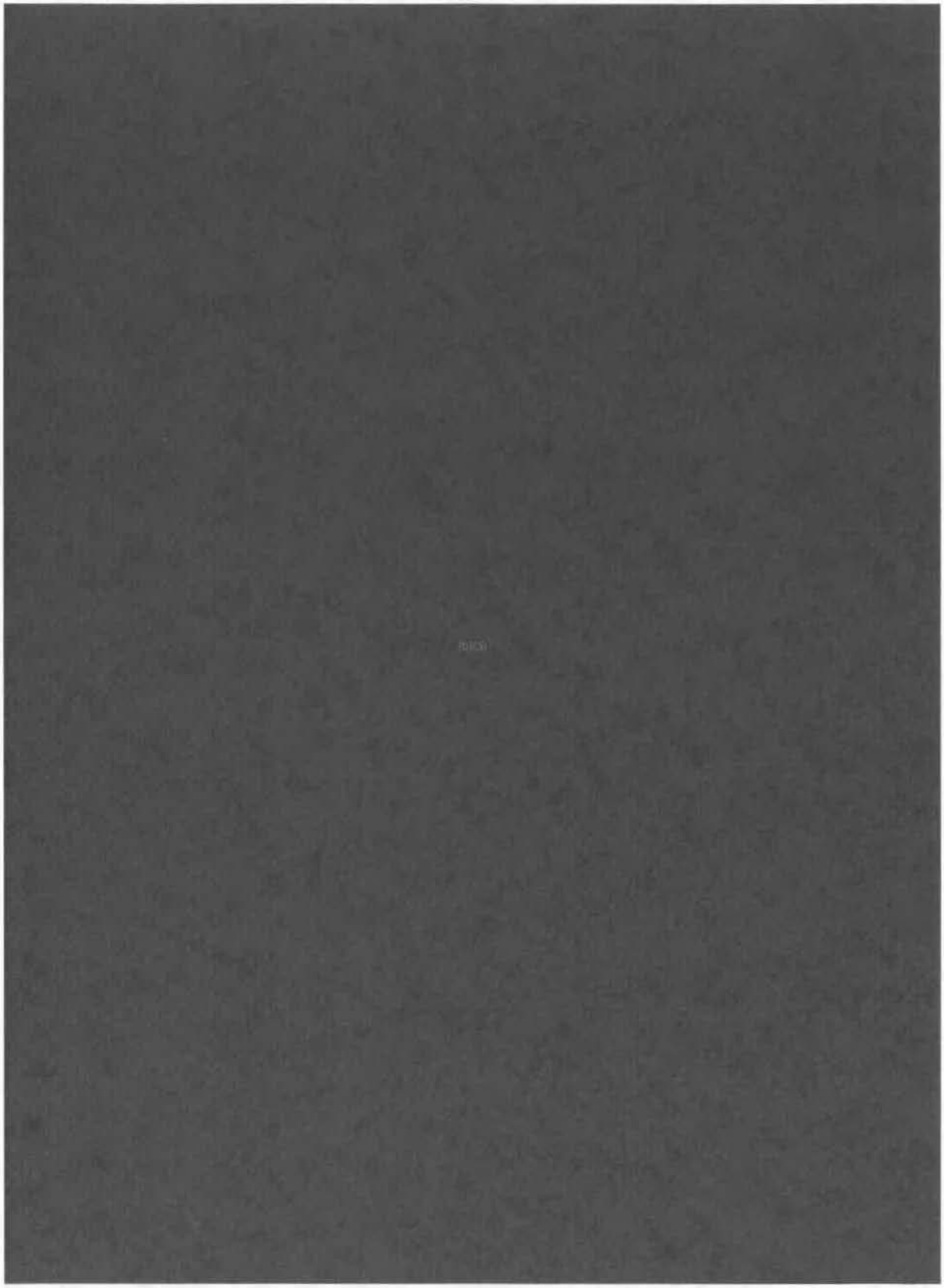

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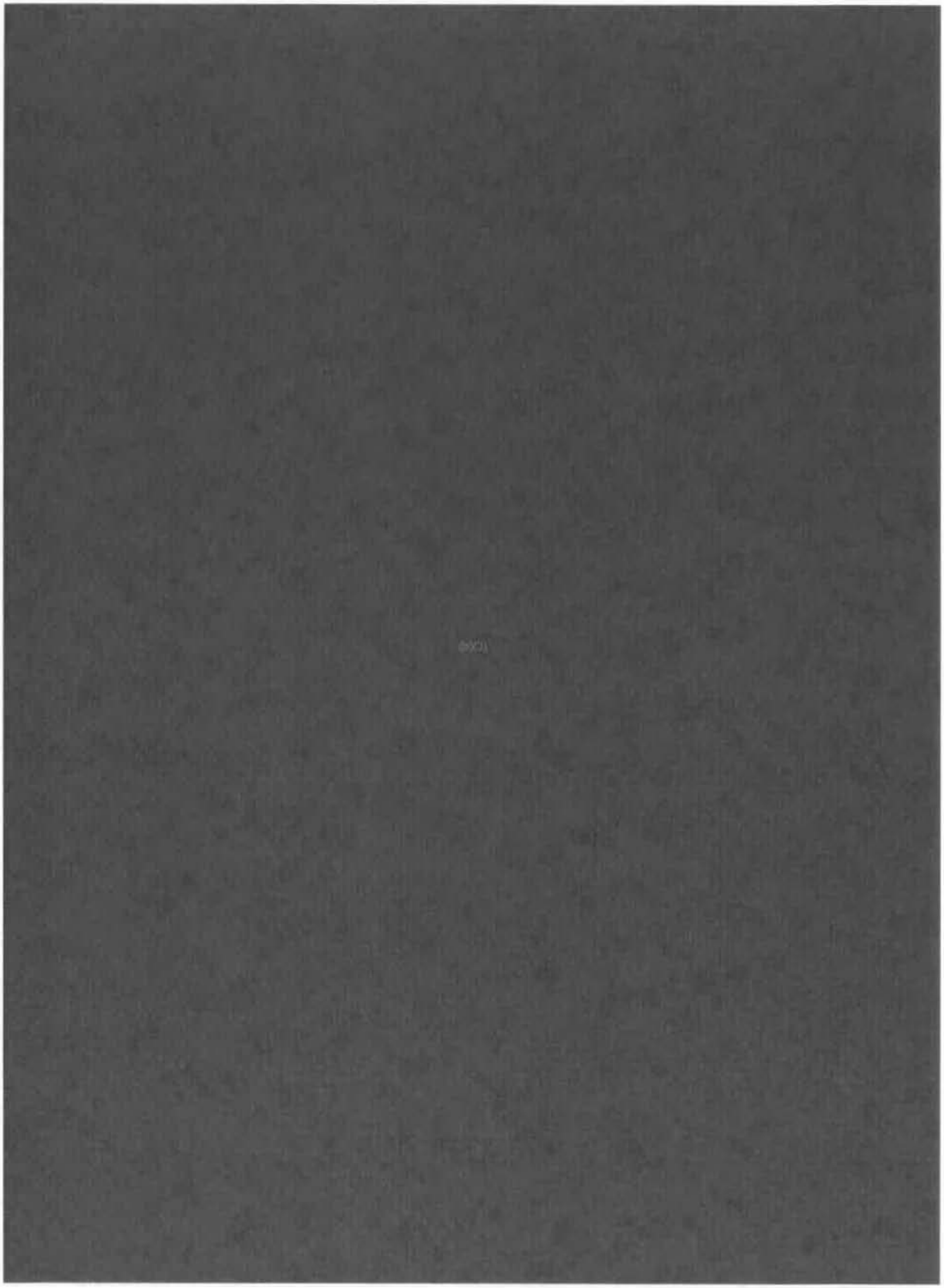

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SECTION 3

100-MW CLASS SOLAR ELECTRIC PROPULSION

3.1 INTRODUCTION

A Solar Electric Propulsion (SEP) system, as shown in Fig. 3-1, consists of a solar photovoltaic power supply, a power processor unit (PPU) which converts the solar array power output to the form required by the thrusters, and the electric thrusters. In this study, a 100-MW class SEP system was analyzed. A similar-sized Nuclear Electric Propulsion (NEP) system is described in the next Section.

Previous studies[1] have shown significant benefits for the Mars cargo mission utilizing NEP systems with a total (power and propulsion) specific mass of 10 kg/kW, an I_{sp} of 5000 lb_f-s/lb_m, and a power level of 1 to 10 MW electric (4 MW typical). This SEP study (and the NEP study described in the next Section) was aimed at investigating ultra-high power SEP (and NEP).

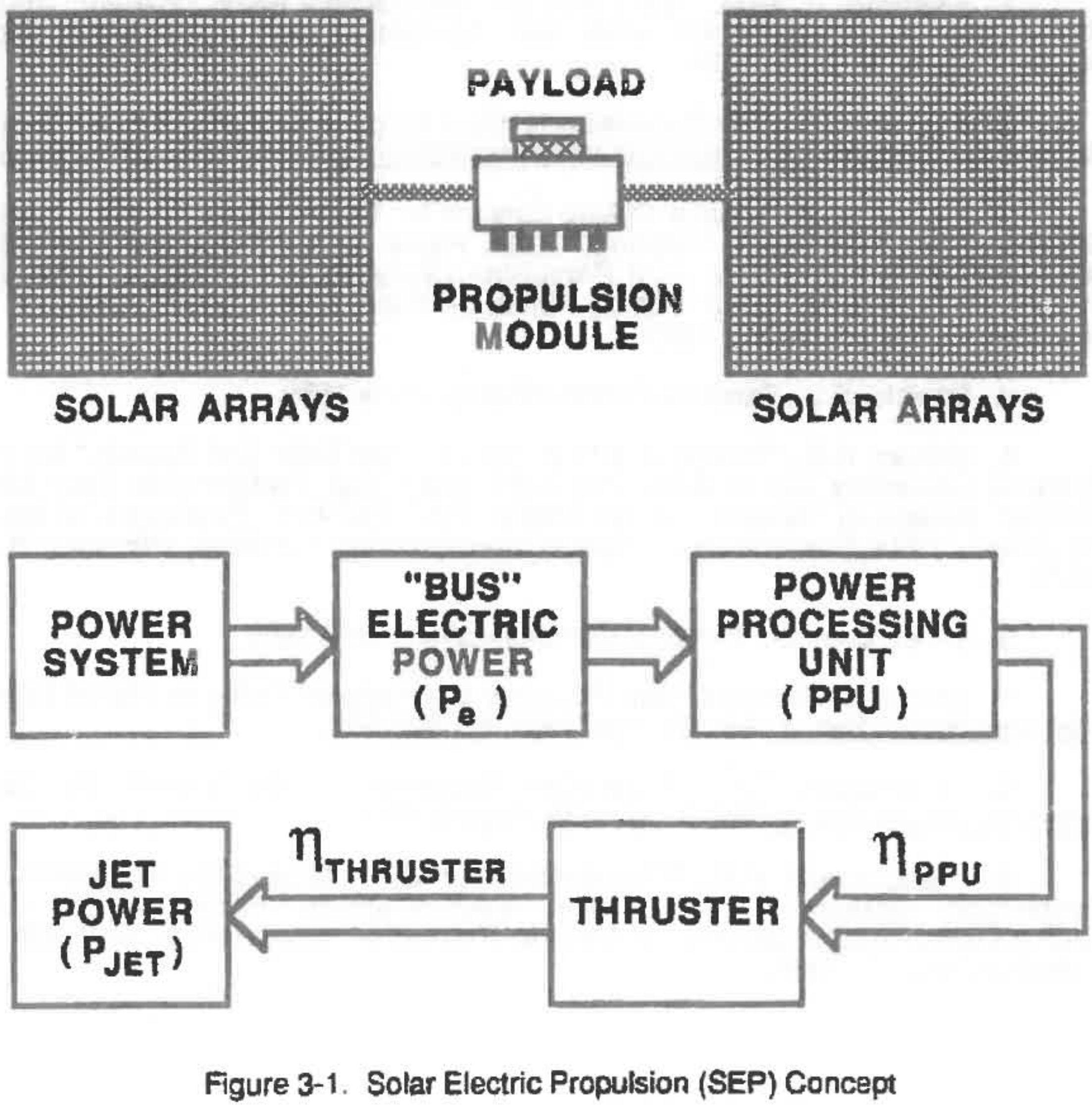

Figure 3-1. Solar Electric Propulsion (SEP) Concept

Note that in discussing SEP (and NEP) concepts, it is the "bus" electric power (P_e) that is quoted; this is the (average) power output from the solar arrays (or nuclear reactor). As shown in Fig. 3-1, the "bus" electric power is then fed to the power processor unit (PPU) and from there to the thruster. There are losses and inefficiencies in the PPU and thrusters, such that the propulsion or jet power (P_{Jet}) is typically 50 to 90 % of the input "bus" electric power.

From an operational point of view, a SEP vehicle has an advantage over a NEP vehicle in that the SEP vehicle can operate from LEO; by contrast, a NEP has a minimum operational altitude of about 1000 km to ensure that no radioactive components enter the Earth's biosphere in case of catastrophic failure of the NEP vehicle. However, the SEP vehicle suffers from shadowing in Earth or Mars orbit, resulting in a longer trip time than a NEP vehicle which has a continuous power source. Similarly, power output from the solar array drops off as the vehicle moves away from the sun. However, the efficiency (sunlight-to-electricity) of the solar array increases with decreasing temperature. Thus, the power output from a solar array drops off more slowly than a $1/R^2$ distance from the sun, as shown in Fig. 3-2.

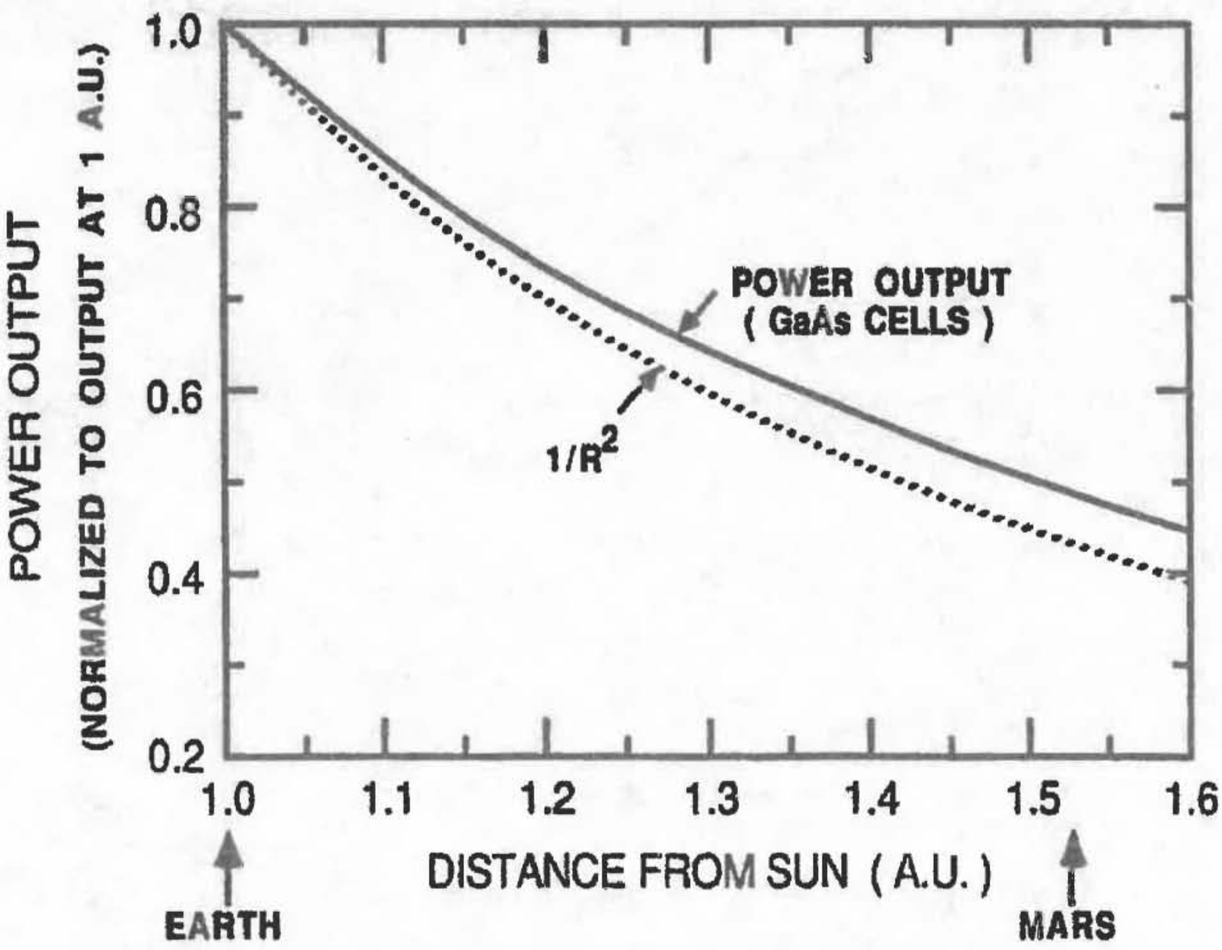

Figure 3-2. Solar Photovoltaic Array Power Output vs. Distance from the Sun

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SECTION 4

100-MW CLASS NUCLEAR ELECTRIC PROPULSION

4.1 INTRODUCTION

A Nuclear Electric Propulsion (NEP) system consists of a nuclear reactor and a thermal-to-electric conversion system, as well as a power processor unit (PPU) and electric thrusters. Unlike solar photovoltaic arrays, which scale approximately linearly with power level (i.e., a roughly constant specific mass), a nuclear power supply has the ability to make use of significant economies of scale. Thus, whereas a solar photovoltaic array in a 100-MW_e class Solar Electric Propulsion (SEP) vehicle has a specific mass of 3.6 kg/kW_e, a similar-power nuclear reactor with a high-temperature, Rankine dynamic power conversion system has a specific mass of only 0.9 kg/kW_e. Therefore, there is the potential for significant mass and trip time savings with NEP over SEP.

This potential benefit, however, is offset by the infrastructure required to base the NEP at a Nuclear Safe Orbit (NSO) of, typically, 700 to 1000 km altitude. This high altitude is required to ensure that, in the event of a catastrophic failure, there will be sufficient on-orbit stay time for any radioactive components to decay to safe levels before re-entering Earth's biosphere. The actual altitude depends on the ballistic coefficient of the vehicle (i.e., mass versus drag) and the levels of harmful nuclear isotopes that must decay to safe levels before air drag causes the vehicle's orbit to decay and re-enter. In this study, a 1000-km NSO is assumed. Also, it was assumed that a combination of stand-off distance and (limited) 4-pi steradian shielding would prevent damage to other vehicles or interference with science experiments (e.g., gamma-ray astronomy) in nearby orbits. However, these issues need to be addressed in detail in future studies of 100-MW_e class space nuclear power systems.

One interesting aspect of NEP operation that has been identified in previous studies is that an NEP vehicle can safely travel once, initially, from low Earth orbit (LEO) to NSO, because the reactor starts out "cold" (little or no harmful nuclear isotope inventory). As the reactor is operated and the vehicle begins to spiral out to NSO, the rate at which harmful nuclear isotopes build up is such that, were the system to fail at that point, the orbital lifetime achieved at that point would exceed the time required for safe decay of the harmful isotope inventory that has been produced to that point. Thus, a NEP vehicle can boost itself the first time to NSO. However, after prolonged operation it cannot return to LEO for periods typically given as several hundred years.

A schematic of a 100-MW_e class NEP vehicle is shown in Fig. 4-1. The vehicle configuration is dominated by the radiators required to radiate waste heat from the thermal-to-electric power conversion system. As will be shown below, the radiators also represent a significant fraction of the vehicle mass. Finally, an Orbit Transfer Vehicle (OTV) infrastructure will be required to transfer payloads and propellants to the 1000-km NSO assumed in this study; however, as seen below, this infrastructure represents a small fraction of the total initial mass in low Earth orbit (IMLEO).

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SECTION 5

MAGNETIC SAILS (MAGSAILS)

5.1 INTRODUCTION

5.1.1 Background

The magnetic sail, or magsail, is a novel concept recently introduced by Robert Zubrin and Dana Andrews.[1-3] A literature survey uncovered no previous description of such a device. Figure 5-1 shows a conceptual diagram of the magsail concept. It consists of a cable of superconducting material, millimeters in diameter, which forms a hoop that is tens to hundreds of kilometers in diameter. The current loop creates a magnetic dipole which diverts the background flow of solar wind. This deflection produces a drag-force on the magsail radially outward from the sun. In addition, proper orientation of the dipole may produce a lift-force which could provide thrust perpendicular to the radial drag-force. The combination of these forces can be used to transport the magsail and cargo on interplanetary or interstellar missions.

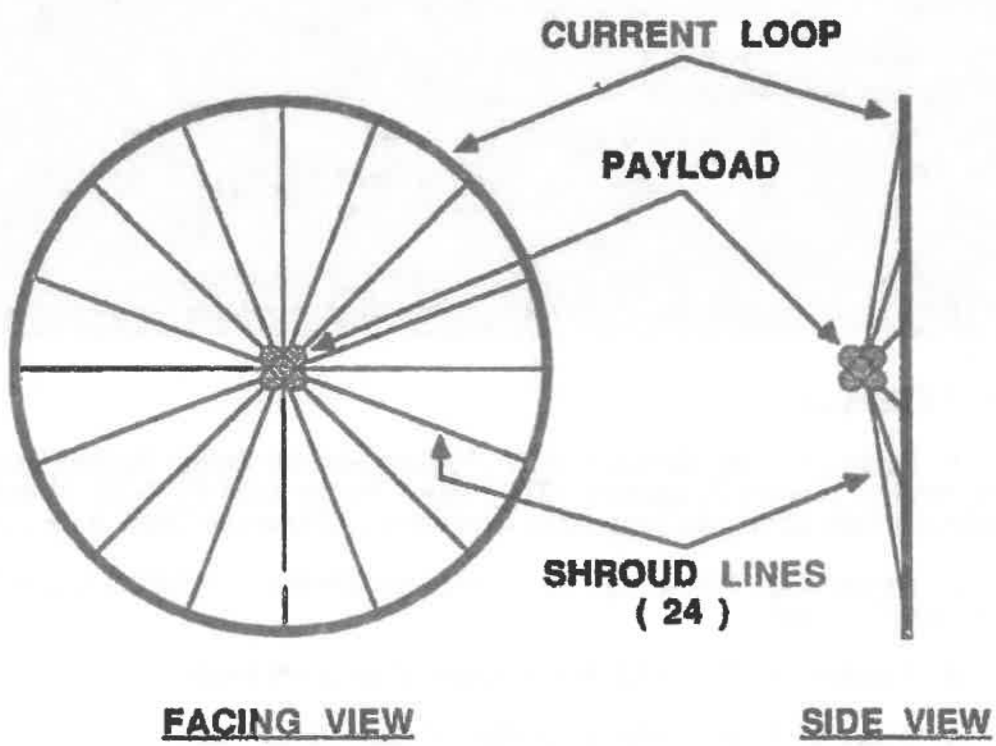


Figure 5-1. Magsail Deployed With Payload, Normal Configuration (Adapted from Reference 3)

5.1.2 Operational and Technical Feasibility Issues

As a relatively new concept, the magsail possesses a number of unresolved operational and technical feasibility issues. Dr. John L. Callas of JPL assisted in the

6

definition and evaluation of these feasibility issues,[4] which include thermal control, structures, radiation, superconductor technology, attitude control, deployment, planetocentric operation, and interaction with the solar wind. Each of these issues is described in detail below. Solutions to some of these issues may require the application of advanced technology (e.g., superconductors), others may only require innovative engineering (e.g., thermal control).

5.1.2.1 Thermal Control. For the current-carrying cable to remain superconducting, its temperature must be maintained below the critical temperature of the embedded superconductor. Preliminary thermal modeling indicates that in addition to passive reflective coatings, some form of active cooling system will be required to maintain the magsail cable below 100 K in Earth-Mars space.

If the thermal control scheme is based upon a continuous capability to orient the sail (i.e., to maintain a "hot side" and a "cold side" with different absorptivity and emissivity), temperature control during deployment and inflation may be difficult. For example, if the superconductor is inadvertently or intentionally quenched, attitude control is lost and the cable may drift from proper orientation, warming it above the superconductor critical temperature (T_c). This event could pose a catastrophic failure mode because attitude control could not be regained until the cable could again be cooled and powered-up.

5.1.2.2 Structures. An initial baseline design discussed by Zubrin and Andrews[2] describes a magsail 64 km in diameter, with a cable diameter of approximately 5 mm. These dimensions suggests that the structure will be susceptible to vibrational motion; the cable material must be very malleable to survive this motion without fracture. Current high-temperature superconductors are like brittle ceramics in terms of their material properties. Whether or not a superconductor material can be manufactured possessing the proper resiliency and malleability is an important feasibility issue. It may be necessary to enclose the superconducting cable in a flexible sheath of Kevlar (or some other material like Kevlar which is effective for tether applications, but is more appropriate for a low-temperature application than Kevlar), to provide flexible tensile support.

5.1.2.3 Radiation. The magnetic field of the magsail may generate local Van Allen-type radiation belts. These belts may pose a significant radiation hazard for payload or crew in the vicinity of the magsail, though not at the geometric center of the magsail hoop. The background solar wind and cosmic-ray radiation may also induce long-term cumulative radiation damage in the superconducting hoop, degrading the superconducting properties of the material.

5.1.2.4 Superconductor Technology. The baseline magsail designs of Zubrin and Andrews rely upon significant advancements in superconductor technology such that the assumed critical current density of 1 to 2 $\times 10^{10}$ Amps/m^2 must be achieved in bulk form in high-temperature superconductors. Recent findings[5] suggest that Type II superconductors designed for high-critical-temperature operation ($T_c > 77$ K) are susceptible to "giant flux creep" (which creates resistance in the superconductor) in the presence of a magnetic field. The superconductor characteristics and operating environment assumed for current magsail designs describe a demanding combination of conduction current density, critical temperature, and magnetic flux density. If no solution is found to the problem of giant flux creep, subsequent reduced superconductor conduction current density and critical temperature will significantly reduce magsail performance.

It is also necessary to design the magsail system to survive a "quench", in which the superconducting material loses its ability to conduct current without resistance. A quench may be caused by a rise in temperature above the superconductor critical temperature, or a rise in the magnetic flux density above the critical field of the superconductor. In addition, there may be situations in which it will be desirable to significantly reduce or eliminate the current in the superconducting cable (e.g., for navigation, or to release charged particles trapped in induced radiation belts). Quench capability could be provided by an external resistor bank.

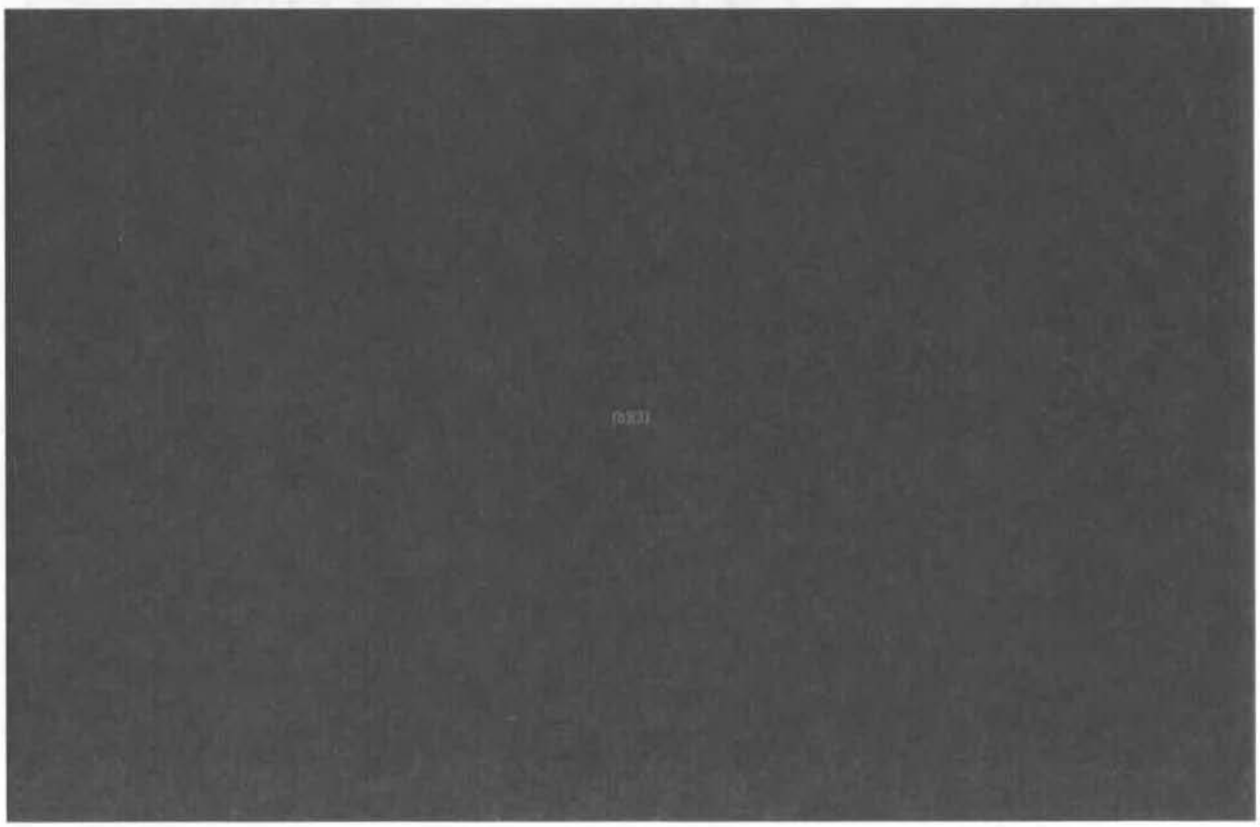
5.1.2.5 Attitude Control. One potential attitude control scheme is similar to that proposed for use on solar sails in which articulated control vanes (separate small superconductor loops for the magsail) are used to modulate the center of pressure of the sail while the center of mass remains fixed. A second approach, again proposed for solar sails, would be to shift the center of mass of the magsail could by moving part of the payload out along a shroud line while the center of pressure remains fixed. The difference in location between the center of pressure of the magsail and the center of mass would induce a torque and small angular acceleration. For example, assuming that 50 MT of payload could be offset 10 % of the hoop radius, the resulting torque could change the orientation of the baseline magsail by 90° in 10 to 12 hours. Several issues arising from this scheme remain unresolved. Local variations in the solar wind density may cause a random perturbation of the center of pressure which complicates the application of this attitude control scheme. In addition, the slow response time (e.g., 10 to 12 hours to rotate 90°) may make it difficult to execute a planetocentric "pumping" orbit-raising maneuver if thrust vectoring is required.

5.1.2.6 Deployment. The size and electrical current in the superconducting magsail cable imply significant energy storage. For example, the energy stored in the cable of a 64-km diameter, 10^{-5} Tesla magsail is approximately 8×10^{10} Joules. A continuously operating 10-kW solar array would require approximately 93 days to energize the cable to full power. This large energy storage suggests two potential problems. First, if magsail deployment and "inflation" require a large amount of time, the magsail may lack attitude control during this period, which could lead to a subsequent loss of thermal control, as well as unusual mechanical stresses. Second, it may be difficult to modulate the current in the magsail cable in the manner required for a "solar-pumping" maneuver described below. One possible method for rapid sail deflation would be to redirect part of the cable electric current to a radiative resistor bank, although this may aggravate the difficult magsail thermal control problem.

5.1.2.7 Planetocentric Operation. Thus far, it is not known if the magsail can be operated near a planet's magnetosphere. In their analyses, Zubrin and Andrews have constrained magsail operation to heliocentric space: "For our reference spacecraft, starting in very high Earth orbit and about to orbit the sun at Earth radius...".[2] Clearly, a magsail cannot be used within a planet's magnetosphere (between the magnetopause and the planet's surface) because there is no solar wind there. The minimum distance from the center of the Earth to the Earth's magnetosphere is 10 Earth radii, or 64,000 km.[6] Other planets have significantly varying magnetosphere sizes based on the planet's magnetic field strength.

It will be difficult for the magsail to operate in planetocentric orbits of even higher altitude (above the magnetopause), because in order to gain altitude in the orbit the magsail must execute a "solar-pumping" maneuver analogous to that originally conceived for solar sail orbit raising and escape. In a planetocentric solar-pumping maneuver, the solar sail is feathered such that solar photon pressure is minimized when the sail is heading sunward. The sail is then re-oriented to maximize solar photon pressure when it is flying away from the sun. In this way, the apogee of the sail orbit is incrementally boosted to achieve higher-energy orbits or escape. However, unlike a solar sail, re-orienting the magsail hoop does not significantly modulate the radial solar wind drag-force. In order for the magsail to execute a solar-pumping maneuver, the drag-force "thrust" (and possibly "lift") would be modulated during each orbit. In order to reduce or eliminate the radial drag force on the upwind leg, the electric current in the magsail cable could be reduced or eliminated. This current-modulation scheme suggests several operational issues. The magsail may lose attitude control, as described above. In addition, the circular shape of the superconducting cable is a result of the solenoidal hoop-stress imparted to a current-carrying cable in an ambient magnetic field; if the magsail were quenched, the hoop may lose its shape as a result of gravity gradient or other perturbation forces. The magsail would be recharged before thrust could be generated on the downwind side. As suggested above, if the cable recharge is constrained by the onboard magsail power supply, then recharging the cable to full power may be time consuming (perhaps beamed power could be utilized). A potential solution to this problem is to execute the solar-pumping maneuver at a reduced magsail energy level to allow quicker magsail inflation and deflation.

5.1.2.8 Modeling of the Solar Wind-Magsail Interaction. Both a particle model and a fluid model have been proposed for calculation of the magsail drag-force radial thrust and lift-induced tangential thrust. A particle-based model of the solar wind-magsail interaction was developed by Callas[4] which roughly confirmed the results of Zubrin's particle model.[1,3] Callas' model predicts a thrust of approximately 200 N for the 64-km diameter, 10^{-5} Tesla magsail. Both Callas and Zubrin have concluded that a plasma fluid model is probably most appropriate for modeling the radial and possible tangential thrust of the magsail. Zubrin's plasma fluid model predicts a minimum (quiet solar wind) thrust of 538 N and an average lift-to-drag ratio of 0.28. In the Mars cargo mission analysis, it is shown that the initial mass in LEO is sensitive to the estimated magsail radial thrust, so this parameter is allowed to vary from 200 to 500 N. Perpendicular thrust (positive lift-to-drag) was not considered in the orbital analysis and mission performance. Further work is needed to fully understand the radial and tangential thrust characteristics of the magsail.



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SECTION 6

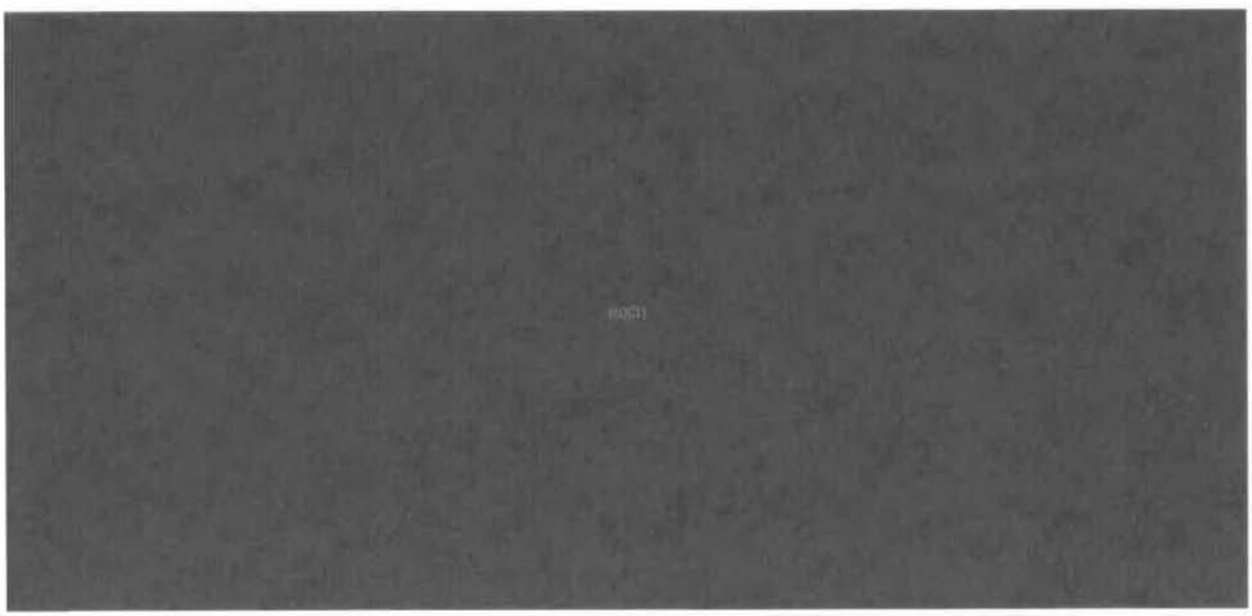
MASS DRIVERS AND RAIL GUNS

6.1 INTRODUCTION

Mass drivers (MD) and rail guns (RG) can be used as electric propulsion thrusters.[1,2] In both concepts, a propellant "pellet" is accelerated in a "bucket" or "container" that couples to an externally applied electromagnetic field. The propellant "pellets" are accelerated to high velocities (e.g., 12 km/s corresponding to an I_{sp} of 1200 lb_f-s/lb_m) and fired from the vehicle to produce thrust.

As shown in Fig. 6-1, the two concepts take different approaches to accelerating the propellant. The mass driver consists of many solenoid magnets which are energized in series to pull a payload bucket which contains its own magnet to couple to the externally applied fields. Very large mass drivers can be used to directly catapult vehicles from bodies such as the Moon which lack an atmosphere. Because any material can be placed in the payload bucket, a mass driver, when used as a reaction engine, can use any material as propellant. In this study, the option of using lunar-produced materials (e.g., lunar soil, oxygen, etc.) for propellant was considered as a means of reducing the initial mass in LEO. In general, mass drivers are large and complex, but have a high electric-to-jet power efficiency (70 to 90 % overall).

The rail gun is currently under consideration for use as a kinetic-energy weapon by the Strategic Defense Initiative Office. Although smaller and simpler than a mass driver, rail guns have a lower efficiency (45 % for the vehicle considered here) than a mass driver. Conceptually, the rail gun consists only of a power supply and two electrically energized rails. A "bucket" with a conductive armature is placed on (between) the rails; current flow through the armature produces a Lorentz force which causes the bucket to accelerate down the rails. Erosion of the rails by the bucket armature is a serious problem that currently limits rail guns to a small number of firings; major improvements in lifetime are required because use as a reaction engine might require on the order of 10^8 firings for a Mars mission.



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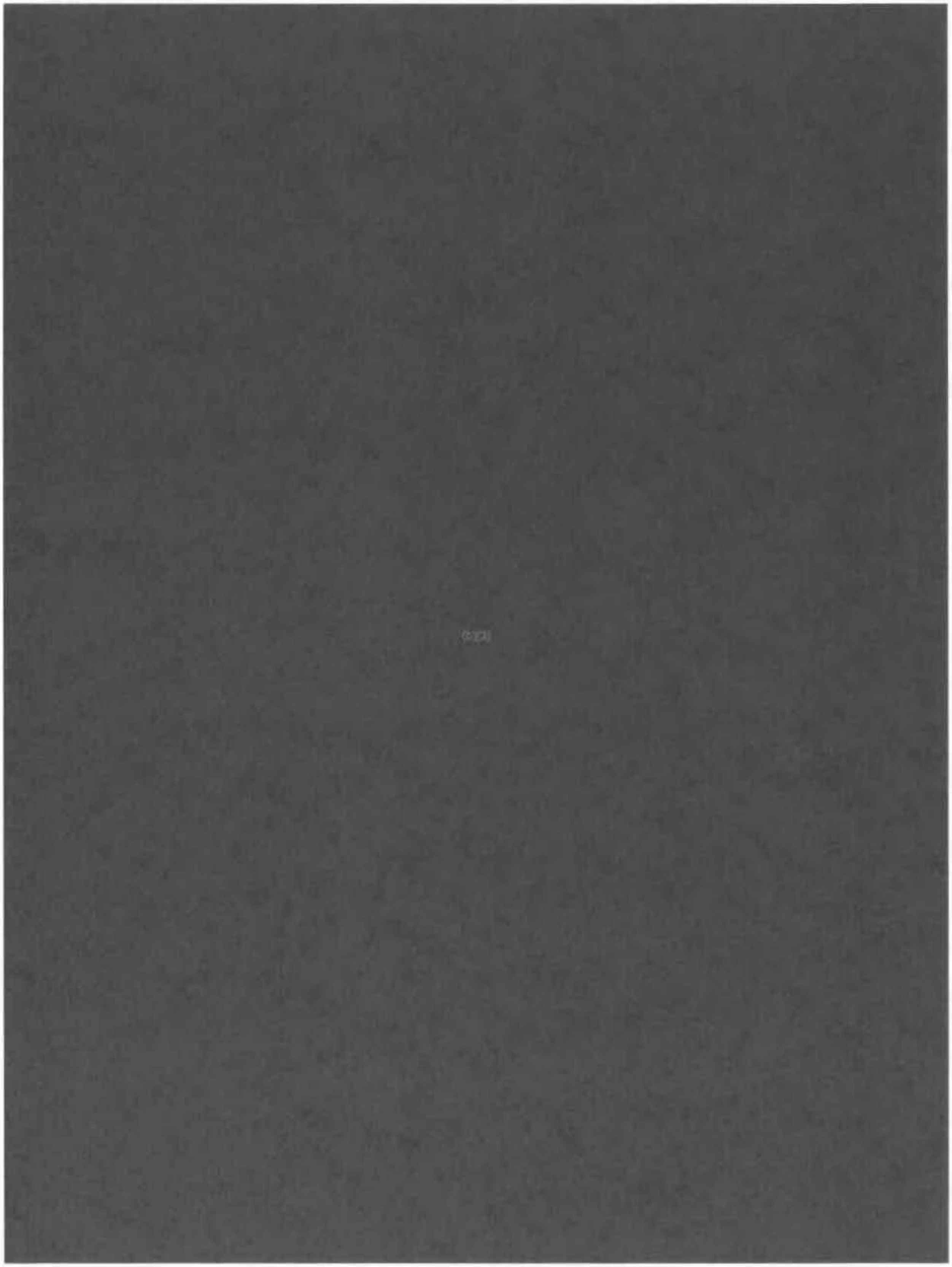

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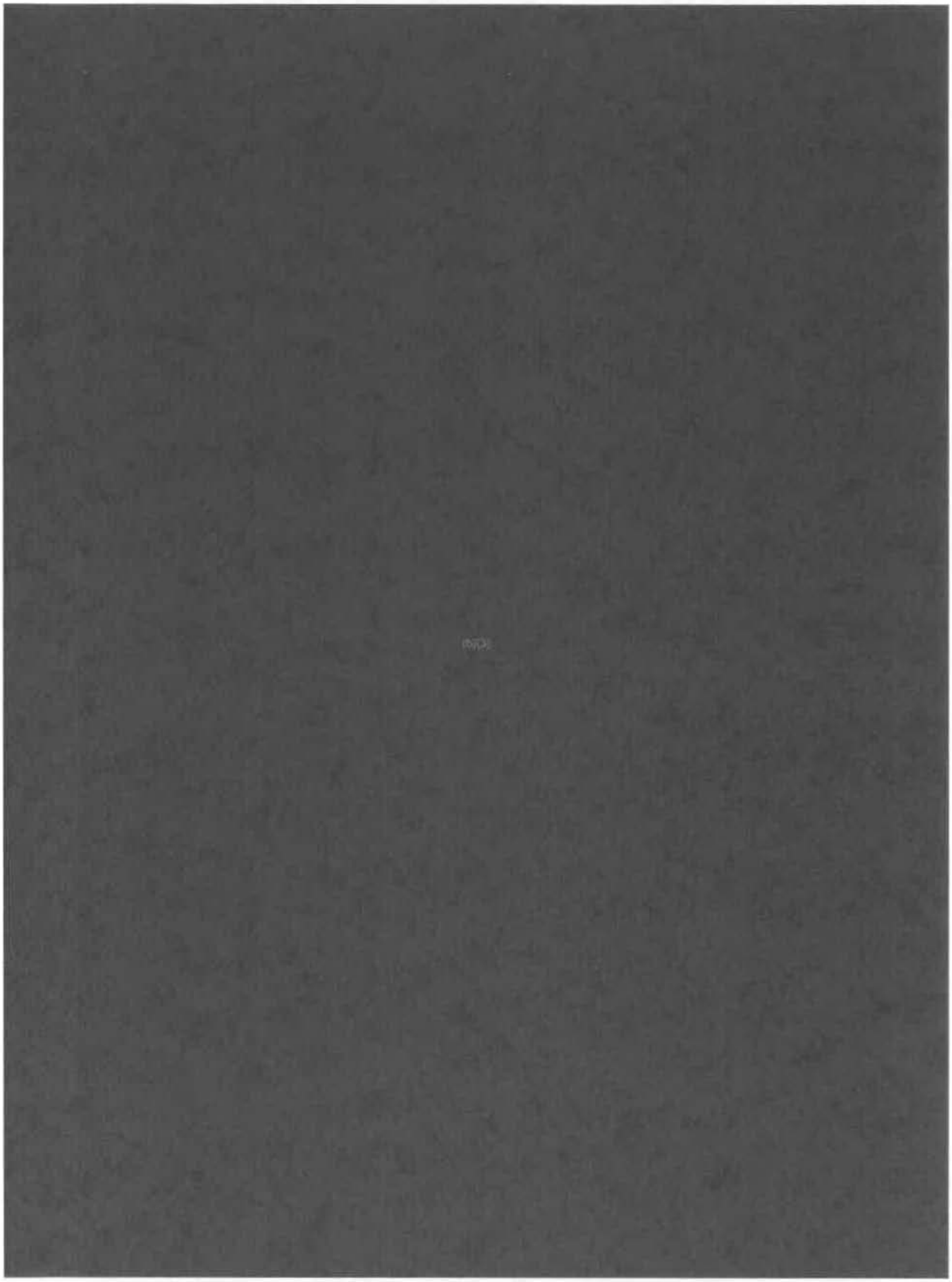

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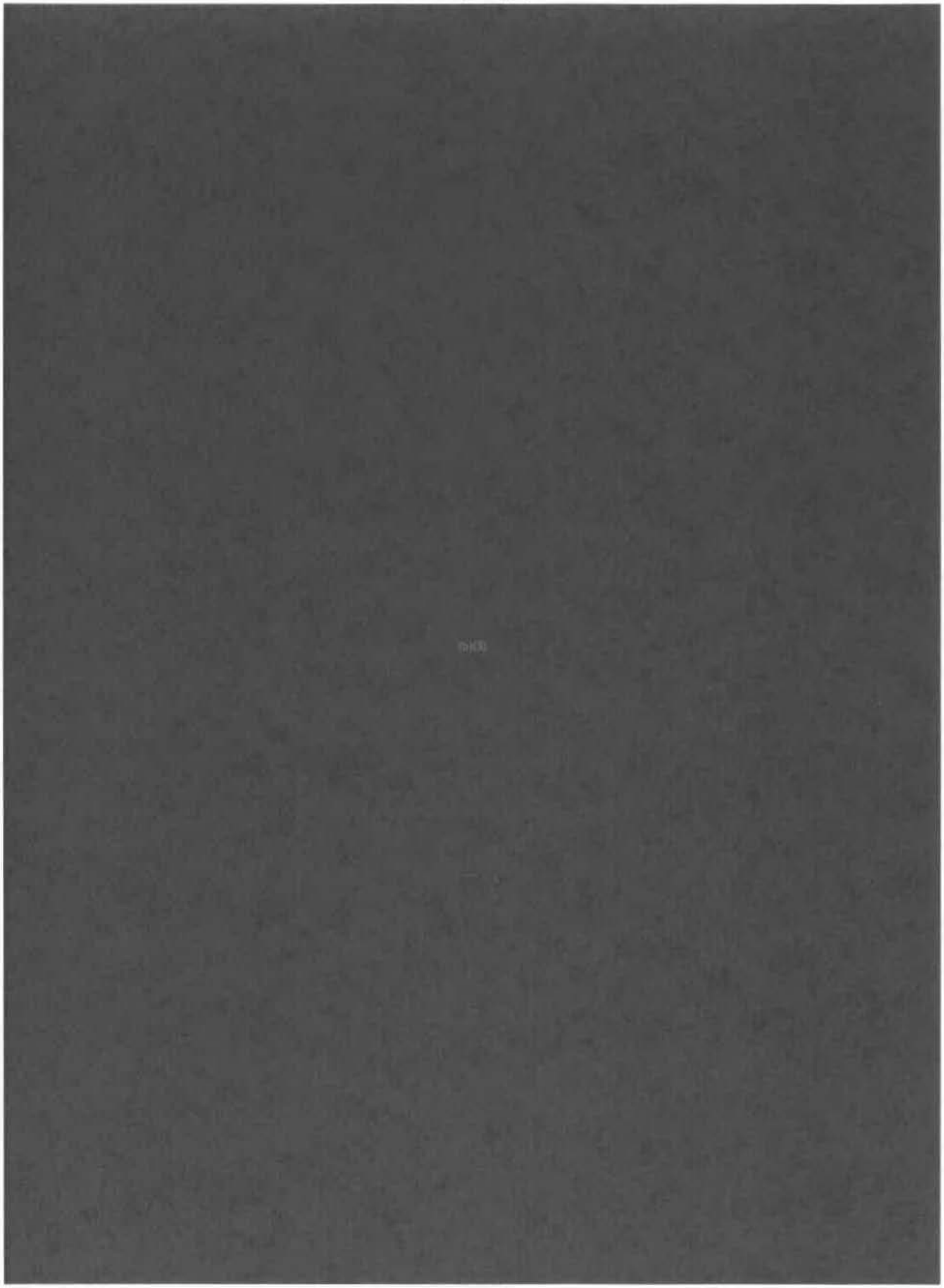

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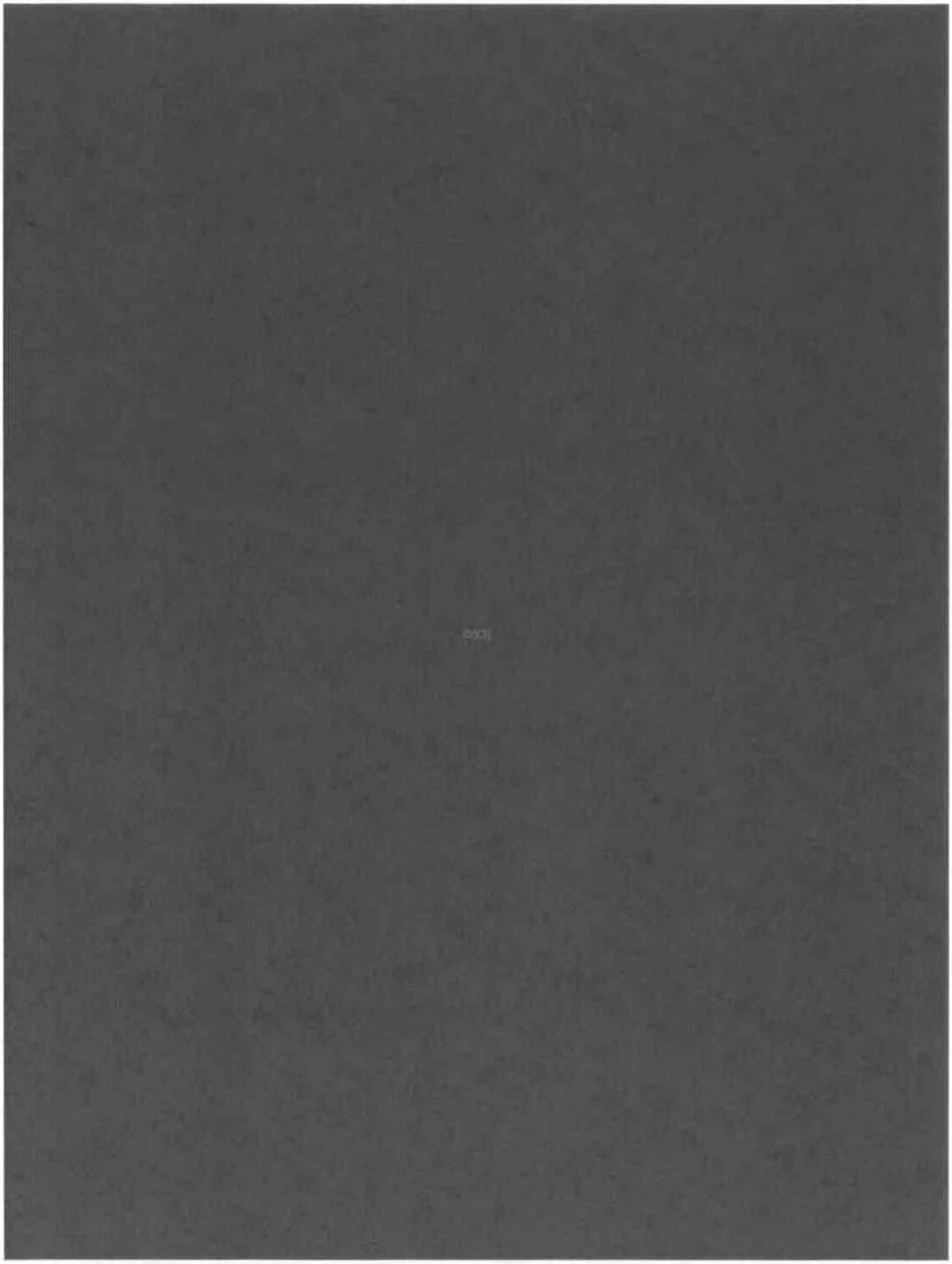

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SECTION 8

SOLAR THERMAL PROPULSION

8.1 INTRODUCTION

In the Solar Thermal Propulsion (STP) concept, shown in Fig. 8-1, sunlight is collected by a large inflatable "mirror" and focused into a thruster where the sunlight is absorbed and used to heat a propellant (e.g., hydrogen) which then expands out through a conventional nozzle. There are several similarities between solar thermal and laser thermal propulsion, as will be described below. The Air Force Astronautics Laboratory is currently funding STP thruster development. A prototype engine, using a rhenium-tube heat exchanger, has achieved specific impulses (I_{sp}) in the 800 lb_f-s/lb_m range. Advanced STP thruster concepts, using particle-bed heat exchangers or particulate absorption directly in the propellant, are projected to achieve I_{sp}s on the order of 1200 lb_f-s/lb_m.[1]

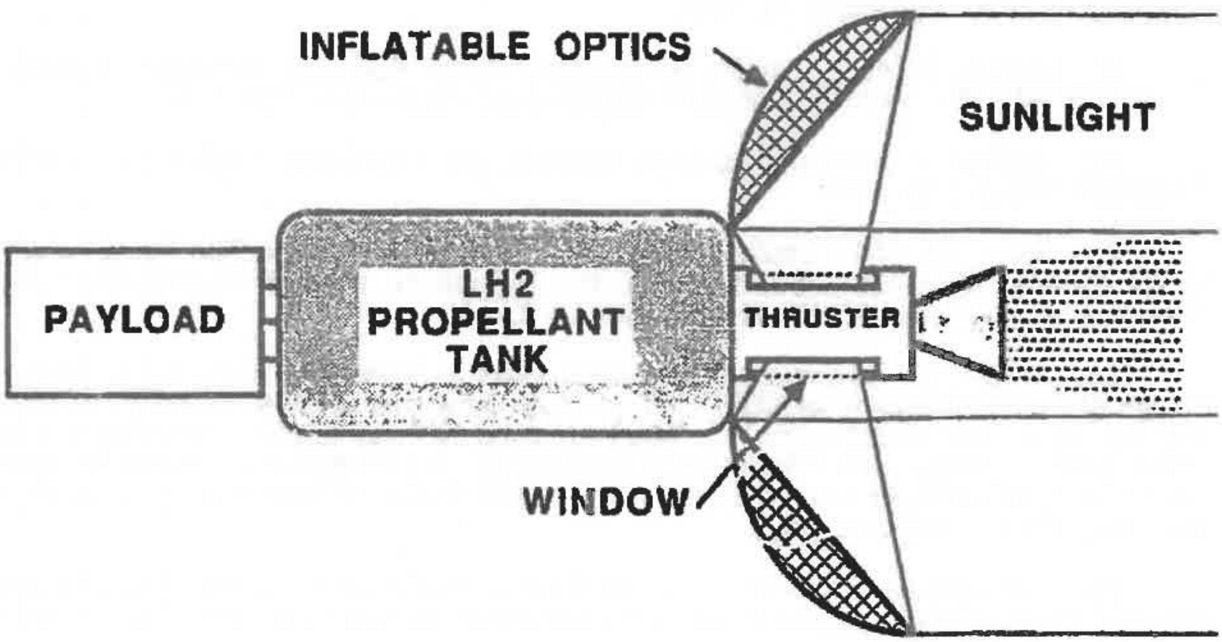

Figure 8-1 Solar Thermal Propulsion Concept

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SECTION 9

TETHERS

9.1 INTRODUCTION

Tether concepts for propulsion and power have been investigated within the last decade for a variety of space missions. Two classes of tether systems are electrodynamic tethers, which interact with a planetary magnetic field, and non-conducting tethers which interact with the gravitational field. The present study investigates the benefits of the latter class for propulsive assist in an unmanned, Earth to Phobos cargo mission.

An object placed in orbit about a planetary body remains in orbit because the inward-directed gravitational force is balanced by the inertial or centrifugal force, in response to which the body moves outwards. The tether systems considered here begin operation with the entire system (which includes the payload, propulsive stages, tethers, and tether station) in a circular orbit. The tether is then deployed with the payload and transfer vehicle at one end of the tether and the station at the other. If the payload orbit is to be raised, the tether is deployed "up" or radially outward. Conversely, if the payload orbit is to be lowered, then the tether is deployed "down" or radially inward. After a period of time the tether reaches mechanical equilibrium in a vertical orientation. In addition, the center of mass is located at an altitude slightly lower than the original altitude because of a net tidal force which has done work on the entire system. Once any transient motions have been damped out, the entire system orbits the planet in a circular orbit with uniform angular velocity. For an outbound mission, the payload and transfer vehicle will be above the center of mass and have a velocity which is super-circular, i.e. , faster than the circular orbital velocity at the payload's altitude. The station, on the other hand, will be located below the center of mass and have a velocity which is sub-circular. The payload is then disengaged from the tether and enters a larger elliptical orbit with its perigee located at the release point. The station enters a lower-energy elliptical orbit with its apogee located at the release point. These orbits are depicted in Fig. 9-1. The tether is then reeled back into the station, after which a pair of propulsive burns are required to bring the station back up to its original circular orbit. The payload and transfer vehicle then perform a propulsive burn to reach the required velocity for injection to Mars (C_3=9.541 km^2/s^2).

In this study , the scenario described above corresponds to operations in low Earth orbit (LEO) where a large (500 MT) station is used to assist a payload and transfer vehicle (64.3 MT) to achieve the required earth escape velocity. At Mars, the procedure is reversed with the payload and transfer vehicle being captured by the Deimos tether station and transferred to the Phobos station. An important operational difference is that Deimos and Phobos are used as tether "stations"; the fact that these moons are orders of magnitude more massive than the payload eliminates the necessity to reboost the "station" back to its original orbit.

A two-stage aerobraked chemical (O_2/H_2) vehicle, similar to the baseline chemical vehicle described in Section 1, was used to inject the payload towards Mars after release from the Earth-orbit tether. The second stage continues to Mars where it aerobrakes into an elliptical orbit which allows it to rendezvous with the Deimos tether. This mission scenario is illustrated in Fig. 9-2.

Various facets of the tether system and Mars mission have been investigated in some detail in the past. Paul Penzo of JPL has outlined design requirements as well as operation for a LEO tether transportation system.[1] In addition, he has considered issues related to operation of a Mars tether transportation system.[2] The change in center of mass due to the net tidal force has been proposed as a means of satellite relocation by Geoffrey Landis of the Lewis Research Center.[3]

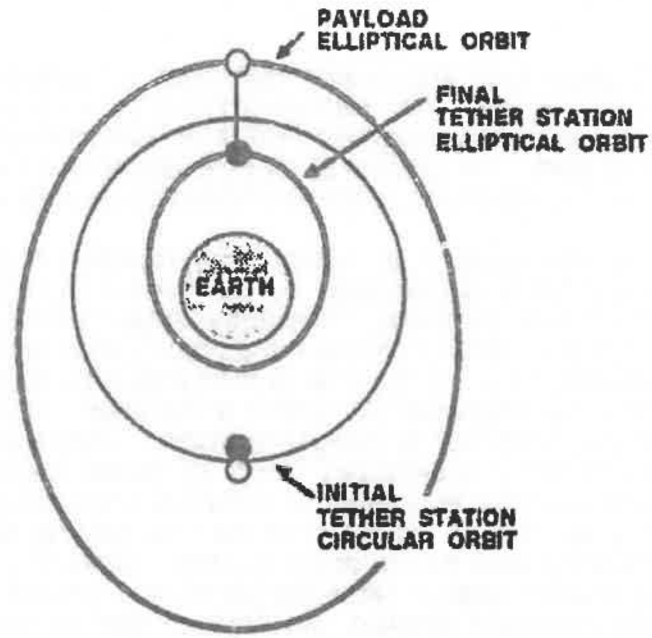

Figure 9-1. Tether System Orbits at Earth

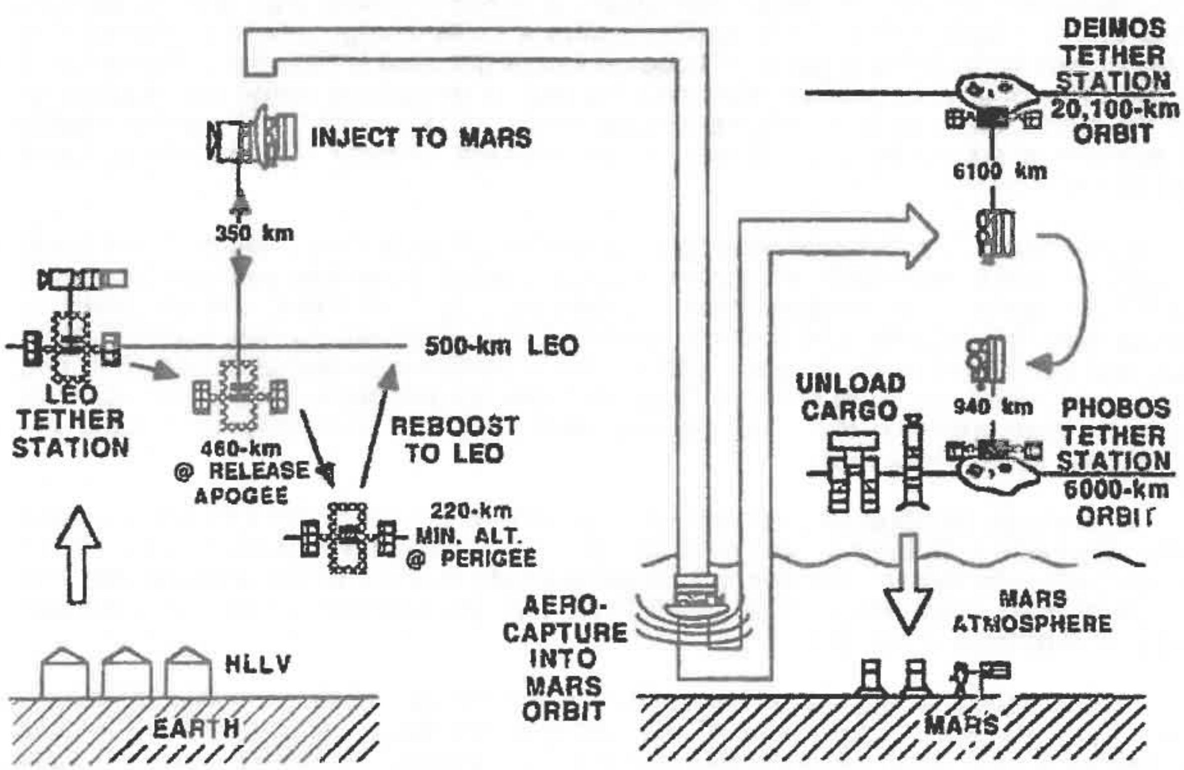

Figure 9-2. Tether Propulsion Mission Scenario

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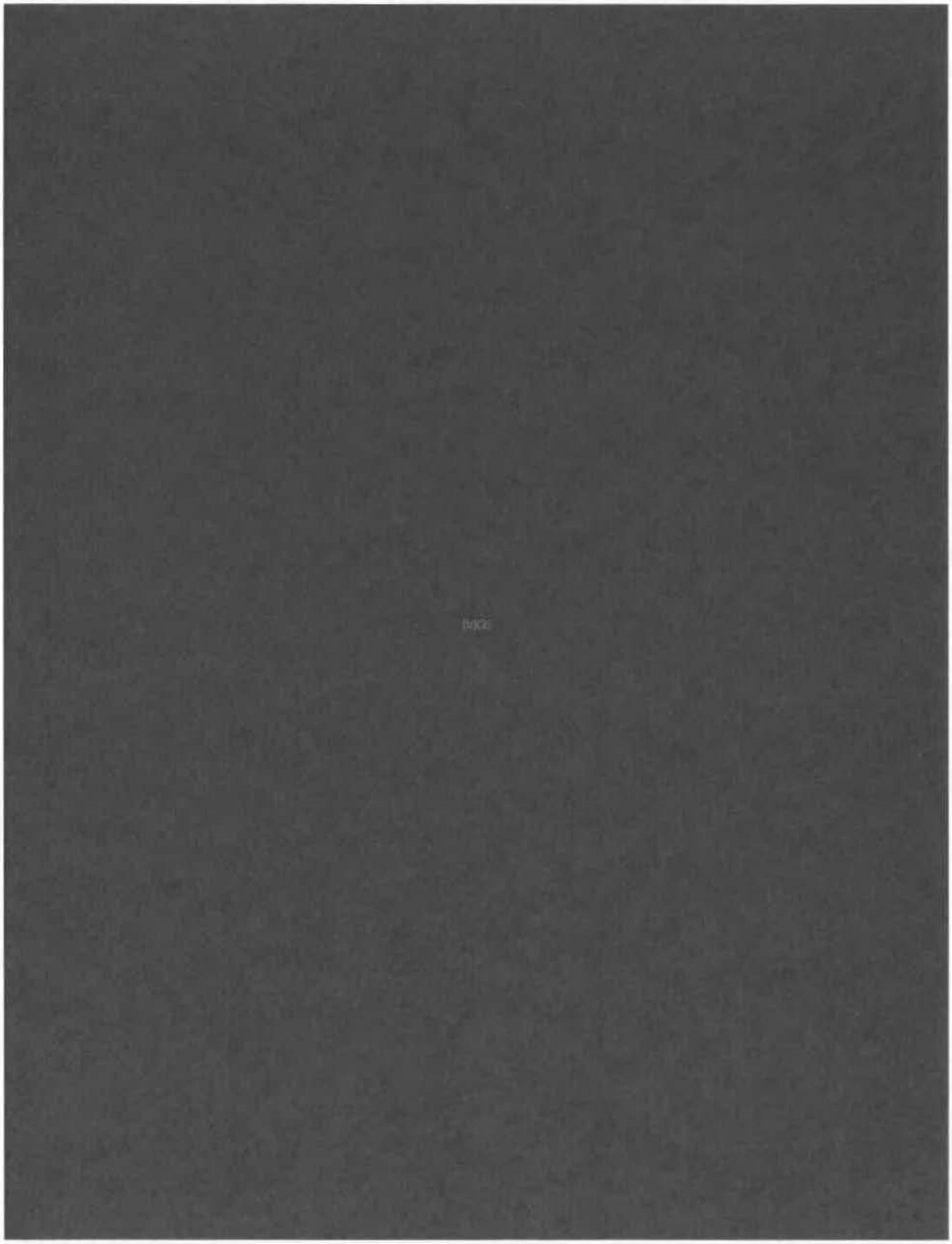

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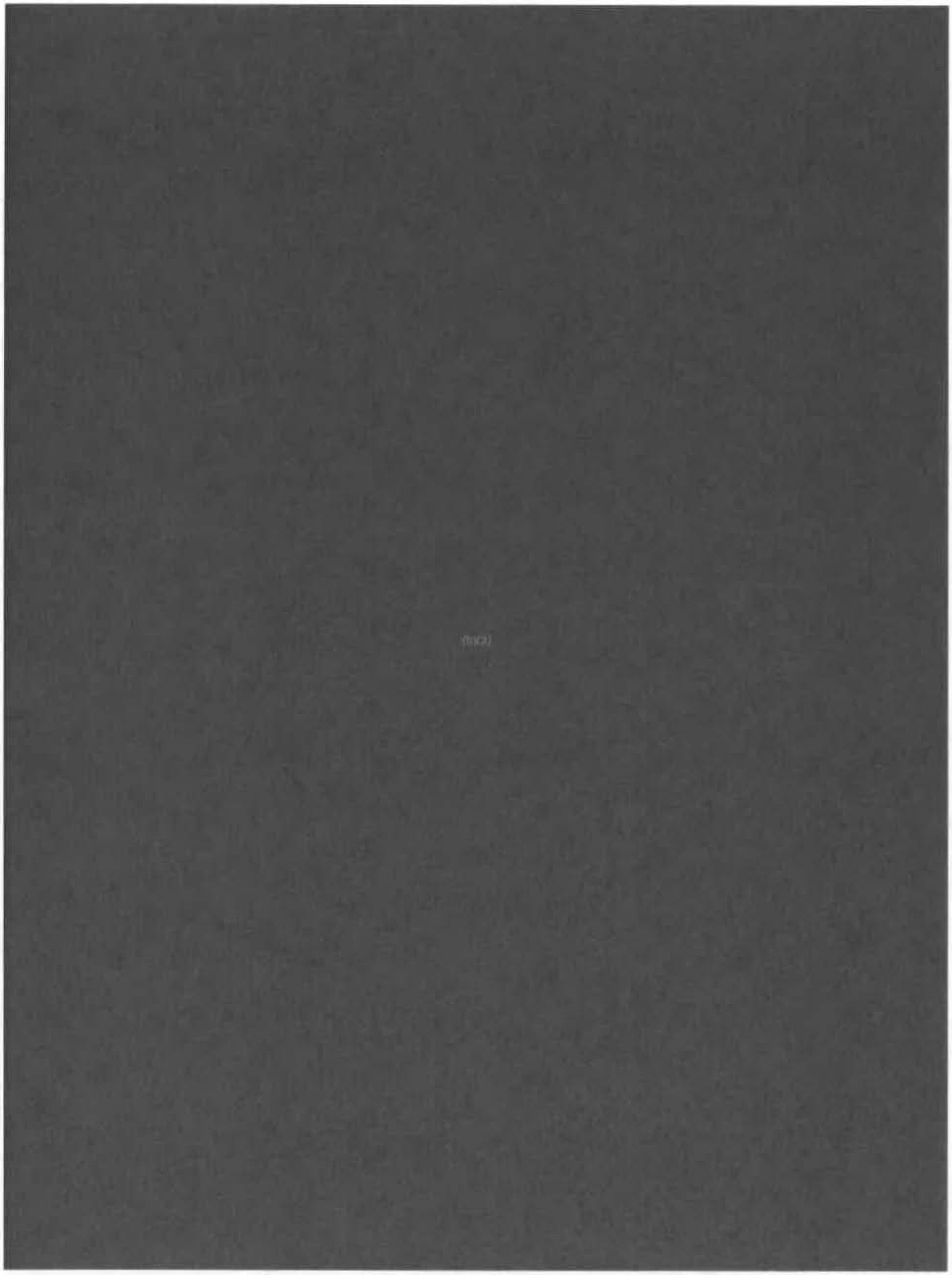

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SECTION 10

CONCLUSIONS

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SECTION 11

APPENDIX

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APPENDIX B

ULTRA-HIGH POWER NUCLEAR ELECTRIC PROPULSION

B.1 INTRODUCTION

The following section describes the first order design of a Nuclear Electric Propulsion (NEP) concept which can be scaled over a range of electric power levels from 20 to 500 MW_e. The concept is based on a lithium-cooled pellet reactor driving a Rankine cycle dynamic conversion system. The thermodynamic cycle is used to turn a turbine which actuates an alternator producing three-phase electric power at high voltage. The three-phase power is rectified and utilized by a mercury ion propulsion system. A spacecraft mass scaling model is presented which gives propulsion system mass as a function of electric power level, payload, propellant mass, thrust time, and specific impulse. A performance model for the propulsion system is also presented which allows calculation of thruster efficiency as a function of specific impulse. The work described in this report was conducted in support of an Inertial Confinement Fusion (ICF) spacecraft propulsion system study team.[1]

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B.5 ACKNOWLEDGMENTS

This reactor concept described here was conceived by David Buden of Science Applications Incorporated (Ref. 2). Dr. Buden was kind enough to provide a wealth of detailed design information which was adapted for this particular application. The dynamic conversion system described here is based on the work of Ms. Kathy Murray of the Energy Technology Engineering Center (Ref. 7). We would like to acknowledge the vital assistance given by both of these people. We would also like to thank Dr. Graeme Aston of JPL for providing valuable guidance in the area of ion thruster technology performance projections (Ref. 21).

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