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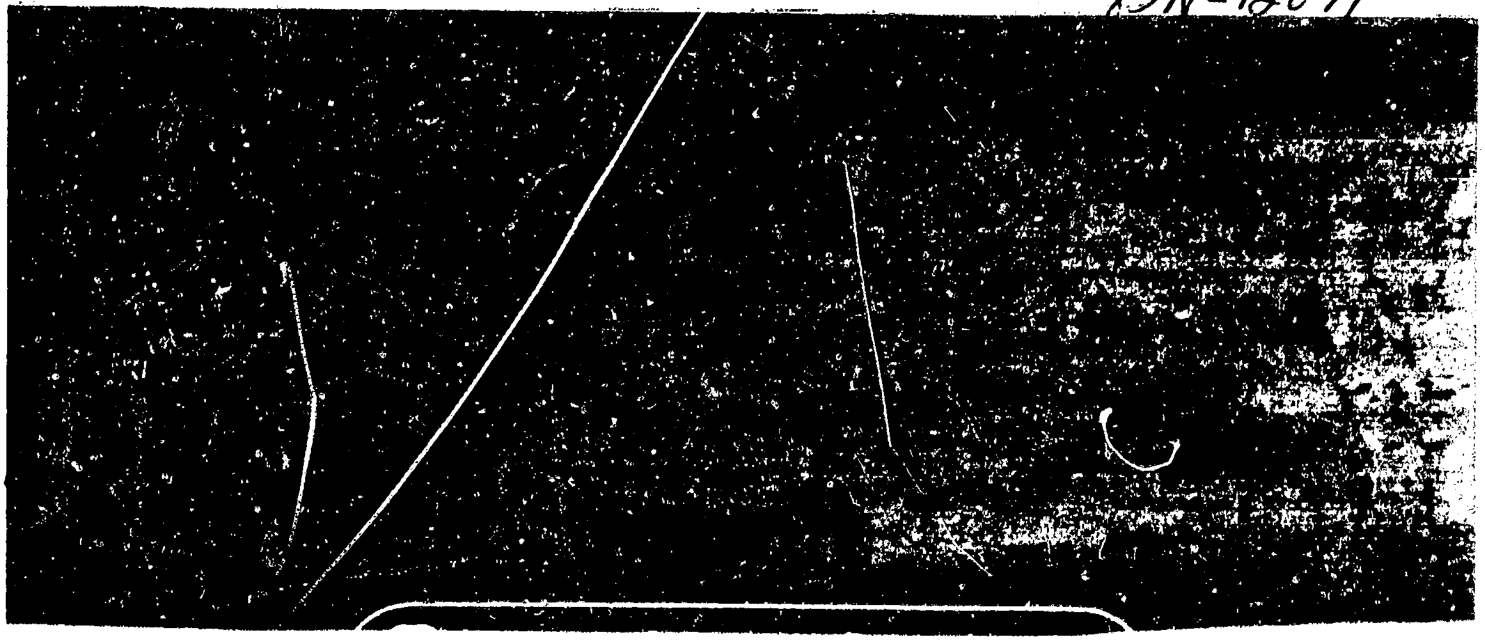

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FINAL TECHNICAL REPORT

on

PRELIMINARY ANALYSIS
OF SPACE MISSION APPLICATIONS
FOR ELECTROMAGNETIC LAUNCHERS

to

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LEWIS RESEARCH CENTER

(Contract Number NAS 3-23354)

August 30, 1984

by

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FOREWORD

This Battelle Columbus Laboratories study was sponsored by NASA Lewis Research Center under NASA Contract Number NAS3-23354, for the purpose of providing a preliminary analysis of space mission applications for electromagnetic launchers. Work was conducted from November 1982 through June 1984. Battelle's assessment involved: (1) development of mission models and requirements for electromagnetic launchers; (2) reference system concept development; (3) preliminary economic assessment; (4) preliminary environmental assessment; and (5) assessment of technology development needs. To support the system concept development, NASA Lewis Research Center supported Dr. Henry H. Kolm and Dr. Peter P. Mongeau, EML Research, Inc., Cambridge, MA, and Dr. O. K. Mawardi, Collaborative Planners, Inc., Cleveland Heights, OH.

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1.0 EXECUTIVE SUMMARY

This Final Technical Report documents the findings of the "Space Mission Analysis of Electromagnetic Launchers," NASA Contract NAS3-23354. The Executive Summary (Section 1.0) contains the study background, objectives, approach, principal assumptions and requirements, and summarizes the reference concepts, major results, conclusions, and recommendations. Sections 2.0 through 8.0 present the technical details of the study results. Appendix A provides the references. Acronyms and abbrieviations are defined in Appendix B, while Appendix C contains metric-to-English conversion factors. Separate NASA-funded studies by Electromagnetic Launch Research, Inc., and Collaborative Planners, Inc., are attached to this report in Appendices D and E, respectively. Appendix F contains the distribution list for this final report.

1.1 Study Background

In 1982, Battelle Columbus Laboratories conducted a feasibility assessment of an electromagnetic Earth-to-space rail launcher (ESRL) to determine the viability of developing a rail launcher system in the 2020 time frame to launch material into space (Rice, Miller, and Earhart, 1982). Based upon an evaluation of potential applications, a reference concept was selected. The reference concept consisted of two missions; the first mission would launch projectiles to solar system escape velocities to dispose of commercial high-level nuclear waste, the second would launch cargo to low-Earth orbit to resupply U.S. space stations. The ESRL system would be based at an equatorial site, with two separate rail launcher tubes placed in underground shafts. One tube would be inclined at 20 degrees from the horizontal for Earth-orbital missions; the other would be vertical for solar system escape nuclear waste disposal missions. Each launcher tube would be 2.04 km in length, and surrounded by 10,200 homopolar generator/inductor units to transmit power to the rails. The two rail launchers would be powered by a common power plant. Total projectile masses were 6500 kg for Earth-orbital missions and 2055 kg for solar system escape missions. These projectile masses corresponded to payload masses of 650 kg and 250 kg respectively. The seemingly large ratios of projectile-to-payload were due to shielding requirements for nuclear waste and orbit-circularization propulsion system requirements for Earth-orbital payloads. Based upon preliminary technical, environmental, and economic analyses, it was concluded that the ESRL system appeared to be technically feasible and economically beneficial.

The previous ESRL study investigated only electromagnetic railgun launchers for Earth-to-space missions. Other types of electromagnetic launchers (EMLs) are currently being studied, the most well known of which are the coaxial accelerators. This study was conducted to investigate all types of EML concepts for performing a variety of space missions.

1.2 Study Objectives

The overall objectives of this study were: (1) to provide NASA Lewis Research Center (NASA/LeRC) with sufficient information such that a comparison could be made between various promising EML space mission concepts; (2) to develop mission models and requirements for EMLs; (3) to define reference system concepts; (4) to conduct preliminary analyses of economics and performance; and (5) to recommend areas of technology development.

1.3 Approach

The study approach emphasized the assessment of factors which would contribute to the comparison of the different types of electromagnetic launchers. These factors include mission definition and requirements, preliminary conceptualization of EML systems, economics, and technology status.

The analysis included four tasks: (1) Characterization of Candidate EMLs; (2) Development of Mission Models and Requirements; (3) EML Concept Analysis; and (4) Technology Assessment. Figure 1-1 emphasizes the overall study approach. The specific study tasks and their interrelationships are outlined.

Initially, the study addressed space mission applications of EMLs beginning in the year 2020 and continuing through 2050. The study was revised midterm to reflect a more near-term operational start-up in 2000 for the Earth-to-orbit mission and in 2010 for the lunar base supply mission.

In order to stay abreast of the rapid developments in this field, it was necessary to maintain contacts with others working in this area. This contact included attendance at two EML conferences, the American Defense Preparedness Association Seminar on Electromagnetic Launchers in February 1983 and the Second Symposium on Electromagnetic Launch Technology in October 1983. Also, experts in coaxial electromagnetic accelerator technology were contracted separately by NASA/LeRC to provide input to the preliminary conceptualization task.

1.4 Study Guidelines

The guidelines which were used in the performance of this study include:

- Battelle made maximum use of related studies and other associated data, as appropriate.
- Battelle considered EML systems available in the open literature.
- Peaceful space applications of EML systems were considered.

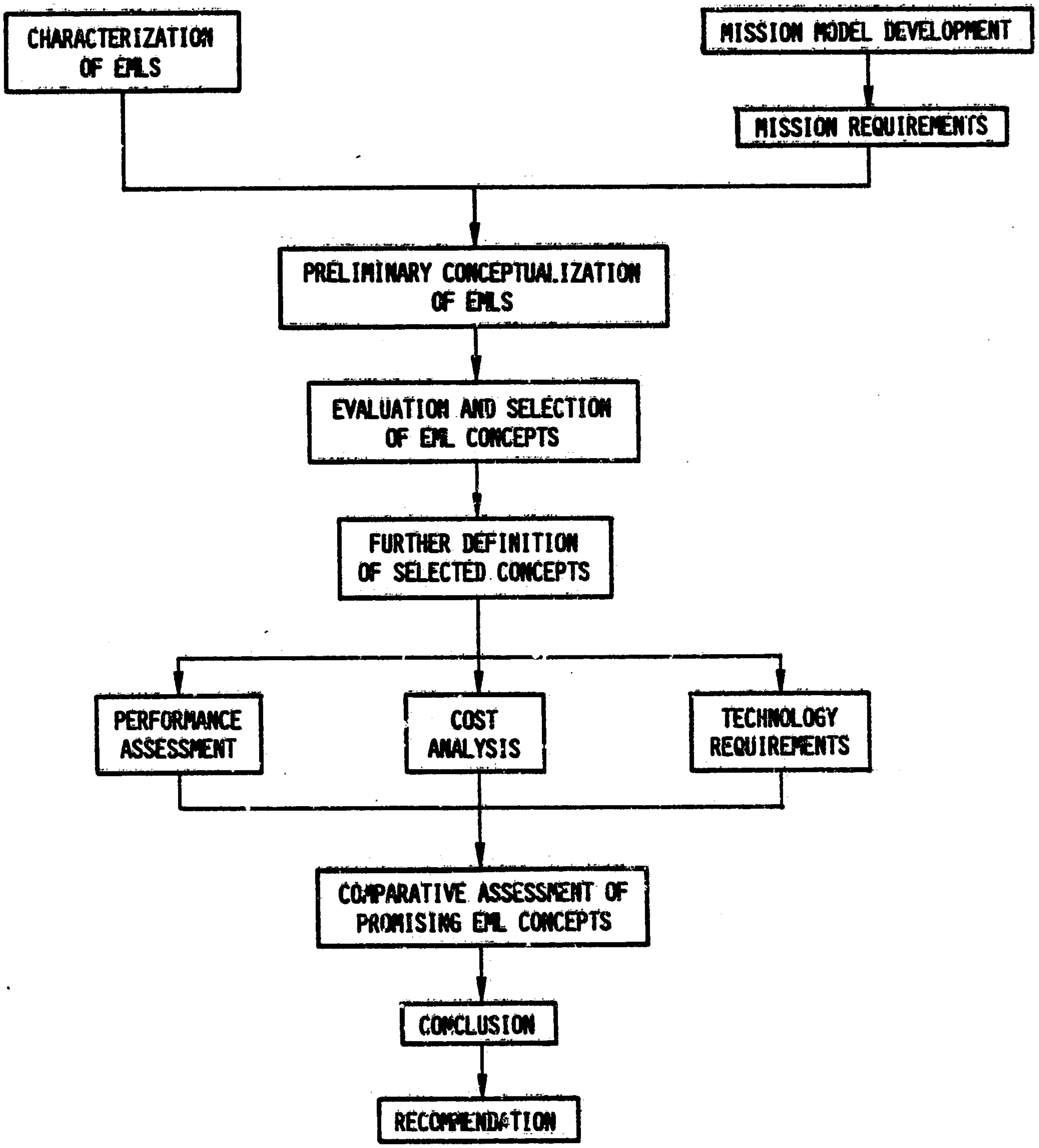


FIGURE 1-1. STUDY APPROACH

- Reusability of potentially recoverable EML projectiles was considered.
- All costs were given in 1981 dollars to compare with the previous Battelle study.
- Study activity was scoped to follow allocated funding resources.

1.5 Reference Concepts

Four reference concepts were selected following preliminary conceptualization and evaluation of the seven identified missions using railgun and coaxial accelerator concepts. The four concepts selected for further study are briefly described in this section.

1.5.1 Earth-to-Orbit Rail Launcher

The Earth-to-orbit rail launcher concept was envisioned to supply materials to an orbiting Space Station. These supply items would include hydrogen/oxygen propellants in water form, life support consumables, spares, and miscellaneous materials.

Projectile systems and certain payloads would be fabricated and transported to the launch site, where they would be placed in storage until launch time. The required number of projectiles (with integrated payloads) would be transported daily to the launcher. The projectile would be placed in the preboost section of the launcher and then launched. The muzzle velocity would be 6.85 km/s, with an additional 2.1 km/s required for orbital insertion supplied by the projectile propulsion system at 500 km altitude.

The launcher would be inclined along a mountain side at 20 degrees to the horizontal, to optimize atmospheric drag losses and orbit-insertion propulsion requirements. The launcher tube would be surrounded by 3600 homopolar generator/inductor units providing energy storage. The power would be provided by a nuclear power plant. The launcher tube would be partially imbedded in a concrete foundation to provide structural support to prevent damage to the bore from the bore stresses.

An overview of the Earth-to-orbit rail launcher system is shown in Figure 1-2. Included in the figure are an illustration of the launcher elevation angle, cross-sectional and side views of the launcher system, and a projectile concept drawing. Detailed discussion of this system is provided in Section 4.2 of this report.

1.5.2 Earth-to-Orbit Coaxial Accelerator

The Earth-to-orbit coaxial accelerator was also conceptualized to deliver supply materials to Earth orbit. The payloads would be the

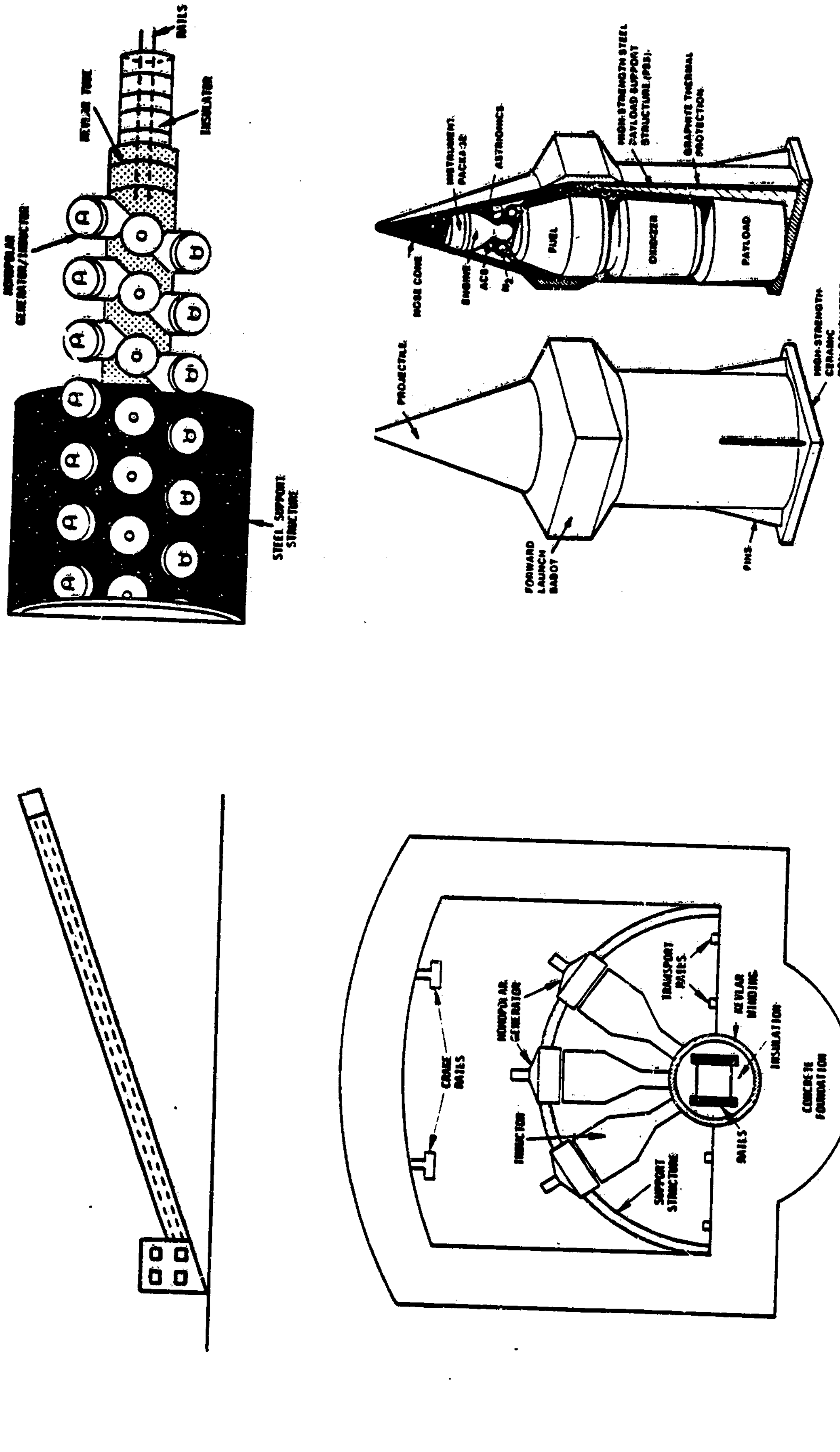


FIGURE 1-2. OVERVIEW OF THE EARTH-TO-ORBIT RAIL LAUNCHER CONCEPT

same as those described in Section 1.5.1. The overall mission scenario is also the same.

The launcher would be built at a 20 degree elevation angle on a mountain side. The launcher tube would be partially imbedded in a concrete foundation. The power would be supplied by a nuclear power plant and stored in a single large Brooks coil inductor. The current is supplied to individual turns of the drive coil.

Figure 1-3 provides an overview of the Earth-to-orbit coaxial system. A cross-sectional view of the launcher system, a projectile concept drawing, and an illustration of the launcher elevation angle are shown in the figure. Section 4.3 of this report discusses this system in more detail.

1.5.3 Hybrid Railgun/Rocket

Payloads would be launched to low-Earth orbit via a hybrid railgun/rocket system. These payloads would be identical to those defined in the Earth-orbital traffic model, including propellants, life support expendables, spares, materials for space_ processing, and miscellaneous items.

The three solid-rocket stages would be manufactured and loaded with propellant prior to delivery to the launch site storage facility. Before launch, the motors would be stacked and the payloads integrated. Projectiles would be transported daily to the launcher. At its scheduled launch time, each projectile would be placed in the breech of the launcher and after the launcher system has been fully charged, the projectile would be launched at 2 km/s. The projectile would continue along its trajectory, through three stage burns, to its destination at low-Earth orbit.

The launcher would be located on a mountain side at a 35-degree angle from the horizontal (no rocket vehicle kick angle is required). Energy storage is provided by 750 homopolar generator/inductor units lined along the 2-km long launcher tube. The power would be provided from commercial utility power plants. The launcher tube structural support is provided by a concrete bed surrounding half of the tube to prevent structural damage to the rails and bore due to launch stresses.

Figure 1-4 provides an overview of the hybrid railgun/rocket system. Cross-sectional and side views are illustrated, as are the projectile concept and launcher elevation angle. Further discussion of this reference concept is given in Section 4.4.

1.5.4 Hybrid Coaxial Accelerator/Rocket

The hybrid coaxial/rocket concept was envisioned to supply the same payloads to low-Earth orbit as the hybrid railgun/rocket. The mission scenario would be the same as well.

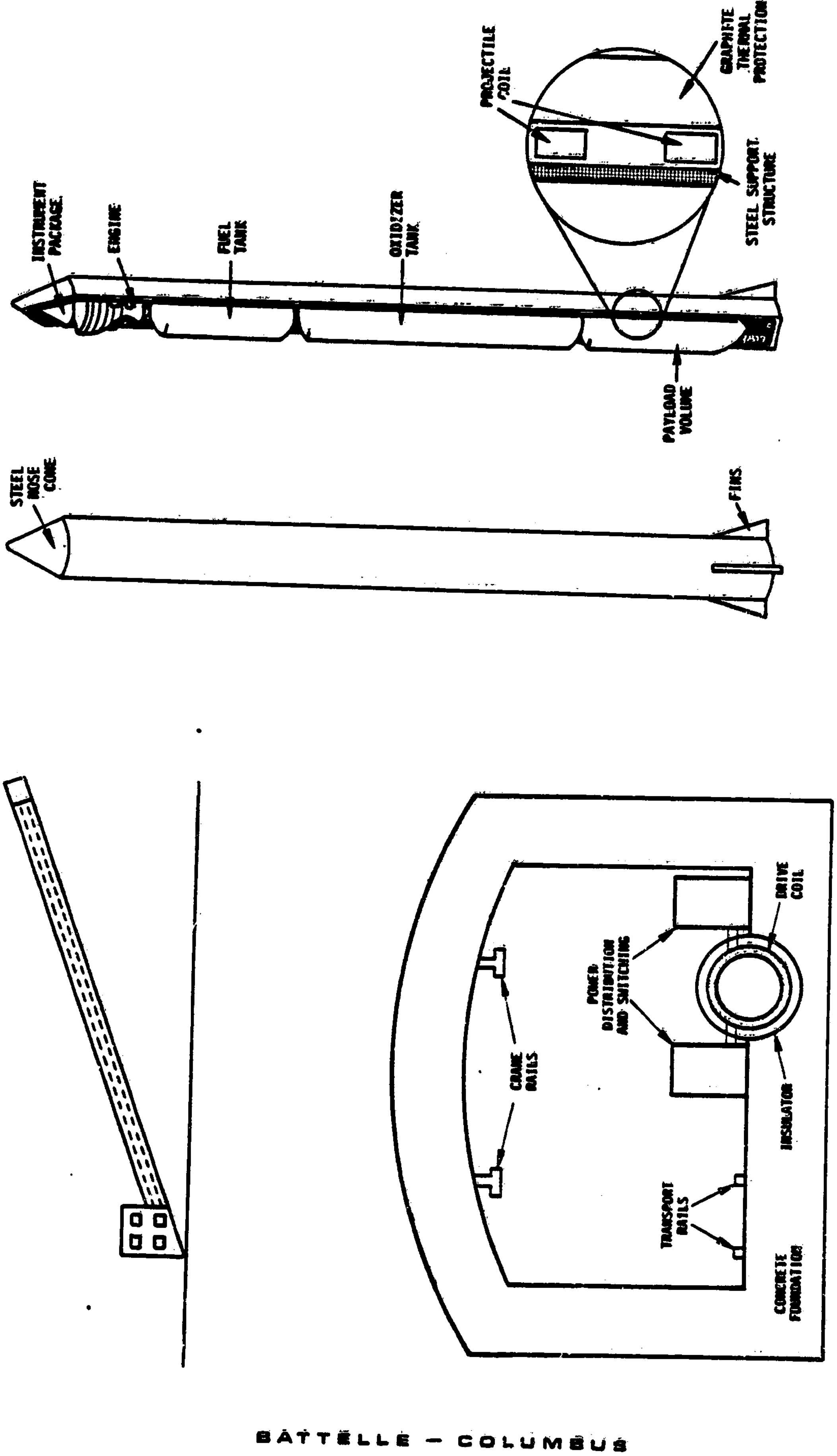


FIGURE 1-3. OVERVIEW OF THE EARTH-TO-ORBIT COAXIAL ACCELERATOR CONCEPT

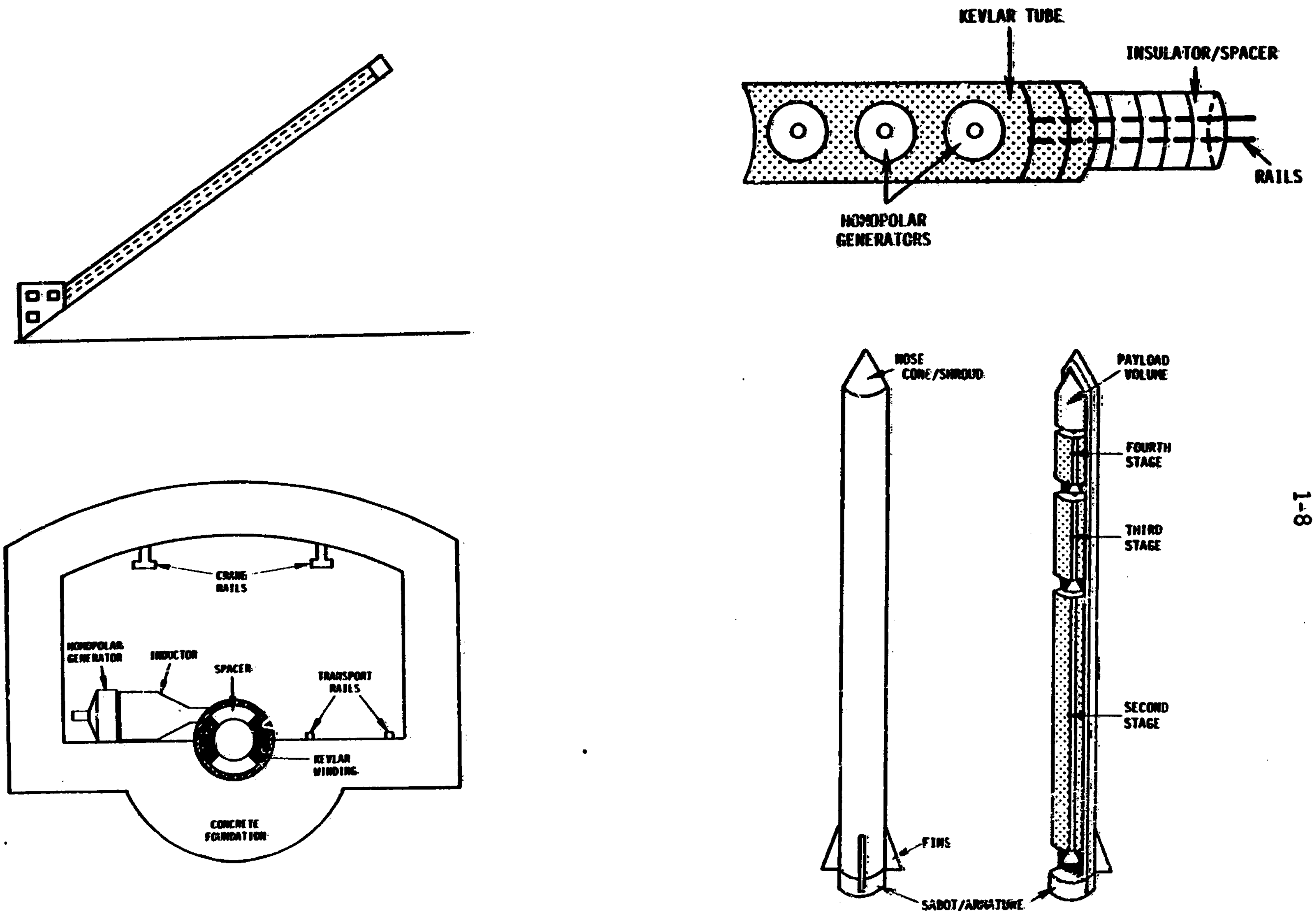


FIGURE 1-4. OVERVIEW OF THE HYBRID RAILGUN/ROCKET CONCEPT

A mountain with a 35-degree elevation angle over a 2-km length is conceptualized. Energy would be provided from a utility power plant and stored in a single large Brooks coil inductor. The launcher tube would be partially imbedded in a concrete foundation to provide structural support to prevent damage to the drive coils from hoop stresses during launch.

An overview of the hybrid coaxial/rocket is shown in Figure 1-5. The launcher cross-section, the projectile conceptualization, and an illustration of the launcher elevation angle are given in the figure. Section 4.5 provides further discussion of the reference concept system.

1.6 Summary of Major Results

The major results of this preliminary assessment of EML space missions are summarized in the following paragraphs.

1.6.1 Alternative EML Concepts

A survey of EML concepts was conducted to identify those which could perform the selected space missions. The open literature (U.S. and foreign) was reviewed and EML experts contacted. Five concepts were identified and reviewed:

- Railgun
- Coaxial accelerator
- Electrothermal thruster
- Electromagnetic rocket gun
- Electromagnetic theta gun.

Of these concepts, two were selected for the reference concepts based upon technical evaluation--railgun and coaxial accelerator.

1.6.2 Mission Models and Requirements

Seven missions were identified for definition in this study and are listed below:

- Earth-orbital launch
- Lunar base supply
- Solar system escape
- Earth escape
- Suborbital launch
- SSTO/TAV boost
- Space-based launch.

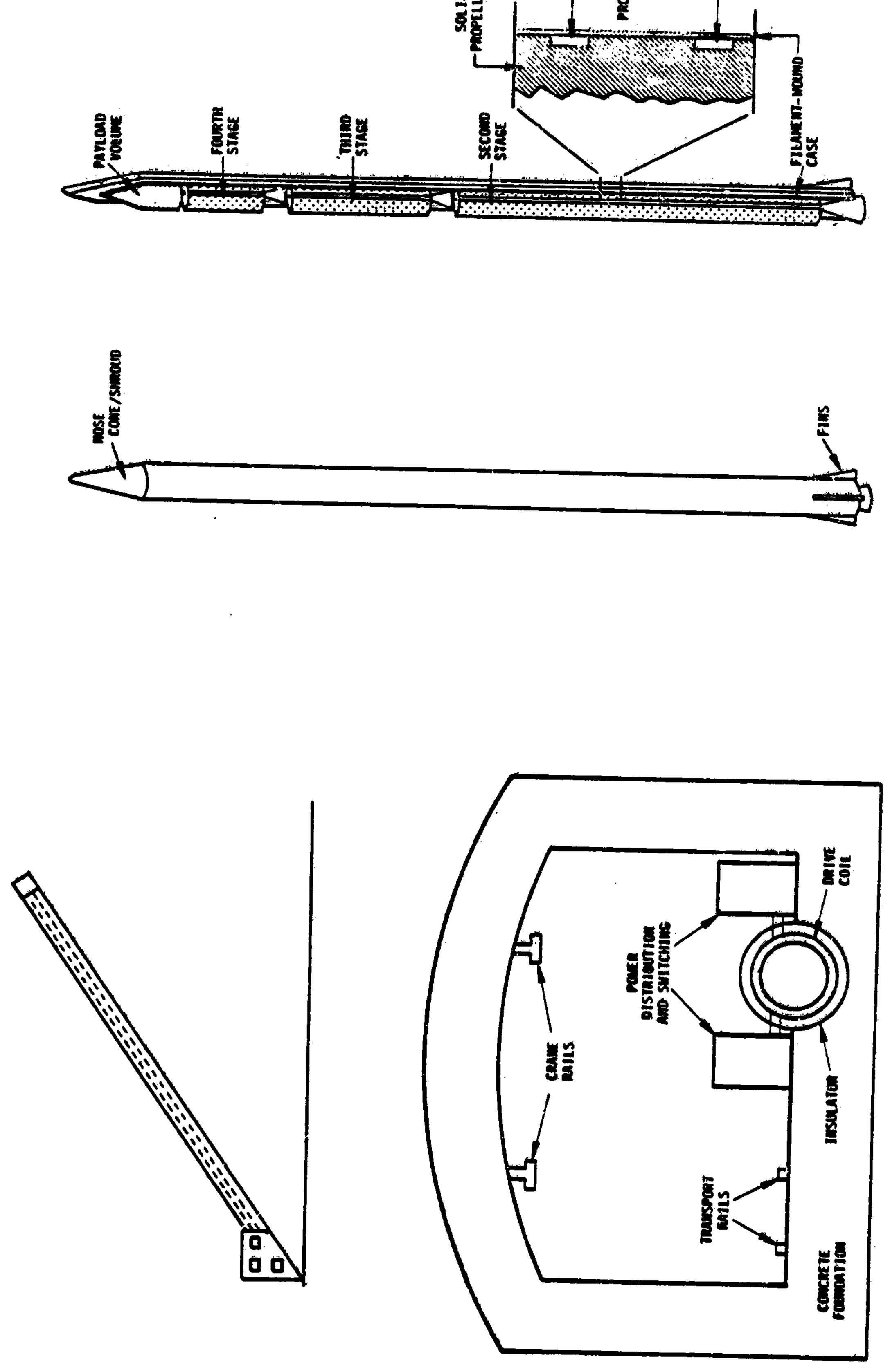


FIGURE 1-5. OVERVIEW OF THE HYBRID COAXIAL/ROCKET CONCEPT

For a particular mission to merit development of an EML system, large amounts of payload delivery are required. The Earth-orbital mission, including Space Station supply, OTV propellant delivery, and raw material supply, had the largest material delivery requirement and was the basis for the reference concepts.

Table 1-1 presents the two Earth-orbital mission models, indicating yearly mass delivery requirements. The low model assumed that there would be 270 persons in orbit in the year 2050; the high model assumed 670 people. Traffic models for the Earth-to-space EML (650 kg maximum payload) and for the hybrid EML/rocket (800 kg maximum payload) are shown in Tables 1-2a and 1-2b. The latter two tables illustrate the daily launch rate for each system between the years 2000 and 2010.

The lunar base supply mission, added to the mission list late in the study, also had large material delivery requirements. Although the costs were not studied in detail, it was felt that an equatorial-based EML system could be a cost-effective method of transporting material to low-Earth orbit, geosynchronous orbit, and the Moon.

1.6.3 Safety and Environmental Impact

A brief safety and environmental impact assessment, evaluating various issues, was conducted. Development and construction of the EML systems are expected to have some environmental effects. Local sonic boom effects are expected; however this is not expected to be a critical issue. Overpressures on the order of supersonic aircraft are expected to occur between 355 and 700 m for the Earth-to-orbit EML and at about 200 m for the hybrid system.

Major accident events for the EML systems are likely to be projectile break-up after launch, liquid-propellant spills, or on-pad fires. Care must be taken to protect workers and the local population from hazardous exposure. The safety risk is expected to be comparable to current space activities if the launch site is selected to avoid overflight of populated areas.

A benefit of using the Earth-to-orbit EML systems may be to reduce adverse environmental impact from the Space Shuttle by reducing the number of Shuttle flights. Reductions would be in the areas of effluent quantities and impact to the ozone layer from Shuttle HCl emissions. Although the hybrid EML/rocket systems would release HCl and Al_2O_3 into the atmosphere during first-stage burn of the three-stage rocket, the emissions would be dispersed and pose less of a hazard than the conventional systems which have a large ground cloud of exhaust. Also, the hybrid systems may add to orbital debris by emitting Al_2O_3 particulate into Earth orbit at Space Station altitudes. Further analysis of a liquid-propulsion system may be desired for the hybrid EML/rocket system.

TABLE 1-1. MASS (MT) OF EARTH-ORBITAL PAYLOADS PER YEAR

Model	Year	Materials for Space Processing (ρ=2.7 g/cc)	OTV Propellants (ρ=1 g/cc)	Life Support (ρ=1.5 g/cc)	Spares (ρ=1.5 g/cc)	Miscellaneous (ρ=1 g/cc)
LOW	2000	15	142	5	1	14
	2005	30	283	7	1	21
	2010	60	340	12	2	35
	2015	118	340	21	3	64
	2020	118	425	29	4	88
	2025	118	425	36	5	100
	2030	118	509	43	6	133
	2035	118	509	51	7	155
	2040	118	594	58	8	177
	2045	118	594	65	9	199
	2050	118	679	72	10	221
HIGH	2000	30	283	5	1	14
	2005	60	396	9	1	28
	2010	118	453	17	2	53
	2015	237	509	35	5	106
	2020	237	566	64	9	194
	2025	237	651	81	11	247
	2030	237	736	113	16	345
	2035	237	849	130	18	398
	2040	237	934	162	22	495
	2045	237	1047	179	25	548
	2050	237	1122	194	27	592

TABLE 1-2a. NUMBER OF EML LAUNCHES PER DAY FOR EARTH-ORBITAL LAUNCHER

Year	Launches Per Day	
	Low Model	High Model
2020	1.4	1.5
2005	2.8	3.9
2010	3.5	5.0
2015	4.1	6.5
2020	5.1	8.0
2025	5.2	9.8
2030	6.3	11.1
2035	6.5	12.6
2040	7.5	14.4
2045	7.7	15.9
2050	8.7	17.0
Fifty-Year Average	5.3	9.6

TABLE 1-2b. NUMBER OF EML LAUNCHES PER DAY FOR HYBRID EML/ROCKET LAUNCHER

Year	Launches Per Day	
	Low Model	High Model
2000	0.7	1.4
2005	1.4	1.8
2010	2.0	2.2
2015	2.6	3.5
2020	2.6	4.2
2025	2.8	4.9
2030	3.2	5.7
2035	3.4	6.5
2040	3.8	7.4
2045	4.0	8.1
2050	4.4	8.7
Fifty-Year Average	2.8	5.0

1.6.4 Cost Estimates

Preliminary cost estimates are given in Table 1-3. The expected costs are shown for each of the four reference concepts. Expected investment costs ranged from \$1.2 B to \$2.2 B, with annual operating costs between \$35 M and \$40 M, not including projectile costs. Section 6.0 discusses the costs in detail.

At high launch rates, the projectile costs dominate capital costs. At low launch rates, the capital costs are not spread over many launches, so the EML system is not as cost effective. When ammortized over a 30-year period, one launch per day results in costs between \$496 and \$757 per kilogram. A launch rate of ten per day corresponds to costs between \$181 and \$234 per kilogram.

Figure 1-6 compares total program costs for the current STS, a conventional four-stage rocket (800-kg payload), and the hybrid EML/rocket system, all for an initial operating capability (IOC) date of 2000. A \$1.5 billion development cost was assumed in this study. With launch rates below two launches per day, the front-end investment causes discounting of cost streams to favor the four-stage rocket for payload delivery. At higher launch rates, the EML front-end investment is recovered and the hybrid system is favored.

Coaxial accelerators have potential for lower costs than railgun systems for several reasons. The projectile stresses are lower because multiple projectile coils distribute the acceleration loads throughout the projectile. This means that structural masses may be less. The launch tube hoop stresses are also lower which correlate to reduced tube structural masses. For the 2000 IOC, a single Brooks coil was assumed for the coaxial accelerator concepts as opposed to multiple homopolar generators and inductors for the railgun systems. The single energy store would be less expensive than the multiple stores.

1.6.5 Technology Requirements

EML technology needs to advance to further define the reference concepts and to improve the cost estimates. Areas of needed technology development include system scale-up, switching and energy distribution, energy storage, brushes, projectiles, and structural support. Battelle has recommended that NASA conduct a supporting research and technology program in experimental research and system studies to further evaluate the potential benefits of the EML for space mission applications.

1.7 Conclusions

Based upon this preliminary assessment, Battelle concludes that electromagnetic launchers appear to be technically feasible and economically beneficial in supplying material to space. However, large-scale EML development can be justified only when large amounts of material are launched. There appears to be no near-term (2000-2010)

TABLE 1-3. EXPECTED COSTS ($, M, 1981)

Cost Category	Earth-Orbital Launcher		Hybrid EML/Rocket	
	Coaxial	Railgun	Coaxial	Railgun
Research & Design	466.0	466.0	300.0	300.0
Development & Investment	1035.0	1518.0	698.0	808.0
Development Test Program	234.0	234.0	234.0	234.0
Total Investment	1735.0	2218.0	1232.0	1342.0
Annual Operations	34.5	40.0	34.5	40.0
Projectile Unit Cost				
--10 per day	0.1211	0.1208	0.1242	0.1422
--1 per day	0.180	0.180	0.190	0.215
	--Payload: 650 kg--		--Payload: 800 kg--	
Annualized costs with 30-year Amortization				
--10 Launches/Day	534.3	554.8	528.9	603.7
$ per kg	225.0	234.0	181.0	216.0
--1 Launch/Day	158.0	179.6	144.9	163.1
$ per kg	666.0	757.0	496.0	558.8

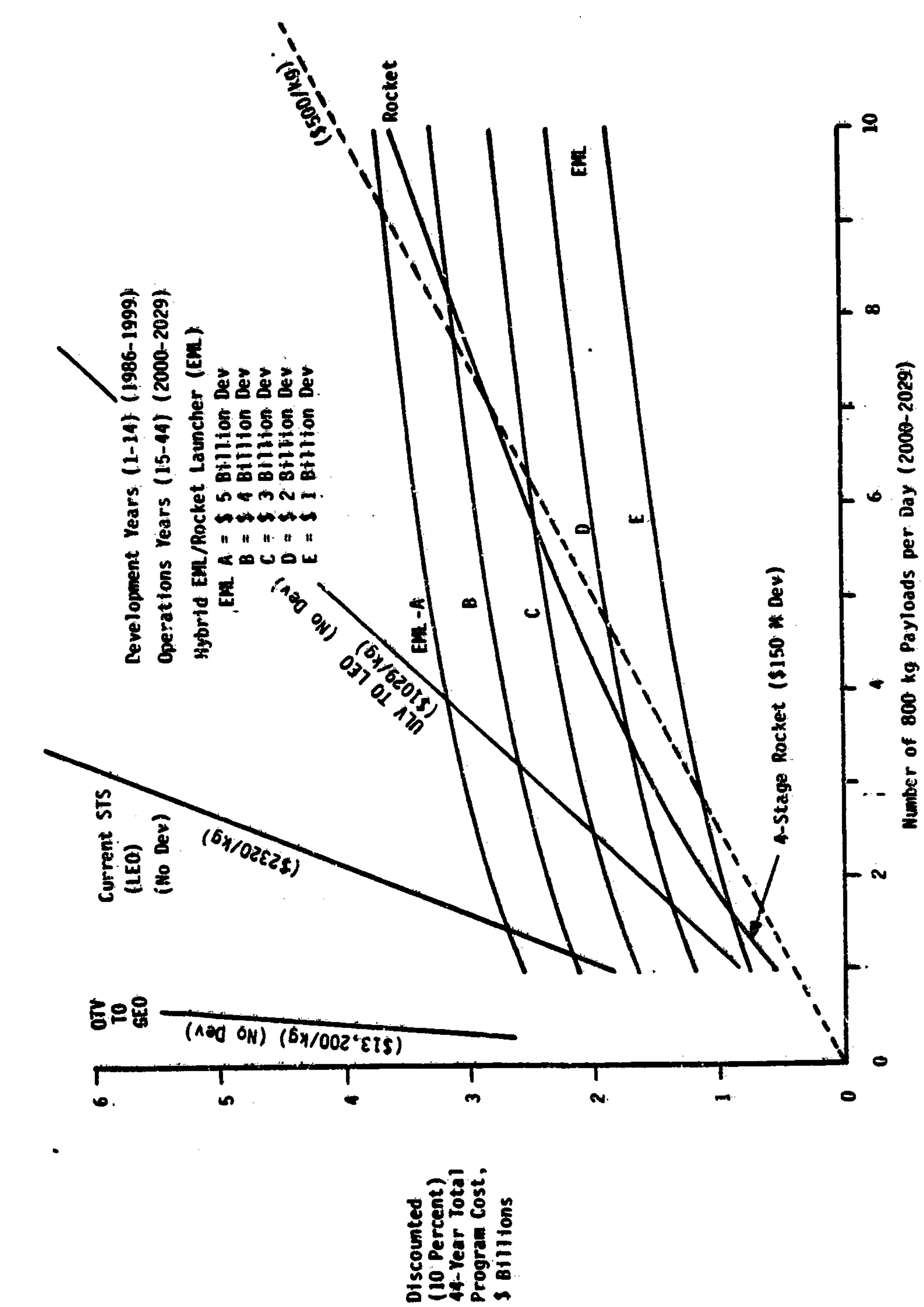


FIGURE 1-6. COMPARISON OF TOTAL PROGRAM COSTS

cost-effective civilian space application for EML systems, but projections of future traffic to space stations and to lunar bases indicate that there is justification for EML development beyond 2010.

1.8 Recommendations

Because of the potential long-term economic benefits of electromagnetic launchers for space applications, Battelle recommends that NASA continue investigations in areas related to these space missions and continue to track technology developments by other government agencies. The NASA investigations should include further analysis of EMLs for lunar base applications. Detailed system and projectile designs are required, as are Space Station studies regarding EML logistics, such as Orbital Maneuvering Vehicle operations, automated rendezvous and docking, and handling and storage issues.

2.0 MISSION MODEL DEVELOPMENT

This section presents the mission models and describes model development for the seven identified electromagnetic launcher (EML) space missions. These missions are:

- Earth Orbital
- Lunar Base Supply
- Solar System Escape
- Earth Escape
- Suborbital
- Electromagnetic Boost
- Space-Based EML.

The mission models include projections of the type of material to be launched, the amount of mass, and number of launches per year for each of the missions. The Earth-orbital mission model spans from 2000 to 2050. The lunar base supply mission model begins in 2010. The timeframe for the other missions under consideration lasts thirty years, beginning in the year 2020.

2.1 Earth Orbital Mission

Several science fiction writers have considered the use of electromagnetic launchers for launching cargo; recently, studies have shown that EMLs are indeed applicable for the cargo-launching missions. These studies show that the high accelerations necessary to maintain reasonable launcher lengths preclude the use of EMLs for transportation of personnel and sensitive equipment. However, for bulk items, including propellants, materials for space processing, and certain food items, EMLs appear to be attractive alternatives to the Space Shuttle and its derivatives.

The Earth-orbital mission model assumes a significant manned presence in space by the year 2020. Table 2-1a indicates the projected personnel growth of an initial Space Station in low-Earth orbit from 1992 through the year 2000. An orbiting Space Station was assumed to be operational in 1992. The initial Space Station would be small and modular with up to eight persons permanently located onboard (ninety-day replacement cycle). By adding additional modules, the Space Station will evolve. In 1995, an increase of four persons was predicted, for a total of twelve persons in orbit. By 2000, sixteen people are assumed to be living and working in a U.S. Space Station. These estimates are based upon the results of the NASA Space Station Task Force Concept Development Group definition and from eight NASA contractors which participated in the Space Station Needs, Attributes, and Architectural Options Study for NASA Headquarters in late 1982 and early 1983. References to the reports of the contractors are listed as follows:

TABLE 2-1a. PROJECTED LEO SPACE STATION PERSONNEL FOR THE YEARS 1991 THROUGH 2000

Year	Number of Personnel
1992	8
1993	8
1994	8
1995	12
1996	12
1997	12
1998	16
1999	16
2000	16

TABLE 2-1b. PROJECTED SPACE STATION PERSONNEL FOR THE YEARS 2000 THROUGH 2050

Year	Number of Space Station Personnel	
	Low Model	High Model
2000	16	16
2005	24	32
2010	40	60
2015	72	120
2020	100	220
2025	125	280
2030	150	390
2035	175	450
2040	200	560
2045	225	620
2050	250	670

- Boeing Aerospace Company (1983)
- General Dynamics Convair Division (1983)
- Grumman Aerospace Corporation (1983)
- Lockheed Missile and Space Company (1983)
- Martin Marietta Aerospace (1983)
- McDonnell Douglas Astronautics Company (1983)
- Rockwell International Corporation (1983)
- TRW (1983).

The projections beyond the year 2000 are categorized in low and high models and are shown in Table 2-1b. The low model assumes that there will be a population of 250 people in Earth orbit by the year 2050; the high model assumes a population of 670 people at this same time. Both models included a small military LEO station and added a manned GEO station. The majority of the personnel in orbit are projected to be in low-Earth orbit in a number of Space Stations.

2.1.1 OTV Propellant Supply

Propellant transportation is a large portion of the EML Earth-to-LEO mission model. The transport of orbital transfer vehicle (OTV) propellants in their LH_2 and LO_2 forms, as well as water, was considered. Many more flights would be required to launch LH_2 and LO_2 (instead of water) because water is 2.89 times denser than an equivalent amount of LH_2/LO_2 propellants. Also, LH_2 and LO_2 tankage is more costly, because cryogenic storage is required. Safety concerns also point to water as the propellant payload of choice. This concept assumes that an electrolysis facility and adequate power would be located onboard the stations. Since the ratio of oxygen to hydrogen in water is 8:1 and most hydrogen/oxygen propulsion systems utilize an oxidizer-to-fuel (O/F) ratio of 6:1, there would likely be an excess of oxygen; however O/F ratios of 8:1 are being considered for propulsion systems. Any excess oxygen delivered to orbit would be used for other station needs, such as for life support or orbital drag make-up.

2.1.1.1 Mission Models

OTV preliminary designs and mission models are widely available and greatly varied. Sources quoted in this section are: Boeing, 1980 and 1983; General Dynamics, 1980 and 1983; and Davis, 1982. All of these sources studied cryogenic systems; recent storable-propellant OTV systems studies are not currently available.

Round-trip (LEO-to-GEO-to-LEO) OTV propellant requirements quoted by the named references range from 15,000 kg to 32,500 kg. In this study, the total propellant requirement for each OTV round trip

TABLE 2-2a. EARLY OTV FLIGHT PROJECTIONS

Year	Number of OTV Flights Per Year			
	NASA/MSFC[a]	Boeing[b]	MDAC[c]	Davis[d]
1991	8	1	4[e]	--
1995	7	6	7	11
2000	14	7	5[f]	20
2005	--	9	--	19
2010	--	--	--	27

Notes: (a) From NASA/MSFC, 1980 (Nominal Model)
(b) From Boeing, 1983
(c) From McDonnell Douglas, 1983
(d) From Davis, 1982
(e) Projection for 1990
(f) Projection for 1999

TABLE 2-2b. OTV FLIGHT PROJECTIONS FOR 2000 THROUGH 2050

Year	Number of OTV Flights[(a)]	
	Low Model	High Model
2000	5	10
2005	10	14
2010	12	16
2015	12	18
2020	15	20
2025	15	23
2030	18	26
2035	18	30
2040	21	33
2045	21	37
2050	24	40

(a)Assumes no lunar base OTV activity.

was assumed to be 22,000 kg (18,855 kg oxygen and 3145 kg hydrogen). If transporting water to be electrolyzed on orbit, this corresponds to 28,300 kg of water required per OTV flight. Assuming losses in transportation, handling, and the electrolysis process, 29,500 kg was used as the baseline for the OTV model.

Tables 2-2a and 2-2b summarize the projected OTV flight schedules. Projections for 1990 through 2010 shown in Table 2-2a are from Boeing (1983), McDonnell Douglas (1983), Davis (1982), and NASA/MSFC (1980). Low and high models are shown in Table 2-2b for the OTV flight projections for the fifty-year period from 2000 to 2050 used for this study. Considered in these tables are the continued use of Earth-orbit for communications, environmental monitoring, etc., and the likely manned traffic between LEO and GEO and between LEO and the Moon.

For an EML propellant supply mission with a launch directly to GEO, propellants carried from LEO to GEO on the OTV for the return delta-V to LEO is avoided. From Section 3.7, with a launch velocity of 11.7 km/s, 430 kg of water could be delivered to a GEO Space Station water holding tank. The water could then be electrolyzed, liquified, and loaded onto the OTV for its return to LEO. The total propellant mass required for the OTV round trip would then be reduced, because the return-trip propellants are not carried as cargo from LEO to GEO.

2.1.1.2 Implications of STS Scavenging

Several studies are currently being funded by NASA to investigate the possibility of scavenging propellants from the Space Shuttle External Tank (ET), Orbiter lines, and tankage in the payload bay. Rockwell is studying payload bay tankage concepts and main propulsion system (MPS) transfer systems for NASA/JSC (Rockwell, 1984), while Martin Marietta is investigating for NASA/MSFC several scavenging concepts using the ET and Aft Cargo Carrier (ACC) tank (Martin Marietta, 1984). NASA estimates cost reductions of $150-$185 M per year through recovery and use of the surplus and residual propellants remaining in the ET and MPS (Gilmore, 1984).

Preliminary results indicate that all OTV missions could be met using two reusable OTVs based at the Space Station with propellants scavenged from the ET and/or surplus tanks in the ACC. Using NASA's nominal Space Shuttle mission model for 1991 to 2000, scavenging from the ET yields 2,901,00 lbs (1316 MT) of available propellants from 204 STS flights. When the ACC is used in conjunction with ET scavenging, 2,276,00 lbs (1032 MT) of propellant is available over 198 flights (Martin Marietta, 1984). The available propellant mass does not account for losses due to boiloff, transfer, and trapped propellants. The mass of the scavenging system hardware must also be taken into account. Nevertheless, the scavenged propellant figures should be sufficient to supply the required OTV propellants.

There are, however, technical issues which must be resolved before an ET propellant scavenging system is put in effect. A preliminary

technical assessment has been performed (Gilmore, 1984). Gilmore indicated a number of areas where special design requirements may be needed:

- Long-term storage
- Zero-g propellant management
- Zero-g propellant gaging
- Zero leak fluid couplings and disconnects.

Multilayer insulation (MLI) was proposed to insulate the storage tanks. Although MLI has been successfully used for many years, Gilmore felt that handling and reuse of the material may be a "challenge". The Cryogenic Fluid Management Facility (CFMF) is a self-contained test package which will be flown in a series of tests in 1988 to demonstrate LH_2 handling and transfer. Propellant management techniques using the surface tension of the fluids are now in use in the RCS and OMS tanks of the Space Shuttle Orbiter (zero-g conditions, but using storable propellants). Critical technology areas cited by Gilmore were zero-g propellant gaging and "foolproof" zero leak fluid couplings and disconnects. These are areas in which the technology has to advance before the propellant scavenging system could be built. These are not considered to be as technically challenging, however, as those issues which face an EML supply system (demonstration of scale-up, etc.).

Another issue which NASA will face if a decision is made to go ahead with a propellant scavenging system is the question of who will pay to transport the propellants to orbit. It has been stated that "the cost of putting these propellants in near orbit has already been paid" (Gilmore, 1984). However, the people paying for the Shuttle launch may not wish to pay to transport someone else's propellants.

When the EML supply of OTV propellants to a Space Station is eliminated, the near-term mission models are severely reduced. OTV propellants represent the primary mission of a 2000 IOC EML system, and, as such, would appear to be the concept driver.

2.1.2 Space Station Supply

Transportation of certain types of non-sensitive payloads for use on Space Stations is another portion of the mission model. The payloads might include food, oxygen and nitrogen for life support systems, spares, and miscellaneous supply items. The requirements are discussed in this section.

2.1.2.1 Life Support

Partially-closed environmental control and life support systems (ECLSS) are envisioned to be on the Space Station. Oxygen and nitrogen must be supplied on a regular basis to support the ECLSS systems. ECLSS

supply requirements were found in a number of sources (primarily Guy, 1982; General Dynamics, 1983; Rockwell, 1983; and Hamilton Standard, 1983). There was a wide variation in the estimates for required ECLSS propellant masses presented in these documents, ranging from 640 kg/yr (8-man station) to 40,000 kg/yr (12-man station) for partially-closed systems. For this study, the ECLSS mass requirement is approximated at 125 kg/person/year between the years 2020 and 2050.

Food requirements for Space Station personnel were estimated from Rockwell (1983) and Carlisle and Romero (1982) with masses of 1.18 and 0.62 kg/person/day, respectively. Averaging the requirement values from these two sources yields 0.9 kg/person/day (food only, does not include packaging). It was felt that only about half of the required food could be launched via the high accelerations of an EML launch (100 to 1200 g's depending upon method of launch), which led to an estimate of 0.45 kg/person/day for EML launch.

By summing the food and ECLSS requirements, the total life support mass requirement is assumed to be 290 kg/person/year or 0.8 kg/person/day. Realistically, this number should decrease with time. Supporting the predicted increasing number of people on orbit would be prohibitively expensive with only a partially-closed system. It is likely that, before a commitment to orbiting a large number of people is made, a completely closed life support system would be developed. For the purposes of this study, however, the life support resupply requirement (less than 1 kg/person/day) was held constant throughout the 50-year period of study.

2.1.2.2 Spares

It was assumed that only one-third of any spares required for Space Station maintenance could be launched from an EML, because many of the spares could not withstand the high accelerations of launch. The spares supply requirement was projected to be about 0.11 kg/person/day. This number represents one-third of the spares mass estimated in Rockwell, 1983.

2.1.2.3 ACS and Drag Makeup Propellants

Several concepts are under study for use as attitude control and drag makeup thrusters on the Space Station. These concepts include Earth-storable, hydrogen/oxygen, and resistojet propulsion systems (NASA, 1984). For simplicity, Boeing's Space Operations Center (SOC) study (Boeing, 1982) was used as a reference for orbit maintenance requirements to determine propellant resupply quantities.

The SOC study calculated propellant requirements for orbit maintenance of a full-up SOC (NASA, 1979). To maintain an orbit at 490 km (265 nmi), 2500 lb sec/day impulse is required. This requirement means that 1633 kg (3600 lbs) of monopropellant hydrazine (Isp of 240 to 256 seconds) are needed per year, corresponding to a daily launch

rate of 4.47 kg/day for each SOC in orbit. If a cryogenic (LH_2/LO_2) system were used instead, approximately 1000 kg/yr of these propellants would be required. Resistojet concepts are included in ECLSS recycling, so no resupply may be necessary.

Hydrazine has approximately the same density (1.008 g/cc at room temperature) as water, so 1633 kg/yr represents 5 launches per year for the Earth-orbital launcher. This function was left off the mission model; however it could be performed if an EML system were available.

2.1.2.4 Fuel Cell Makeup Propellants

Should primary fuel cells be used for Space Station power, an EML system could be used to transport the oxygen and hydrogen necessary for operation.

From NASA/LeRC Space Station PIR-18 (February 1983), the reactant consumption of a H_2/O_2 fuel cell was given as 0.42 kg/kWh. The oxygen-to-hydrogen fuel cell reactant ratio is 8.1, which is the same ratio as that occurring in water. For continuous operation at an average of 25 kW power, 91,975 kg of oxygen and hydrogen are required each year (81,756 kg oxygen and 10,219 kg hydrogen). This corresponds to a daily requirement of 252 kg/day (less than one launch per day).

These figures however do not assume the use of regenerative fuel cells. The regen cells include an electrolyzer as part of the system to convert the water produced during the process back to its original LH_2/LO_2 form. Using regenerative fuel cells would severely reduce the amounts of reactants necessary for operation.

Since solar arrays are likely to be selected as the Space Station primary power source, this requirement has been dropped.

2.1.2.5 Miscellaneous Supply Items

The miscellaneous category includes such items as personnel equipment, clothing, hygiene supplies, ship stores, EVA supplies, and maintenance items. The daily requirement was estimated at about 3.22 kg/person/day from Rockwell, 1983. Three-quarters of this figure (2.42 kg/person/day) was used in developing the mission models, because it was felt that some miscellaneous items may not be able to withstand the high accelerations of an EML launch.

2.1.3 Materials for Space Processing

A major function of a Space Station could be materials processing in microgravity. Several experiments are currently in progress and many more are planned. The 1982 ESRL report (Rice, et. al., 1982) projected at least one launch per day (650 kg/day) was necessary to support the materials processing in space activity from 2020 to 2050

with EML launches of non-sensitive materials. The same requirement was used as the high model in this study from 2010 on. Between 2000 and 2010, the requirement increased up to one launch per day. It was felt that during this period, technology development would still be occuring, preceding the full-scale manufacture of products. The low model assumed one-half the high model requirements.

2.1.4 Traffic Models

Based upon the payload requirements set forth in Sections 2.1.1 through 2.1.3, EML traffic models were determined for the Earth-to-Earth-orbit mission. Table 2-3 presents the masses to be launched per year for the various payloads. The estimated payload densities are indicated under each column heading. These densities were used to estimate payload mass per EML launch. In Section 3.1.5.2 of the ESRL report (Rice, et al, 1982), payload masses as a function of payload density are given for the ESRL projectile (650 kg maximum payload). Several payload densities and the corresponding masses are shown below:

Type of Payload	Density (g/cc)	Payload Mass (kg)
Water	1.0	320
Life Support/Spares	1.5	440
Materials Processing	2.7	650

The masses were used to determine the number of flights that are required per year. The resulting traffic models for the Earth-orbital launcher, indicating the number of flights per year for each model category, are shown in Table 2-4. The total daily launch requirement for the Earth-orbital mission is summarized in Table 2-5.

2.2 Lunar Base Supply Mission

A lunar base represents a logical step beyond the placement of an Earth-orbiting Space Station. Much of the technology gained from Space Station development would be directly applicable to the build-up of a lunar base. The base could be used for astronautical and astrophysical observations, life sciences and ecosystems studies, analysis of engineering/industrial resources, and colonization. Additionally, the Moon could be used for military purposes. However, this application was not considered in this study. Currently, NASA/JSC is supporting efforts to analyze the requirements and development approach of a lunar base. The purpose of this mission analysis is not to justify a lunar base nor to specify a lunar base concept, but rather to develop an understanding of typical support required and to determine the generic

TABLE 2-3. MASS (MT) OF EARTH-ORBITAL PAYLOADS PER YEAR

Model	Year	Materials for Space Processing (ρ=2.7 g/cc)	OTV Propellants (ρ=1 g/cc)	Life Support (ρ=1.5 g/cc)	Spares (ρ=1.5 g/cc)	Miscellaneous (ρ=1 g/cc)
LOW	2000	15	142	5	1	14
	2005	30	283	7	1	21
	2010	60	340	12	2	35
	2015	118	340	21	3	64
	2020	118	425	29	4	88
	2025	118	425	36	5	100
	2030	118	509	43	6	133
	2035	118	509	51	7	155
	2040	118	594	58	8	177
	2045	118	594	65	9	199
	2050	118	679	72	10	221
HIGH	2000	30	283	5	1	14
	2005	60	396	9	1	28
	2010	118	453	17	2	53
	2015	237	509	35	5	106
	2020	237	566	64	9	194
	2025	237	651	81	11	247
	2030	237	736	113	16	345
	2035	237	849	130	18	398
	2040	237	934	162	22	495
	2045	237	1047	179	25	548
	2050	237	1122	194	27	592

TABLE 2-4. NUMBER OF FLIGHTS PER YEAR FOR EARTH-ORBITAL LAUNCHER

Model	Year	Materials for Space Processing	OTV Propellants	Life Support	Spares	Miscellaneous
LOW	2000	23	444	11	2	44
	2005	46	884	16	2	66
	2010	91	1063	27	5	109
	2015	182	1063	48	7	200
	2020	182	1328	66	9	275
	2025	182	1328	82	11	313
	2030	182	1591	98	14	416
	2035	182	1591	116	16	484
	2040	182	1856	132	18	553
	2045	182	1856	148	20	622
	2050	182	2122	164	23	691
HIGH	2000	46	884	11	2	44
	2005	92	1238	20	2	88
	2010	182	1416	39	5	166
	2015	365	1591	80	11	331
	2020	365	1769	145	20	606
	2025	365	2034	184	25	772
	2030	365	2300	257	36	1078
	2035	365	2653	295	41	1244
	2040	365	2919	368	50	1547
	2045	365	3272	409	57	1713
	2050	365	3506	441	61	1850

effectiveness of an electromagnetic launcher (EML) in supplying a lunar base. From recent NASA efforts and past studies of lunar utilization, a lunar base characterization, supply estimate, and traffic model are presented.

TABLE 2-5. NUMBER OF EML LAUNCHES PER DAY FROM EARTH-ORBITAL LAUNCHER

Year	Launches Per Day	
	Low Model	High Model
2020	1.4	1.5
2005	2.8	3.9
2010	3.5	5.0
2015	4.1	6.5
2020	5.1	8.0
2025	5.2	9.8
2030	6.3	11.1
2035	6.5	12.6
2040	7.5	14.4
2045	7.7	15.9
2050	8.7	17.0
Fifty-Year Average	5.3	9.6

2.2.1 Lunar Base Characterization

Recent efforts at NASA/JSC propose a five-phase program for lunar base development. The first phase consists of preliminary surface explorations and lunar mapping. Phase 2 represents the initial, temporary manned base with very limited research capability and transportation versatility. Phase 3 consists of a permanently manned base and begins to exploit the lunar resources (especially the extraction of lunar oxygen for propellants). Phase 4 is an advanced, fully operational base including a moderate manufacturing facility, preliminary lunar industrialization, and experimentation of biological life support and self-sufficiency. Finally, Phase 5 consists of a self-sufficient base with an operational Controlled Ecological Life Support System (CELSS) and a growing manufacturing capability. After reaching the fifth phase the lunar base would be fully operational as a transportation node, laboratory, and industrial complex that is nearly independent of Earth supply.

Major systems of the lunar base throughout all phases include the habitat and life support system, power and thermal system, manufacturing and research facilities, lunar transportation and

manipulator system, and Earth-to-Moon transportation systems. The type, amount, and size of the equipment and capabilities would change as the lunar base evolves. Characteristics and possible alternatives of the systems are described below.

2.2.1.1 Life Support Systems

The initial life support system (LSS) will undoubtedly be an extension of the Space Station LSS using mechanical and chemical waste processing systems. Thus, the initial system will be only partially closed, with CO_2 recycling and partial water recovery. As technologies grow, these systems will evolve into the biological and ecological cycles which are more favorable toward self-sufficiency. Table 2-6 depicts typical LSS supply requirements for the different phases of the lunar base. As the LSS matures from Phase 2 to Phase 3, water recycling is likely to become more efficient (assumed here to save 60 percent more water than Phase 2). Further reduction of miscellaneous clothing and materials supply would result from increased recycling capability. The fourth phase lunar base was assumed to have water recycling 90 percent of the initial LSS, miscellaneous materials recycling 10 percent of Phase 2, an initial biological CELSS to provide 50 percent of the food and almost total O_2 recovery. A goal of the lunar LSS is the Closed Ecological Life Support System (CELSS) which consists of plant growth providing CO_2/O_2 exchange and food. The last phase of the lunar base claims self-sufficiency; however, realistically, vitamins and small food supplements may amount to about 3 percent of the initial food resupply. In addition to the growth of the habitat LSS, the extravehicular activity (EVA) will evolve from Shuttle and Space Station capability to support longer duration EVAs on the lunar surface.

TABLE 2-6 LIFE SUPPORT RESUPPLY ESTIMATES REQUIRED FOR LUNAR BASE EVOLUTION (KG/PERSON/DAY)

	Phase				
	1	2	3	4	5
O_2	--	1.00	0.20	--	--
H_2O (drink)	--	0.50	0.20	0.50	--
H_2O (wash)	--	2.60	1.04	0.26	--
Food	--	0.70	0.70	0.35	0.02
Miscellaneous	--	0.20	0.15	0.02	--
TOTAL	--	4.90	2.29	0.68	0.02

A Boeing Aerospace study has estimated that a lunar CELSS that contributed 50 percent of the crew diet would pay off in 5.5 years. The remaining 50 percent would have to be supplied as packaged food. A CELSS supplying 97 percent of the diet (3 percent supplemented by vitamins and condiments) would become economically feasible in seven years (Spaceflight, December, 1983).

2.2.1.2 Power Systems

The power and thermal systems must supply adequate support for both the 14-day cycles of day and night. The temperatures may range from about 400 K to 120 K for these two different environments. Power requirements are estimated at the 100-kW level for the initial base and grow by a factor of 10 for each of the subsequent development phases mentioned above (NASA/JSC, 1984). Thus, by the fifth phase where the lunar base is self-sufficient, up to 100 MW of power may be required (NASA/JSC, 1984c). Major uses of power in the latter phases of the lunar base include lunar resource processing (especially lunar liquid oxygen-LLOX) and environmental control of large volumes (for CELSS). Solar, nuclear, and regenerative fuel cell systems have been considered.

The preferred systems for power generation may initially be the solar array with regenerative fuel cell supplement during the night cycle. As the base evolves and power requirements increase to provide power for lunar resource processing (such as LLOX), the higher-capacity continuous nuclear power generators will be much more efficient and effective. In these latter phases, a combination of both solar and nuclear may be the best choice to provide steady and peak power requirements, as well as system redundancy.

2.2.1.3 Lunar Transportation

Transportation requirements on the lunar surface will initially be small as preparation of the operating base will be the primary mission. Equipment may be limited to a short distance transporter for crew mobility and a mechanized shovel to dig and bury habitats. However, as the missions expand, longer distances and longer duration travel will be required. Rockwell estimated an average lunar sortie mission of 45 days (North American Rockwell, 1971). Thus, equipment required in latter phases may also include a long distance transporter with large payload capability and a transportable, temporary habitat for remote operations.

In addition to the actual transportation equipment, the crew would need machined assistance for manipulation of bulk supplies and equipment (such as solar arrays), and for assembly of piece parts into larger systems. The manipulation may be done with crane-type robotic arms that can be adapted with end effectors for grappling, lifting, and digging. These manipulators and devices may be attached to vehicles as appropriate for a specific mission.

2.2.1.4 Earth-to-Moon Transportation

The most costly operational expense of a lunar base is for transportation from the Earth to the Moon. The lunar base may never become totally self-sufficient and would certainly not be close to total self-sufficiency for many years after the initial base. Thus, early in the program, a convenient, inexpensive method for logistics transport would be beneficial. This section briefly identifies characteristics of the Earth-Moon transportation systems.

Three major classes of cargo exist: crew, equipment, and bulk material. Assuming no reduction in crew requirements on the Moon, the flow of crew would always be either predominantly from Earth to Moon or with equal flow in both directions. The flow of equipment would nearly always flow from Earth to Moon with little or none returned to Earth. The initial flow of bulk supply materials such as fuel, food, and other consumables would initially be from Earth to Moon; however, after useful operation of the lunar base is achieved, the flow direction may be equal in both directions or may become Moon-to-Earth dominated. Supply of materials is deferred to discussion in the next section.

Earth-Moon transportation alternatives must operate in three different orbit regimes, namely Earth orbit, Earth-Moon transfer orbit, and the Lunar orbit regimes. Transportation systems exist that treat each regime separately (segmented systems) and that combine two or more regimes (non-segmented systems). Segmented transportation systems treat these regimes with specialized vehicles. A key example of such a system is a Shuttle-OTV-Lunar Lander alternative. The Space Shuttle operates most efficiently between the Earth's surface and LEO, where high thrust is required. The OTV operates most efficiently in orbit, never having to land on a planetary surface. Similarly, the Lunar Lander is optimized as a launcher/lander in a 1/6-g environment. The specialization of vehicles increases transportation efficiency and thus, the mass of payload capability. However, these vehicles must dock and exchange payloads. Although this type of operation seems "natural" with the operational STS, upcoming LEO Space Station, and LEO-to-GEO OTV, three vehicle systems and additional orbiting facilities are required. Many alternatives exist for each of the vehicles including use of expendable launch vehicles (ELVs) versus reusable vehicles. Table 2-7 identifies the major alternative vehicles throughout these regimes.

The non-segmented transportation systems ideally combine all the transportation regimes, but usually only combine two of the three regimes. The EML concept is a primary example of a high-impulse system that can combine the Earth-Moon transfer orbit with either an Earth or lunar launch. Given certain launch constraints, an EML payload launched from Earth may traverse to orbit the Moon or to impact the Moon's surface with some guidance propulsion. Similarly, payloads can be launched from the lunar surface at high velocity to intersect Earth orbit or other orbits. With sufficient onboard propulsion (approximately 3 to 4 km/s), bulk material could be soft landed on the Moon directly

from Earth, or could be placed in an orbit about the Moon from the Earth. However, since very-high accelerations (thousands of g's) occur during launch, payloads are restricted largely to bulk supply material. Thus, the disadvantages of the EML-type transport system are the lack of pointing capability (this may also be overcome with increased onboard propulsion) and the requirement of supplemental transportation for crew and special equipment.

TABLE 2-7. ALTERNATIVES TO VEHICLES IN THE EARTH-MOON TRANSPORT REGIMES

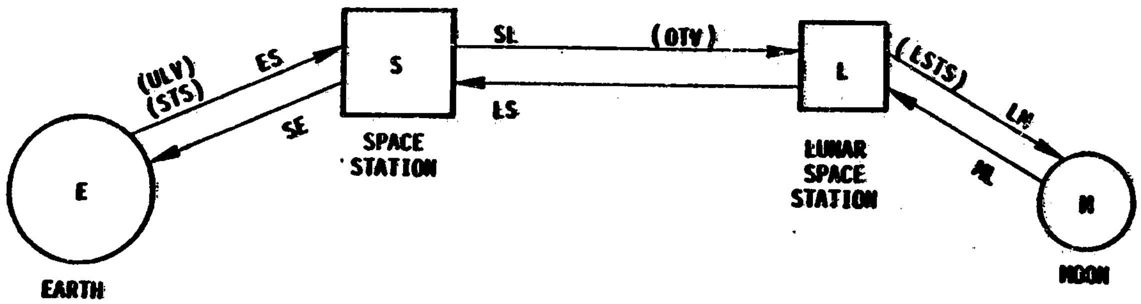
Earth to LEO (Vehicle)[a]	LEO-Lunar Orbit (Propellant Type)	Lunar Orbit-Lunar Surface (Propellant Type)
Shuttle	LH_2-LOX (OTV)	LH_2-LOX
ELVs	Storable Propellants	Storable Propellants
HLLV/ULV	Lunar-Derived Propellants[b]	Lunar-Derived Propellants[b]
EML	Nuclear-Ion Drive	
Hybrid EML	Solar Electric	

(a) ELV = Expendable Launch Vehicle; HLLV = Heavy-Lift Launch Vehicle; ULV = Unmanned Launch Vehicle.
(b) Lunar-derived propellants include O_2/SiH_4, O_2Al powder, and O_2 thermal propulsion systems all using lunar-processed oxygen. (presented in NASA/JSC, 1984)

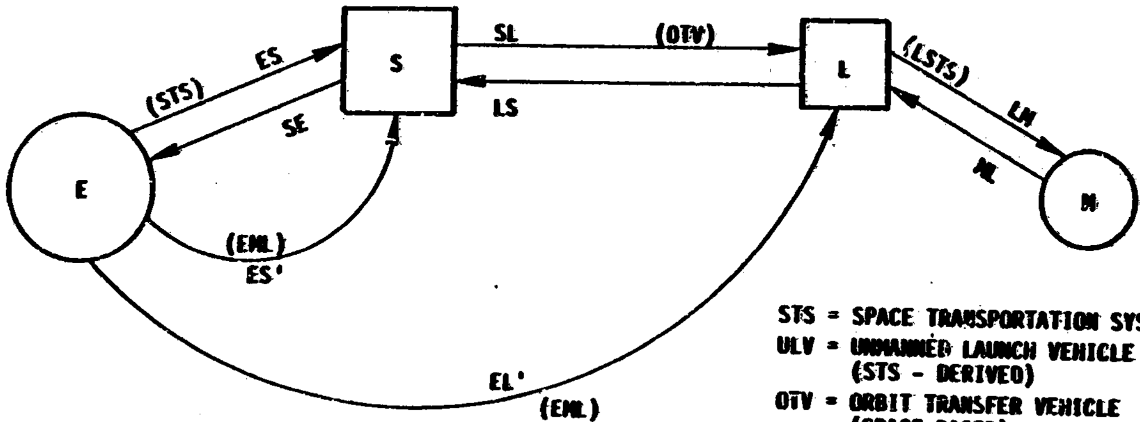
2.2.2 Lunar Base Supply Scenarios

By the year 2010, a permanently-manned lunar base may be operational and would require supplies from Earth such as crew, life support consumables, propellants, equipment, and bulk materials. Of these supplies, an EML could supply the life support consumables, propellants, and bulk materials. Crew and most equipment are too sensitive to withstand the high accelerations of launch (2500 to 3600 g's).

Two lunar logistics scenarios (an OTV-based scenario and an EML-based scenario) were developed to provide insight to possible advantages and disadvantages of the use of an EML. Figure 2-1 shows the proposed traffic flows between the Earth (E), Earth space station (S), a lunar space station (L) and the Moon's surface (M). The baseline OTV lunar logistics scenario would use conventional hydrogen/oxygen propulsion between all transportation nodes (STS, Unmanned Launch Vehicle--ULV, OTV, and lunar STS--LSTS). The scenario which used an



BASELINE LUNAR LOGISTICS SCENARIO



STS = SPACE TRANSPORTATION SYSTEM
ULV = UNMANNED LAUNCH VEHICLE (STS - DERIVED)
OTV = ORBIT TRANSFER VEHICLE (SPACE BASED)
LSTS = LUNAR SPACE TRANSPORTATION SYSTEM
EML = ELECTROMAGNETIC LAUNCHER

EML-BASED LUNAR LOGISTICS SCENARIO

FIGURE 2-1. LUNAR LOGISTICS SCENARIOS

equatorial-based EML to supply materials to the lunar and Earth space stations would reduce the number of STS, ULV, and OTV flights. The materials launched by EML would include life support consumables, propellants, and other materials. As in the Earth orbit supply mission, hydrogen and oxygen propellants would be transported in the form of water to increase payload density, and simplify storage and handling over cryogenic propellants. Propellant issues and tradeoffs are discusssed in Section 2.6. The STS, ULV, OTV, and LSTS systems would still deliver crew, equipment, and other payloads sensitive to high acceleration.

In the baseline lunar logistics scenario, the STS would consist of the partially-reusable Shuttle with 29.5-MT capability to LEO and 15.4-MT return capability. The ULV was assumed to be a Shuttle-derived vehicle using existing STS Solid Rocket Motors, an ET, reusable enginepod, and a cargo volume protected by a large shroud. The ULV was assumed to transport 68 MT of payload to LEO in a single flight. The Earth-orbiting space station would provide a base for OTV refurbishment and refueling, a storage depot for propellants and materials, and a relay station for crew and equipment between the STS/ULV and OTV systems. The OTV was assumed to be a 17-MT cryogenic system, with a 480-second specific impulse and 8-MT payload capability. This assumption is reasonable for the 2010 timeframe and is compatible with current space-based OTV concepts. The lunar orbiting space station may be derived from the Earth space station to provide similar basing, cryogenic storage, and rendezvous capability for the OTV and the LSTS systems. The LSTS would be optimized for lunar surface landing and return to the lunar orbiting station. For compatibility with the OTV, LSTS was assumed to be a 15-MT cryogenic vehicle with a specific impulse of 480 seconds and payload capacity of 8 MT. Both the OTV and LSTS would accommodate manned modules for crew transportation.

The EML-based lunar logistics scenario contains the same four nodes: Earth, Earth space station, lunar space station, and a base on the Moon's surface. However, no ULV flights are included for Earth-to-space station (ES) transportation. The equatorial EML system would accelerate a 500-kg payload at 3600 g's, at a 12-km/s launch velocity to the vicinity of the lunar space station, and would accelerate a 650-kg payload at 2500 g's to the vicinity of the LEO Space Station. For each case, additional on-board propulsion is required for orbit circulation at the stations. The preliminary EML concepts for this mission are discussed in Section 3.1.

2.2.3 Lunar Base Supply Mission Model

Two lunar base timeframes were addressed for each scenario. The first, at year 2010, would consist of a modest base and lunar space station with permament habitation for 12 crew, three of the crew being rotated every 90 days. Logistics for lunar base build-up was not analyzed. The life support system (LSS) was assumed to be a growth of the Earth space station LSS so that food and a portion of consumed

water would be a supply requirement. Propellants for the OTV must be supplied to the Earth space station for the trip to the Moon (by either the ULV or the EML) and to the lunar space station for the return trip (by either the OTV or the EML). Propellants for the LSTS must also be supplied (by either the OTV or the EML) to the lunar space station in addition to equipment needed for build-up of base capability and self-sufficiency (by the OTV only).

A self-sufficient base with closed life support system was assumed for the year 2030. A crew of 48 has been assumed with crew rotation of three persons every 45 days (2-year stay time per person). OTV and LSTS propellants must be supplied, along with a small amount of equipment and materials to maintain the base and supplement lunar resource processing.

The supply categories have been categorized as the crew, life support consumables, equipment, materials (bulk), and OTV and LSTS propellants. Tables 2-8 and 2-9 summarize the total annual lunar base supply requirements for each scenario in the years 2010 and 2030. The derivation of these requirements is discussed below.

2.2.3.1 Crew Supply Requirements During Transport

Crew rotations must be accomplished with conventional space transportation because of the high accelerations of EMLs. A five-day trip to lunar orbit, a one-day changeout in lunar orbit, and a five-day return trip were assumed for a crew of three. The STS would not require additional equipment for manned transport. However, a 5.1-MT manned module for the OTV and a 2.4-MT manned module for the LSTS would be required to provide life support and a habitat for three crew members (Roberts, 1984). Each person was estimated to weigh 136 kg, including clothing and peripherals. For the year 2010, four rotations were assumed each year, giving each crew member a one-year stay on the Moon. During the transit, life support consumables would be approximately 4.9 kg per man per day (see Table 2-6).

For the year 2030, eight rotations of three crew members would give each person a two-year duty cycle. The same manned modules were assumed; however, the transport consumables would likely be reduced to approximately 0.7 kg/person/day (see Table 2-6).

2.2.3.2 Lunar Life Support Consumables

The life support system of the initial lunar base would most likely be an extension of the Earth space station having a closed CO_2 cycle and partially-closed water cycle. Table 2-7 lists the estimates of life support requirements for the various phases of the lunar base. The year-2010 capability would be best represented by Phase 3 at 2.3 kg/person/day for food, water, and miscellaneous supplies. Total annual life support for a crew of 12 would then be 10.1 MT/yr. The

TABLE 2-8. ANNUAL LUNAR BASE SUPPLY REQUIREMENTS (MT/YR) IN 2010[a]

Transport Regime[b]	Baseline Lunar Logistics Scenario	EML-Based Lunar Logistics Scenario
ES	1444.7	10.0
SE	1.6	1.6
SL	304.8	31.0
LS	22.7	22.7
LM	78.3	29.3
ML	11.3	11.3
ES'	--	162.0
EL'	--	136.4

(a)Individual supply items are discussed in Section 2.2.3.
(b)Transport regime acronyms are defined in Figure 2-1.

TABLE 2-9. ANNUAL LUNAR BASE SUPPLY REQUIREMENTS (MT/YR) IN 2030[a]

Transport Regime[b]	Baseline Lunar Logistics Scenario	EML-Based Lunar Logistics Scenario
ES	2450.8	7.4
SE	3.6	3.6
SL	532.2	48.8
LS	48.8	48.8
LM	127.8	26.4
ML	26.4	26.4
ES'	--	309.3
EL'	--	241.2

(a)Individual supply items are discussed in Section 2.2.3.
(b)Transport regime acronyms are defined in Figure 2-1.

year-2030 capability (Phase 5) was assumed to have a nearly closed life support system with negligible supply requirements (see Table 2-6).

2.2.3.3 Equipment

Equipment supplied to the lunar base would support maintenance and growth of various systems and capabilities. NASA has estimated that approximately 40 MT of equipment would be required to develop an advanced base from the initial, permanently-manned base (NASA/JSC, 1984c). Assuming equipment would be transported to achieve an advanced base in five years from 2010, the annual supply of equipment would be 8 MT/yr. Spares must also be delivered; however, the initial amount of spares would likely be small and increase as equipment ages. Thus, 8 MT/yr has been assumed to be constant throughout the 2010 to 2030 timeframe. Beyond the 2030 timeframe, a simple estimate of 4.0 MT/yr was assumed based on the likelihood of increased automation and larger tasks on the Moon.

2.2.3.4 Materials

Bulk material supply to the Moon may include metals, gases, polymers, and organic materials that may be processed on the Moon or lunar space station or may be combined with lunar resources to create useful (and possibly marketable) products. Lunar resource materials may become valuable in LEO so that material transfer from the Moon to Earth orbit is a possibility. Such a production process was not seen to be operational in the initial 2010 timeframe; thus, no materials are transferred to or from the Moon. However, an advanced lunar base would likely require material transportation. An annual materials supply of about 1 MT may supplement the production of metallic alloys, lunar-derived oxygen, and cement-type products. If materials-processing plants become economical on the Moon, this number could easily increase. Mass flow from the Moon to LEO was estimated at 4.0 MT/yr by 2030.

2.2.3.5 Propellants

Major uses of propellant include the OTVs, operating between Earth orbit, and lunar orbit and the LSTS, operating between lunar orbit and the lunar surface. Lunar surface vehicles, the lunar space station, and regenerative fuel cells consume only a small amount of fuel, and were considered to be negligible in the overall mission model.

The LSTS would deliver payload to the Moon's surface including the propellant required for the return trip. This propellant requirement is independent of the lunar logistics supply scenario. Tables 2-8 and 2-9 show the propellants that must be carried as payload for the years 2010 and 2030 to supply the required crew, consumables, equipment and materials. The LSTS is assumed to be a 15 MT vehicle (dry) capable of delivering 8 MT with cryogenic propulsion system of 480-second specific impulse. The propellants for the LSTS would either be supplied at the lunar space station by the OTV or the EML system a 30 percent packing factor was assumed for propellant delivery in the OTV and ULV/Shuttle.

The OTV incurs about a 4-km/s velocity increment for the trip to the Moon and about 1 km/s with aerobraking capability on the return trip. Generally, the propellant weight is 60-70 percent of the total weight of an OTV; thus, a great deal of propellant is required to and from the Moon. For the baseline lunar logistics scenario, the propellant supply at the lunar space station (for OTV and LSTS use) would be transported from the Earth Space Station to the Moon by the OTV. These propellants delivered would have been supplied to the Earth space station by the ULV.

Introducing an EML into the transportation system would relieve the amount of OTV and STS/ULV-delivered propellant required. By transporting water (propellant) to an Earth space station and lunar space station (the OTV, LSTS departure nodes), and electrolyzing it into hydrogen and oxygen and liquifying, the number of OTV flights needed to support the missions is dramatically reduced. Even after taking out the 30-percent packaging factor shown in Tables 2-8a and 2-9a, more than 80 percent of the baseline propellant supply to the Earth space station would be saved using an EML. More than 30 percent of baseline propellant delivery would be saved with EML propellant delivery to a lunar space station.

2.2.4 Traffic Model

Tables 2-10 and 2-11 present the number of flights required for both logistics scenarios (see Figure 2-1). A 100 percent load factor for each vehicle was used, assuming the following maximum payloads:

Shuttle	29.5 MT
ULV	68.0 MT
OTV	8.0 MT
LSTS	8.0 MT
EML	0.5 MT

The tables show the Earth to Earth space station traffic (ES) in Shuttle flights. The Shuttle flights would be required for manned transfer; thus, at least four STS flights would be needed in 2010 and at least eight in 2030. If an Unmanned Launch Vehicle (ULV) were available, 20 and 33 ULV flights may be used the years 2010 and 2030, respectively, to supplement the required manned STS flights.

Introduction of the EML into the traffic model would eliminate the need for a heavy transport vehicle between Earth and the Earth space station (ES). The STS would supply only those flights where acceleration-sensitive payloads must be transported. The EML would also reduce the number of OTV flights by approximately 20 percent. The entire propellant supply could be transported with 577 and 1000 EML launches per year for the years 2010 and 2030, respectively. With the EML-supplied propellant reserve at the Earth and lunar space stations,

TABLE 2-10. PAYLOAD MANIFESTING IN 2010 (NUMBER OF VEHICLE FLIGHTS)(a)

Transport Regime(b)	Baseline Lunar Logistics Scenario	EML-Based Lunar Logistics Scenario
ES SE	48(1.0), 1(0.98) 4(0.03), 45(0)	1(0.29), 3(0.02) 4(0.03)
SL LS	22(1.0), 1(0.20) 4(0.35), 19(0)	3(1.0), 1(0.9) 4(0.71)
LM ML	3(1.0), 1(0.65) 4(0.35)	3(1.0), 1(0.65) 4(0.35)
ES'	--	324(1.0)
EL'	--	253(1.0)

(a)Table assumes vehicles can be loaded to 100 percent of capability. Number in parentheses indicates the fraction of vehicle capability.

(b)Transport regime acronyms are defined in Figure 2-1.

TABLE 2-11. PAYLOAD MANIFESTING IN 2030 (NUMBER OF VEHICLE FLIGHTS)(a)

Transport Regime(b)	Baseline Lunar Logistics Scenario	EML-Based Lunar Logistics Scenario
ES SE	83(1.0), 1(0.07) 8(0.03), 72(0)	1(0.22), 7(0.02) 8(0.03)
SL LS	38(1.0), 1(0.15) 1(1.0), 1(0.9), 6(0.7), 31(0)	1(1.0), 1(0.9), 6(0.7) 1(1.0), 1(0.9), 6(0.7)
LM ML	1(0.91), 7(0.35) 1(0.85), 7(0.35)	1(0.85), 7(0.35) 1(0.85), 7(0.35)
ES'	--	618(1.0), 1(0.6)
EL'	--	480(1.0), 1(0.4)

(a)Table assumes vehicles can be loaded to 100 percent of capability. Number in parentheses indicates the fraction of vehicle capability.

(b)Transport regime acronyms are defined in Figure 2-1.

the OTV would become more efficient as it would not have to carry return propellants and tankage for that propellant.

2.2.5 EML Lunar Base Supply Advantages and Disadvantages

Because an EML system could be an efficient method of transporting water, many advantages exist in supplying water in space for propellant production over conventional supply. However, several disadvantages are also introduced by using the EML.

The major advantage of the EML supply relates to a dedicated propellant-supply system that lessen the STS, ULV and OTV flight and maintenance burdens. The OTV would be lighter and operate more efficiently in delivering the desired cargo (crew and equipment). At the lunar station, the EML-supplied propellant would eliminate the need for lunar-derived propellants (such as oxygen; see Section 2.2.6). Also, the EML projectiles may be taken to the Moon's surface and used as pre-processed raw materials.

The advantage of supplying H/O in the form of water is that large cryogenic storage is reduced, handling is easier, and boil-off losses from dewar storage tanks is nearly eliminated. Cryogenic propellant storage is kept to a minimum because only enough H_2/O_2 to fill the next OTV or LSTS vehicle would need to be available. The disadvantage to this is that electrolysis production equipment must be onboard the space stations. The energy for this production may be obtained from solar or nuclear power. Other disadvantages inherent to the EML system are the launch constraints caused by fixed launcher tubes. Also, the high accelerations do not allow delivery of crew or sensitive equipment. Finally, the EMLs have relatively small payloads, which create additional retrieval and handling burdens at the space stations.

2.2.6 Lunar Base Supply Issues

Many issues and trades have been identified that need to be studied in further detail. Major issues involve the OTV payload capabilities, space station capabilities, type and source of propellants, amount of crew transportation needed, and the amount of materials delivery to and from the Moon. In addition to these issues, EMLs may be placed on the lunar surface and used to supply O_2 to the transportation nodes.

The effects of a larger OTV may reduce the number of OTV flights; however, the propellant requirement will likely remain the same order of magnitude. An OTV sized for payloads larger than the manned module could transport large amounts of bulk material or propellant in addition to the manned module.

The capabilities of the Earth and lunar space stations wi.l affect the economic savings apparent in an EML supply system. The level of on-orbit manpower, facilities, and resources needed to refuel and refurbish OTVs is an important consideration.

The propellant alternatives (especially for the LSTS) are cryogenic and storable. Cryogenic propellants would likely be stored in large dewars and house re-liquification systems. This type of facility could be costly to maintain. An alternative is to use storable propellant systems (such as MMH/NTO), especially for the LSTS, to reduce the storage and handling burdens. However, these propellants are not as efficient (specific impulses at 300-350 seconds) and therefore larger masses of propellants would be required. An alternative to cryogenic storage is to transport water, store it and electrolyze and liquify the hydrogen and oxygen where needed for each flight. Although the electrolysis and liquification equipment add additional cost, the water is easily and safely stored and the power may be provided from a solar or nuclear source.

An alternative to propellant supply from Earth is lunar-derived propellants. Oxygen is very plentiful on the Moon, as it is available in oxides of all major metals. Hydrogen is not abundant. An estimated 5500 m^3 of lunar soil must be processed to obtain 1 MT of hydrogen (hydrogen is trapped in the soil from the solar wind). Hydrogen would have to be supplied if large quantities of propellant are needed. Increasing the oxidizer-to-fuel ratio from 6-to-1 to 8-to-1 may prove advantageous. Alternative propellants have been suggested that use more of the lunar materials and reduce or eliminate the requirement for hydrogen. Propellant systems such as O_2/SiH_4, O_2/Al powder, O_2/Al/HTPB hybrid, and O_2 electric systems are possible, but have lower performance and some production problems (NASA/JSC, 1984c). All of these propulsion alternatives have an effect on the mass of lunar supply.

An EML could be used to supply materials to construct the lunar base, including the lunar space station. Piece parts of various systems may be delivered by EML and then assembled in lunar orbit or taken to the surface.

2.3 Solar System Escape Mission

Launching high-level nuclear waste to solar system escape velocities was Mission A of the Reference Concept in the Battelle 1982 ESRL Study. The study considered complementing the U.S. nuclear waste mined geologic repositories by launching domestic high-level waste (HLW) out of the solar system. Based upon DOE and NASA funded studies, it was determined that the HLW mission would probably not be a driver for ESRL development (Rice, 1982). This disposal approach still left a majority of the nuclear waste to be buried in the repositories. An alternative to this concept was investigated during the current study. By launching HLW and transuranic (TRU) wastes via solar system escape trajectories, the need for mined geologic repositories would be totally eliminated, thus significantly reducing the cost of ground-based disposal systems.

2.3.1 ESRL Mission A Summary

The nuclear waste disposal concept investigated in the ESRL study looked only at launching the high-level waste out of the solar

system, while leaving the lower-level and short-lived high-level (Cs and Sr) nuclear wastes in underground burial sites (see Figure 2-2). This was an effort to compare this method with the so-called "Standard Space Disposal" concept, sponsored by NASA Marshall Space Flight Center. The standard concept disposed of the HLW in a heliocentric orbit midway between Earth and Venus, at 0.85 A.U. In this concept, two spherical nuclear waste packages are launched to low-Earth orbit by a Shuttle-derived vehicle where they are docked with an orbit transfer vehicle and a solar orbit insertion stage. The dual propulsion system combination delivers the nuclear waste to its final orbit around the Sun (see Rice et al, 1982, for further details regarding the standard space disposal concept).

The ESRL study considered launching the HLW using a railgun system (see Figure 2-3). The ESRL Mission A projectile contained 250 kg of nuclear waste in cermet form. After reviewing the availability of high-level nuclear waste for space disposal (Rice, et al, 1982), the traffic model was established, with a requirement of two launches per day to dispose of all the HLW.

2.3.2 Transuranic Waste

Transuranic (TRU) waste is that radioactive waste which is contaminated with transuranic alpha-emitting radionuclides or U-233 at levels greater than (traditionally) 10 nano-Curies per gram (10 nCi/g or 10×10^{-9}Ci/g) of waste. In the early 1970's recognition was given to the higher risk to human health posed by the long-lived alpha-emitting radionuclides and the Atomic Energy Commission began segregating and storing TRU waste for future disposal in a federal repository. Gradually, the commercial shallow-land-burial disposal facilities for radioactive wastes ceased accepting TRU wastes.

The value of 10 nCi/g, was established on the basis that it was similar to the concentration of naturally occurring levels of radium in soil. Recently, promulgated regulations by the NRC will permit burial of wastes up to 100 nCi/g, if properly packaged and buried in licensed burial grounds. This increased activity level will permit more accurate segregation of TRU and non-TRU and will result in some reduction in the volume of TRU waste requiring disposal.

In commercial nuclear power activities, TRU waste is generated only by reprocessing spent fuel and by refabricating the recovered fuel (plutonium and uranium) into new reactor fuel. Transuranic elements do not occur prior to the fissioning of uranium in the power reactor. Defense activities conducted by DOE produce large volumes of TRU wastes which are planned to be disposed of in the Waste Isolation Pilot Plant now under construction in New Mexico.

2.3.2.1 Reprocessing and Refabrication Wastes

During fuel reprocessing, spent fuel rods are chopped into small pieces and the fuel is dissolved from the cladding by an acid

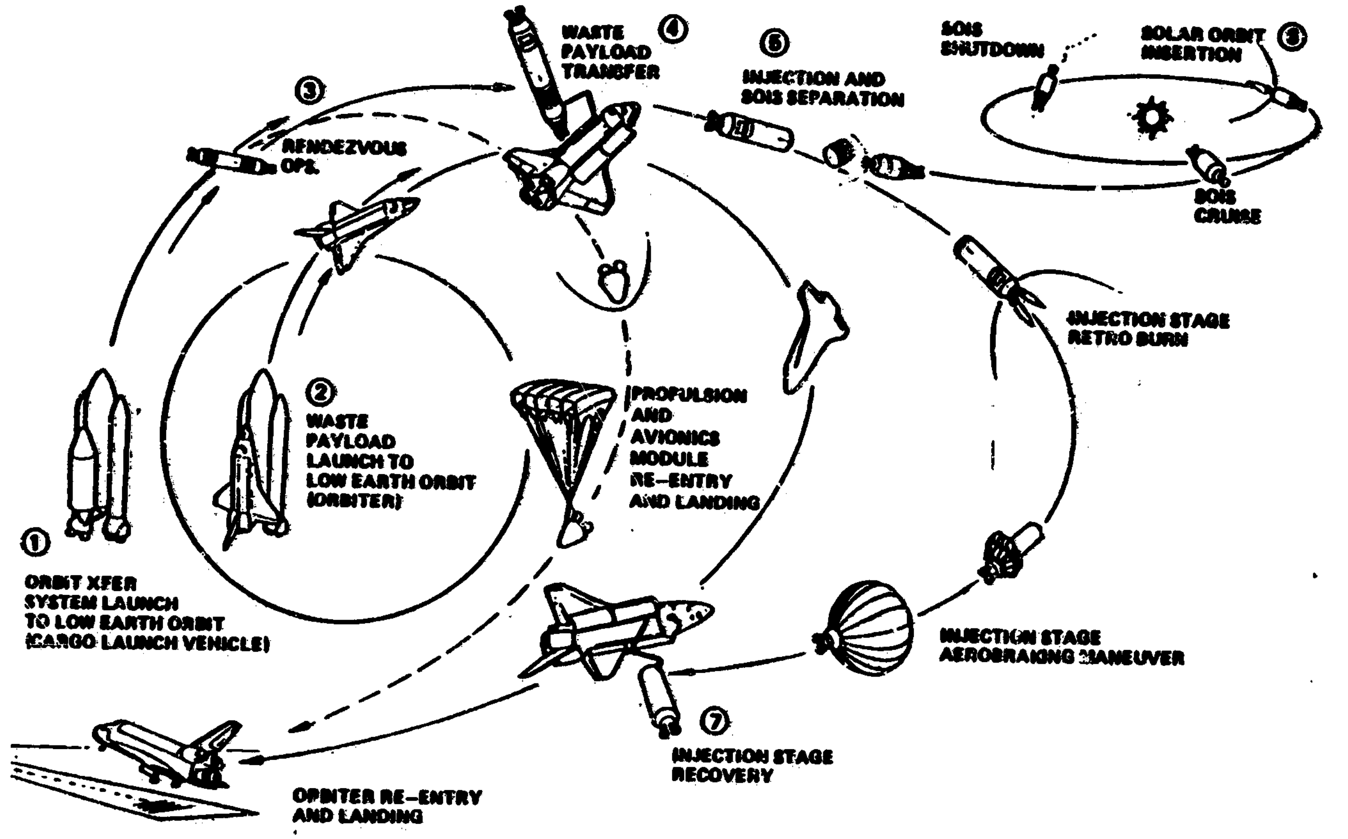


FIGURE 2-2. OVERVIEW OF "STANDARD" NUCLEAR WASTE DISPOSAL IN SPACE MISSION

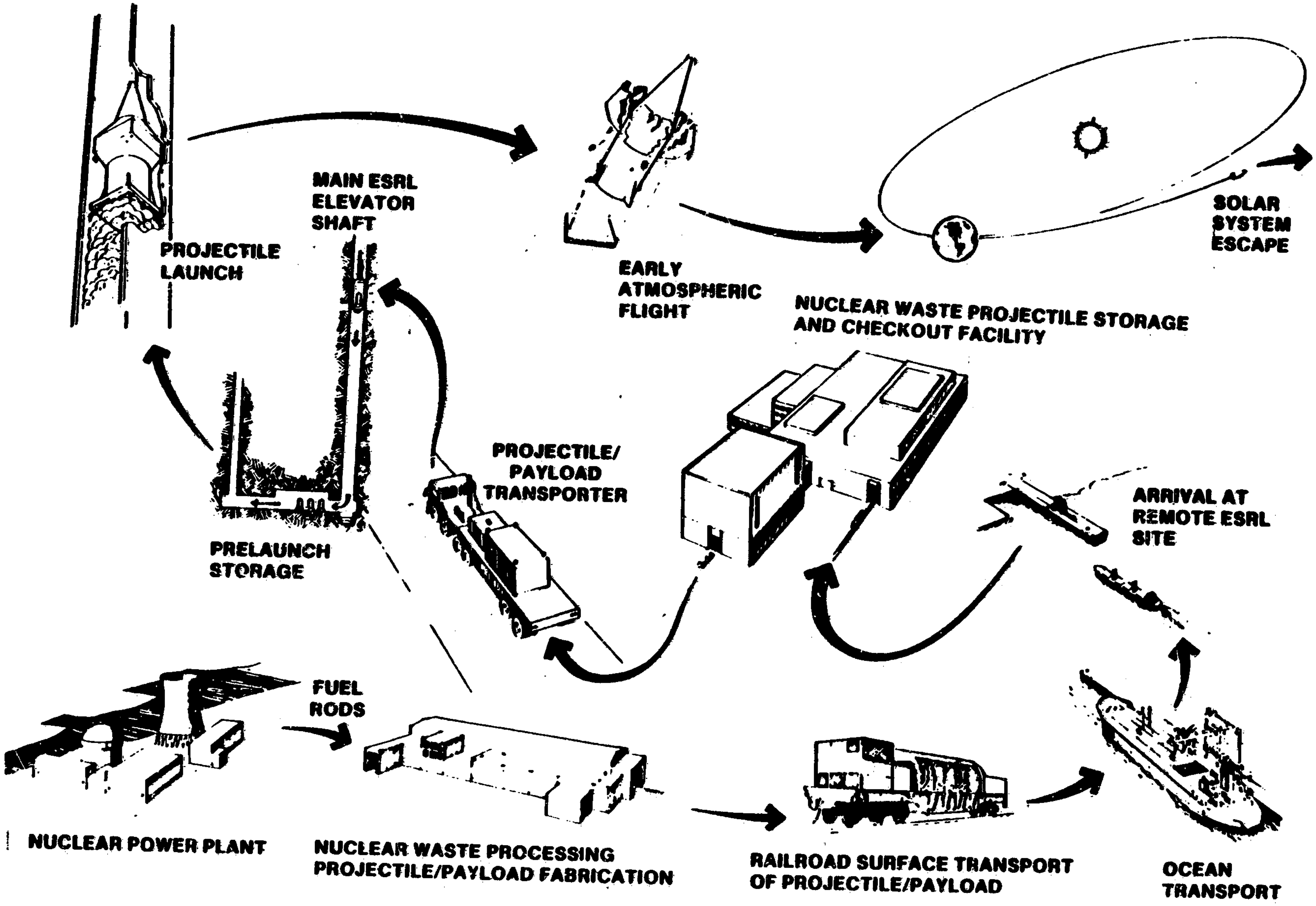


FIGURE 2-3. OVERVIEW OF ESRL NUCLEAR WASTE DISPOSAL IN SPACE MISSION

solution. The uranium and plutonium are extracted from the acid solution by an organic solvent and subsequently separated into individual streams for eventual recovery. The residual acid solution contains the fission products and remaining transuranic elements and is, by definition, high-level liquid waste. This very radioactive waste stream, although a TRU waste, must be treated and disposed of by special regulations being prepared by the NRC. The other TRU wastes generated in fuel reprocessing activities, which are the subject of this analysis, are contaminated with much lower levels of radioactive materials. The TRU wastes from reprocessing fall into the general categories of hulls, filters, process trash, failed equipment, fluorinator ash, ion exchange resins, silica gel, degeraded solvents, and concentrated liquids.

Transuranic wastes in a fuel refabrication plant occur from materials contaminated with plutonium which has been recovered from spent fuel and which is mixed with uranium to produce mixed oxide fuel. The fission product contamination is very low in this waste and packages will have low surface dose rates. There are five major categories of this waste: filters, process trash, failed equipment, process liquids, and incinerator ash. The assumed methods for treatment are similar to those for the same categories of waste from reprocessing.

2.3.2.2 TRU Waste Available for Space Disposal.

The treated waste volumes and masses for each category of TRU waste are given in Table 2-12 expressed per metric ton of heavy metal of fuel reprocessed (MTHMp) or in Table 2-13 as metric ton of heavy metal of fuel refabricated (MTHMf). It is assumed that 1 MTHMf results from 5 MTHMp.

The data are calculated from results in DOE/ET-0028, Technology for Commercial Radioactive Waste Management, the technical basis for the Generic Environmental Impact Statement (GEIS). The waste volumes and masses are taken directly from the reference source which assumed one reprocessing plant operating at a capacity of 2000 MTHM/yr supplying plutonium to one refraction plant operating at 400 MTHM/yr.

2.3.3 Total Mission Summary

The total nuclear waste disposal mission consists of the available TRU waste added to the high-level nuclear waste available for space disposal. The mission summary is presented in two tables. The total annual volumes and masses of HLW and TRU waste for space disposal are given in Tables 2-14 and 2-15 assuming the spent fuel reprocessing projection of Rice, et al, 1982. This projection assumes that fuel reprocessing and refabricating capacity and waste treatment capacity are available to meet the predicted rates.

The traffic model, which indicates the required number of launches per year to dispose of the nuclear waste, is presented in

TABLE 2-12. TRU WASTE FROM FUEL REPROCESSING

Category	m^3/MTHMp	kg/MTHMp
Hulls	0.076	266
Filters	0.040	12
Process Trash		(included in failed equipment and ash)
Failed Equipment (and noncombustible waste)	0.045	225
Fluorinator Ash	0.034	51
Silica Gel		(2.8 included in ash)
Degraded Solvent		(included in ash)
Ion Exchange Resins		(included in ash)
Concentrated Liquid Residue		(included in ash)
Incinerator Ash	0.094	24
TOTAL	0.289	578

TABLE 2-13. TRU WASTE FROM FUEL REFABRICATION

Category	m^3/MTHMf	kg/MTHMf
Filters	0.025	16
Process Trash	Combustibles to incinerator Noncombustibles to failed equip.	
Failed Equipment	0.05	250
Liquid Residue	0.045	89
Incinerator Ash	0.022	5.5
TOTAL	0.142	360
	m^3/MTHMp	kg/MTHMp
Total	0.03	72
Total (both plants)	0.32	650

TABLE 2-14. VOLUME OF U.S. COMMERCIAL WASTE AVAILABLE FOR SPACE DISPOSAL

Year Waste Available	HLW Volume (m^3)	TRU Volume (m^3)	Total Volume (m^3)
1989	43	1885	1928
1990	13	576	589
1991	15	672	687
1992	18	778	796
1993	20	886	906
1994	23	1008	1031
1995	25	1107	1132
1996	26	1206	1232
1997	29	1267	1296
1998	30	1338	1368
1999	32	1389	1421
2000	33	1434	1467
	307	13,546	13,853

TABLE 2-15. MASS OF U.S. COMMERCIAL WASTE AVAILABLE FOR SPACE DISPOSAL

Year Waste Available	HLW Mass (MT)	TRU Mass (MT)	Total Mass (MT)
1989	279	3828	4107
1990	85	1170	1255
1991	100	1365	1465
1992	115	1580	1695
1993	131	1800	1931
1994	149	2048	2197
1995	164	2249	2413
1996	166	2450	2616
1997	188	2574	2762
1998	198	2717	2915
1999	206	2821	3027
2000	212	2912	3124
	1993	27,514	29,507

Table 2-16. Twenty-six launches per day are required to dispose of the HLW and TRU wastes in the years 2020 to 2050.

TABLE 2-16. NUCLEAR WASTE DISPOSAL IN SPACE TRAFFIC MODEL (HLW AND TRU WASTES)

Year	Launches Per Year		
	HLW	TRU	Total
2020	730	8760	9490
2025	730	8760	9490
2030	730	8760	9490
2035	730	8760	9490
2040	730	8760	9490
2045	730	8760	9490
2050	730	8760	9490

It should be noted that the mission model summarized above is based upon a risk assessment done for NASA in 1981-1982 (Rice, Denning, and Friedlander, 1982) which assumed a nuclear power capacity of 200 GWe in the year 2000. Current projections indicate that 164 GWe is a more likely figure (U.S. Department of Energy, 1982), and would reduce the total mission projections to 0.82 (164/200) of the original estimates.

2.4 Earth Escape Mission

The Earth-escape mission model was developed for planetary exploration missions in the thirty-year period, 2020 to 2050. In 1980 NASA appointed a Solar System Exploration Committee (SSEC) to investigate possible planetary missions form 1988 through 2000. The primary purpose of this committee was to recommend a program strategy to revitalize the U.S. planetary science program. Morrison and Hinners (1983) summarize the first task of the SSEC which was to develop a "core program of low- and moderate-priced missions". The core program excludes the larger Viking-type mission which many U.S. scientists feel are high-priority mission, but which the SSEC felt were too expensive to be included in the basic core program. Nevertheless, the SSEC program consisted of 14 missions to be launched between 1988 and 2000, which corresponds to a launch rate of greater than 1 per year. These missions include one lunar, three Mars, two Venus, and three Saturn (including Titan) missions, and four to study the comets and asteroids.

The ESRL Study estimated planetary launches to range from one to four launches per year. The mission model for this study is shown below.

Model	Launches per Year
Low	0.5
Medium	1.0
High	2.5

The low model represents a launch every two years; the medium model, a launch every year; and the high model, five launches every two years. These launch figures are considered to be reasonable projections, especially if the recommendations of the SSEC are followed; that is, to implement the core planetary program and supplement it with other larger important scientific programs.

2.5 Suborbital Mission Model

Electromagnetic launchers could perform a variety of suborbital missions. This section presents the mission model development for the EML suborbital launches. The missions under consideration in this study include atmospheric, astronomy, physics, planetary, and re-entry studies. At the present time, these studies are supported primarily by the NASA Suborbital Program with airborne, balloon, and sounding rocket launches.

2.5.1 NASA Sounding Rocket Program

NASA has used sounding rockets to launch scientific payloads since 1959. The program has grown in size, peaking in the late 1960's and early 1970's. Since then, sounding rocket launches have declined and seem to have stabilized at approximately sixty launches per year. The decline has been caused by a number of factors, namely high inflation rates, limitations in NASA budgets, and increased costs due to larger, heavier, and more sophisticated payloads. The launch history through 1980 is presented in Figure 2-4. The decline in sounding rocket launches is not, however, an indication of lowered demand, as the demand, measured by applications from scientific investigators wanting to fly their experiments on sounding rockets, consistently exceeds the number of available rockets (Teeter and Reynolds, 1982).

Sounding rockets are used for those scientific packages which do not require long viewing times or heavy masses. The average package mass is 205 kg, with a typical viewing time of several minutes, longer for payloads with parachutes. Airplanes and balloons are used for heavier payloads with longer viewing times (up to several days). However, sounding rocket payloads have increased in mass throughout the years, as is shown in Figure 2-5.

2.5.2 Mission Summary

EMLs would likely replace only the sounding rockets for suborbital missions for several reasons. The EML impulsive launch is similar to that of the rockets, in that a ballistic trajectory is

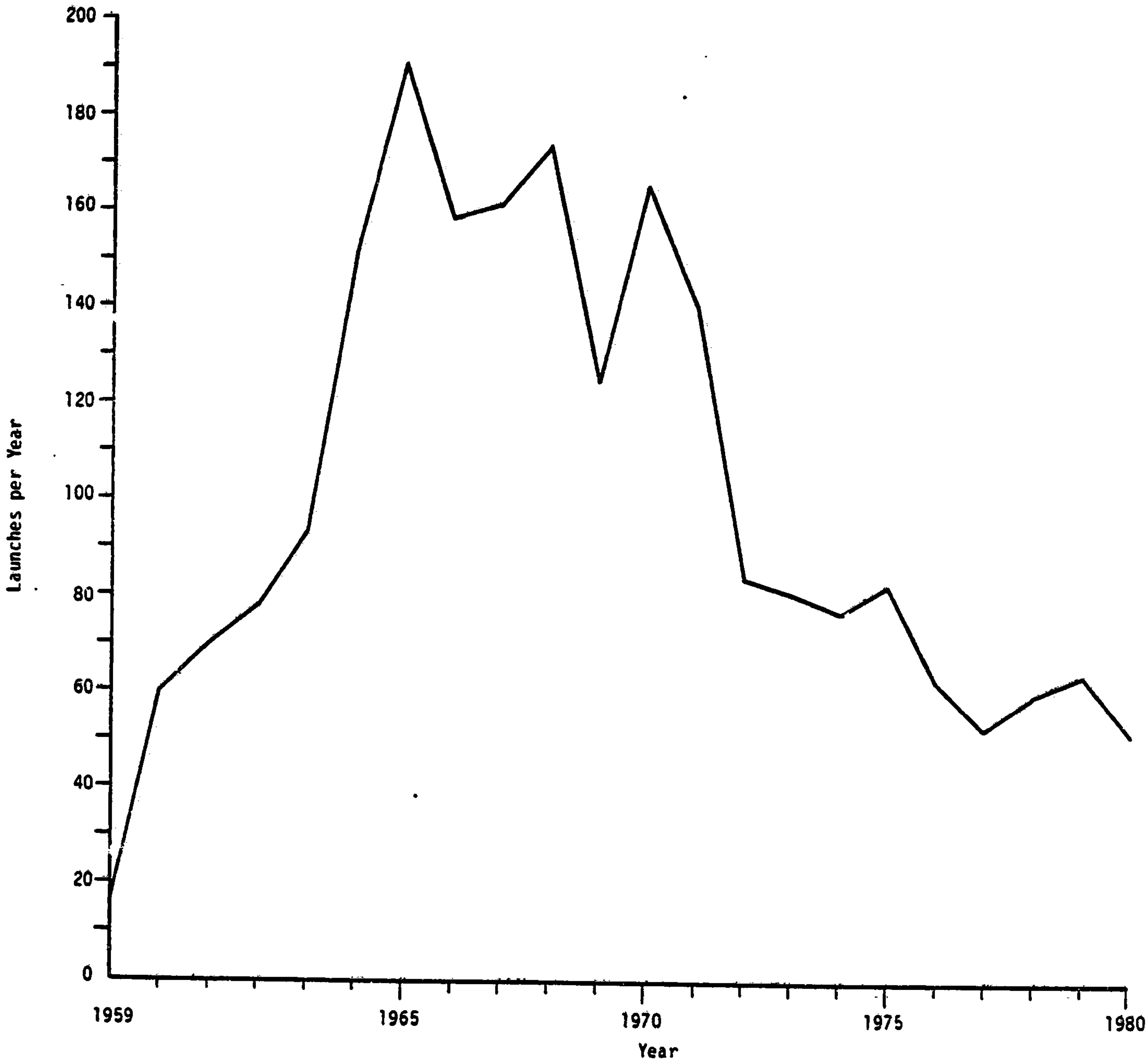


FIGURE 2-4. U.S. SOUNDING ROCKET LAUNCH HISTORY

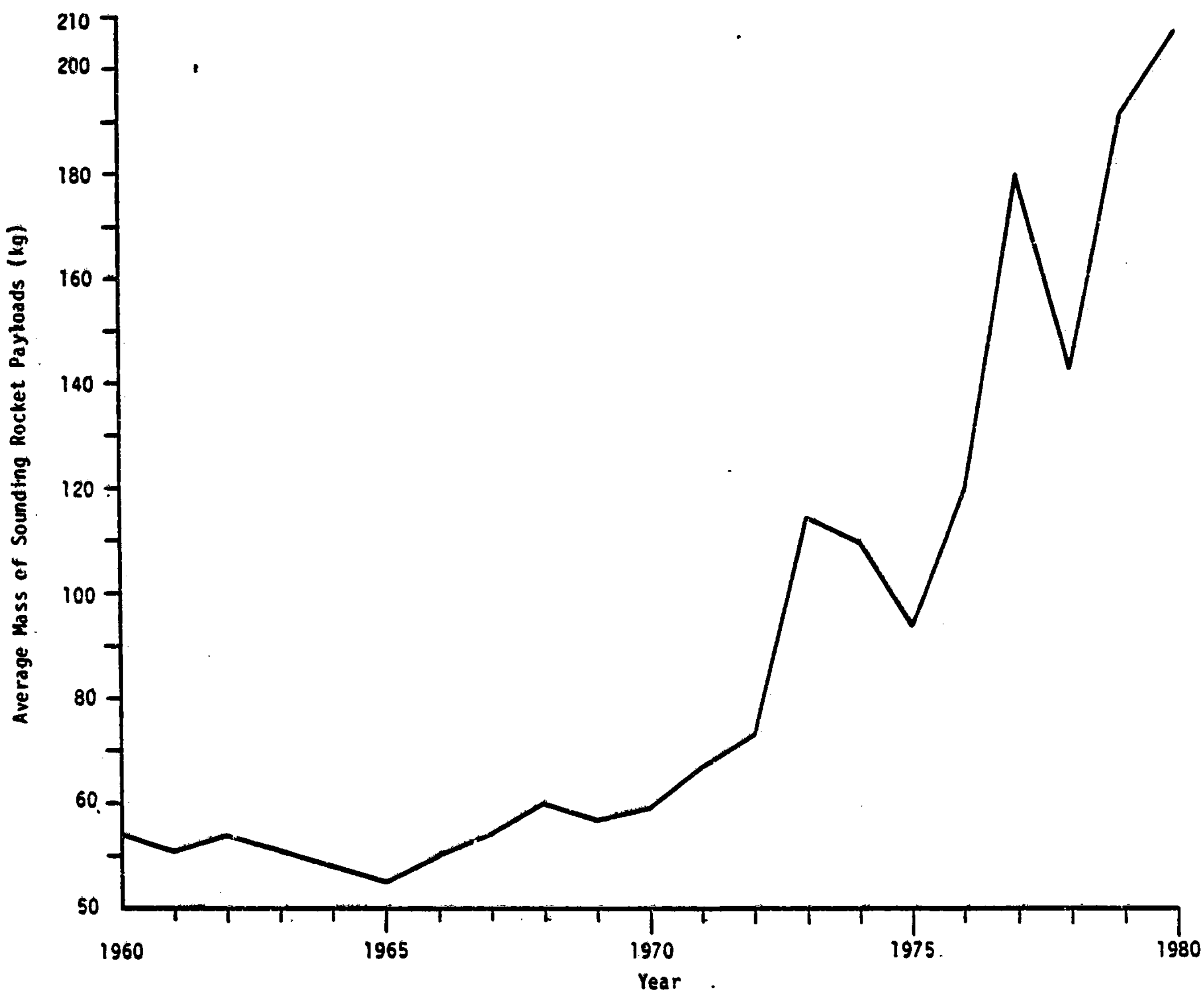


FIGURE 2-5. SOUNDING ROCKET PAYLOAD MASS TRENDS

followed. There is no opportunity for the longer viewing times of aircraft and balloons with an EML launch. The accelerations of launch for sounding rockets and EMLs would be similar, and higher than those experienced by aircraft and balloon payloads.

The suborbital mission model for EML launch during the thirty-year period of 2020 to 2050 is summarized below:

Model	Launches per Year
Low	50
Medium	100
High	150

The projections were considered to be representative of demand for suborbital launches during the timeframe in question. There would of course be some variation depending upon any new discoveries or technology developments which may occur or upon the political climate of the times.

2.6 Electromagnetic Boost Mission

Mission models are presented in this section for two types of EML space applications, where the EML provides the initial boost phase of launch. The first mission is an EML launch of a chemically-propelled rocket to deliver cargo to orbit, the so-called hybrid rocket/EML. This mission, which was originally conceived by Henry Kolm, is a vehicle concept called the Electro-Scout. The second application uses the EML to give an initial velocity to a manned single-stage-to-orbit vehicle (SSTO) or to a manned Transatmospheric Vehicle (TAV), a new vehicle concept currently under study, sponsored by the U.S. Air Force (Rice, et al, 1983).

2.6.1 Hybrid EML/Rocket

Initially, the hybrid EML/rocket was considered only to replace the currently-available small U.S. expendable launch vehicles. These launch vehicles, the Scout in particular, primarily launch small satellites into low Earth orbits. The demand for small satellites is not significant at the present time, and is not expected to increase much in the future. However, if an inexpensive method of launching were available, three to ten launches per year could be expected in the thirty-year period from 2020 to 2050.

Later investigation concluded that a hybrid EML/rocket system could launch the payloads currently studied for the Earth-orbital launcher (e.g., space station supply missions). Table 2-3 indicates the mass available for launching to orbiting Space Stations. A traffic model

indicating number of launches per year for the available payload (compared to the Earth-orbital projectile payload) and is shown in Table 2-17. Table 2-18 reduces the traffic model information to indicate daily launch rates to support the mission model.

TABLE 2-18. NUMBER OF EML LAUNCHES PER DAY FOR HYBRID EML/ROCKET LAUNCHER

Year	Launches Per Day	
	Low Model	High Model
2000	0.7	1.4
2005	1.4	1.8
2010	2.0	2.2
2015	2.6	3.5
2020	2.6	4.2
2025	2.8	4.9
2030	3.2	5.7
2035	3.4	6.5
2040	3.8	7.4
2045	4.0	8.1
2050	4.4	8.7
Fifty-Year Average	2.8	5.0

2.6.2 SSTO/TAV Booster

An EML system was also studied to give a small initial velocity (on the order of 500 m/s) to an airbreathing or rocket system. These sytems might include single-stage-to-orbit (SSTO) vehicles or Transatmospheric Vehicles (TAVs). The systems would be manned, and therefore require very low accelerations.

The vehicles under consideration would, under normal circumstances, have a relatively low launch demand (estimated at a weekly launch rate). Should circumstances change, the vehicles would be required to launch on demand. Therefore, the mission model for TAV/SSTO vehicles was projected for the years 2020 to 2050 as follows:

Model	Launches per Year
Low	50
High	200

In this case, the low model (an average of one launch per week) represents the expected launch rate. However, the system should be designed to withstand daily launches.

TABLE 2-17. NUMBER OF FLIGHTS PER YEAR FOR HYBRID EML/ROCKET LAUNCHER

Model	Year	Materials for Space Processing	OTV Propellants	Life Support	Spares	Miscellaneous
LOW	2000	19	215	6	1	21
	2005	38	429	9	1	32
	2010	75	515	15	3	53
	2015	148	515	26	4	97
	2020	148	644	36	5	133
	2025	148	644	45	6	167
	2030	148	771	54	8	202
	2035	148	771	64	9	235
	2040	148	900	73	10	268
	2045	148	900	81	11	302
	2050	148	1029	90	13	335
HIGH	2000	38	429	6	1	21
	2005	75	600	11	1	42
	2010	148	686	21	3	80
	2015	296	771	44	6	161
	2020	296	858	80	11	294
	2025	296	986	101	14	374
	2030	296	1115	141	20	523
	2035	296	1286	162	23	603
	2040	296	1415	203	28	750
	2045	296	1586	224	31	830
	2050	296	1700	243	34	897

2.7 Space-Based Missions

Two space-based EML systems were considered in this study: a satellite kick system, to transfer satellites from low-Earth orbit to geosynchronous orbit, and a high-level nuclear waste disposal system. The mission model development for each system is discussed in this section.

2.7.1 Satellite Kick System

The NASA Outside Users Payload Model (Battelle, 1983) contains payload schedules for non-NASA, non-DOD reimbursable payloads to be flown by non-Soviet-block countries in 1983 to 1998. Detailed low and high models are given for twelve mission categories, including international, U.S. domestic, and foreign regional communications; U.S. and foreign observations (both geosynchronous and low-Earth orbits); materials processing; and scientific/technical development. Those mission categories which may be serviced by a space-based EML satellite kick system are the communications and GEO observation satellites. The Outside Users Payload Model projections for the years 1983 to 1998 are summarized below.

Category	Low Model	High Model
U.S. Communications	107	143
Foreign Communications	98	157
U.S. GEO Observations	8	11
Foreign GEO Observations	16	27
Total	229	338

The average number of payloads flown per year over the defined sixteen-year period for these mission categories is 14.3 for the low model and 21.1 for the high model. The trends indicated in the model summaries show that communications satellite launch schedules dip in the early 1990s and increase again in the late 1990s. The observations satellite launches appear to remain constant over the indicated timeframe.

By the years 2020 to 2050, communications satellites in GEO should be saturated with satellites placed in modular platforms. Earth observations missions were projected to continue at approximately the same launch rates as currently experienced. Therefore, the mission model for a space-based satellite kick system was projected as follows:

Model	Launches per Year
Low	12
Medium	18
High	24

2.7.2 Space-Based Nuclear Waste Disposal

A combination of and alternative to the "Standard Space Disposal" concept described in Section 3.2.1 and the ESRL Mission A concept is presented here. Small billets of high-level nuclear waste (HLW) were launched from an orbiting electromagnetic launcher system to solar system escape velocities. The HLW billets are delivered in shielded spherical containers to the space-based EML via a Shuttle-derived vehicle. The billets considered here are the same as those studied in Boeing, 1982. Each cylindrical billet (with radius of 2.926 cm and height of 5.858 cm.) has a mass of 1 kg. Therefore, to dispose of all US commercial HLW (Rice, Miller, and Earhart, 1982), 500 launches per day from the space-based system are required to dispose of the high-level waste. Disposal of TRU wastes (Section 2.3) in this manner was not considered economically feasible because of the extremely large number of Space Shuttle flights required.

3.0 EML SYSTEMS ANALYSIS

This section summarizes the results of the systems analysis conducted during the course of this study. Section 3.1 presents the preliminary mission scenarios and system concept sizings based upon each mission payload model. Table 3-1 summarizes the payload requirements for each mission derived from data in Section 2.0. Additional subsections describe other analyses that were performed, including radiation estimates (from nuclear waste payloads), launch windows, flight mechanics, and projectile considerations.

3.1 Preliminary Analysis of Candidate Concepts

Preliminary concept sizings were performed for the missions identified in Section 2.0 and were based upon the payload information shown in Table 3-1. The railgun sizings were done by Battelle in the same format as that done in the previous ESRL study (Rice, et al, 1982). Sizings for the coaxial accelerator concepts were done by EML Research, Inc., under a parallel NASA contract.

3.1.1 Railgun Concepts

Railgun concepts for space mission applications were addressed in the 1982 Battelle-ESRL Study (Rice, et al, 1982), and are only briefly summarized in this section. The simplest railgun consists of two conducting rails which are shorted by a moveable solid metal armature. A magnetic field is produced by the electrical current passing through the rails and armature. The Lorentz force, $F = Nqu \times B = i \times B$, propels a projectile down the bore of the railgun (see Figure 3-1). The solid armature is restricted to operation below 2 or 3 km/s, due to the fundamental heating limits of metals.

A plasma armature may replace the solid armatures for velocities beyond the 2 or 3 km/s limit. The plasma is a good conductor, which supports the high pressures necessary to propel the projectile along the rails. A problem with plasma armatures occurs during the first 1 km/s or so of travel, where the plasma causes erosion of the rails. Currently, this problem is avoided by preboosting; that is, accelerating the projectile to 1 km/s by other means before it enters the railgun proper.

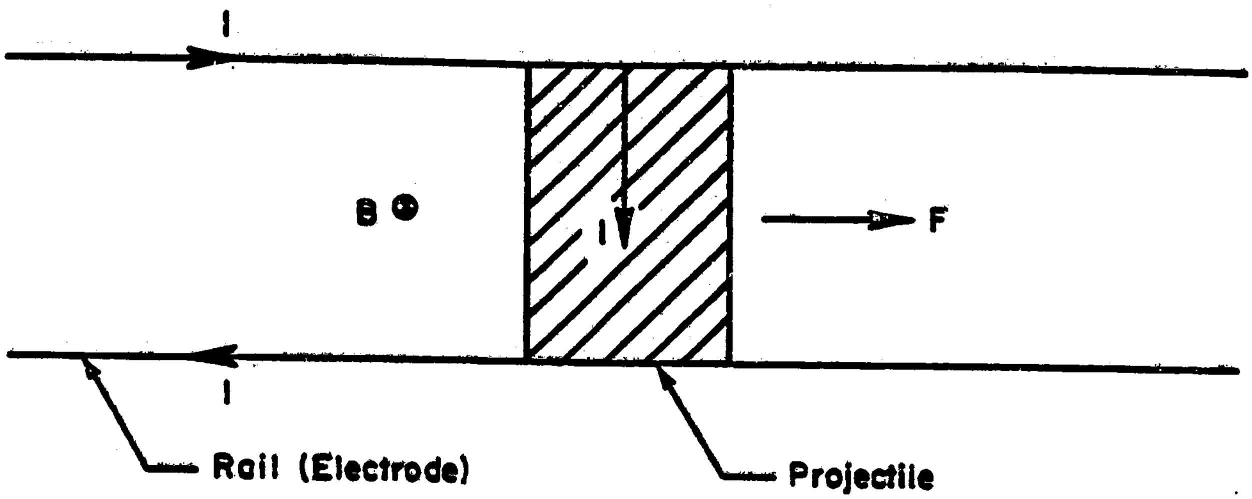
For high velocity launches, a single energy store system may not be appropriate, because longer rail lengths require a very large energy store to maintain the current throughout the launch. Also rail resistance becomes a problem. Distributed energy storage (DES) is a potential solution. Energy stores are distributed along the length of the railgun; energy is switched into the rails as the projectile travels down the bore. Switching becomes a much larger problem for DES railguns than for single-stage systems, where the switching is done at the beginning of the acceleration, where it is the simplest to accomplish.

TABLE 3-1. EML PAYLOAD REQUIREMENTS SUMMARY

Mission	Payload Mass (kg)	Launch Frequency	Acceleration Limit (g)	Velocity (km/s)
GROUND-BASED				
• Earth Orbital System	650	1 - 17/day	2,500	7
• Solar System Escape	250 (HLW)	2/day[a]	10,000	20
	285 (TRU)	24/day[a]	10,000	20
• Earth Escape	600	1 - 3/yr	10,000	12 - 20
• Suborbital	50 - 2,000	50 - 150/yr	2,500	1 - 5
• Electromagnetic Boost				
- Solid Rocket Vehicle	800	1 - 9/day	20 - 100	2
- TAV/SSTO	900,000	50 - 200/yr	5	0.5
• Lunar Supply	500	1140/yr	2,500	12
SPACE-BASED				
• Satellite Kick System	5,000	12 - 24/yr	1,000	Up to 2.5
• Nuclear Waste Disposal	1 (HLW)	500/day[b]	Limited By Launcher	11

Notes: (a) Daily 6 hour launch window centered at dawn (equatorial launch site)

(b) Nineteen-minute launch window on each orbital pass for a launch velocity of 11 km/s (One launch every 36 sec. during window)



Some concern has been expressed that, because of the high voltages required for Earth-to-space railgun operation, the bore would not be able to stand off these voltages. Table 3-2 (excerpted from CRC Press, 1975-1976) lists the spark gap lengths for different peak voltages for the worst case (needle point electrodes). These are the spark gap lengths in air. The table indicates that, when operating at a voltage of 100 kV, the bore of a railgun must be greater than 15.5 cm to avoid arcing between the rails. For the Earth-orbital railgun (Section 4.2) the peak voltage is 65 kV and bore diameter is 1.0 m, which is an order of magnitude above that required to avoid arcing. It should be noted, however, that the environment inside the railgun would be different from that defined for the table; but with the order-of-magnitude difference, no problem is anticipated.

Railgun sizing data for each of the eight missions are summarized in Table 3-3. These concepts are discussed in Sections 3.1.3 through 3.1.10.

3.1.2 Coaxial Electromagnetic Accelerator Concepts

The second type of electromagnetic launcher selected for consideration in this study is the coaxial electromagnetic accelerator. Several coaxial accelerator concepts which could be used to perform space missions are briefly discussed here and in Section 8.0. For more detailed discussions, refer to Appendices D and E and to Mongeau, 1981.

A coaxial electromagnetic accelerator operates by passing a projectile coil through a drive coil. Figure 3-2 illustrates the relationship between projectile coil position and the mutual inductance gradient. The thrust of the launcher is proportional to the mutual inductance gradient and is given by the equation:

$$F = \frac{dM}{dz} I_p I_d$$

TABLE 3-3. RAILGUN CONCEPT SIZING SUMMARY

Mission	Railgun Type	Projectile Mass (kg)	Exit Velocity (km/s)	Length (m)	Bore Area (cm^2)	Force (MN)	Current (MA)	Kinetic Energy at Muzzle (GJ)	Back EMF Voltage (kV)
GROUND-BASED									
• Earth Orbital System	DES	5,900	7	2,040	10,000	71	17	145	60
• Solar System Escape	DES	2,060	20	2,040	2,600	202	28	411	280
• Earth Escape	DES	2,060	20	2,040	2,600	202	28	411	280
• Suborbital	DES	225	5	330	1,060	5.5	4.7	2.8	12
	SS	225	5	490	1,060	5.5	4.7	2.8	12
• Electromagnetic Boost									
-Solid Rocket Vehicle[a]	DES	15,000	2	10,200	11,300	2.9	3.4	30	3
-Solid Rocket Vehicle[b]	DES	15,000	2	2,040	11,300	15	7.7	30	3
-TAV/SSTO	SS	1,030,000	0.5	3,830	90,000	51	14	130	4
• Lunar Base Supply	DES	5,900	12	2,940	10,000	145	24	425	140
SPACE-BASED									
• Satellite Kick System[c]	SS	10,000	2.5	480	40,000	98	20	31	25
• Satellite Kick System[d]	SS	10,000	2.5	480	10,000	98	20	31	25
• Nuclear Waste Disposal[e]	DES	1.5	11	86	48	1.1	2.1	0.093	12
	SS	1.5	11	130	48	1.1	2.1	0.093	12

Notes:
(a) Twenty-g acceleration
(b) One hundred-g acceleration
(c) Pusher plate concept
(d) External concept
(e) Rail stress one-half of the allowable stress for AMZIRC

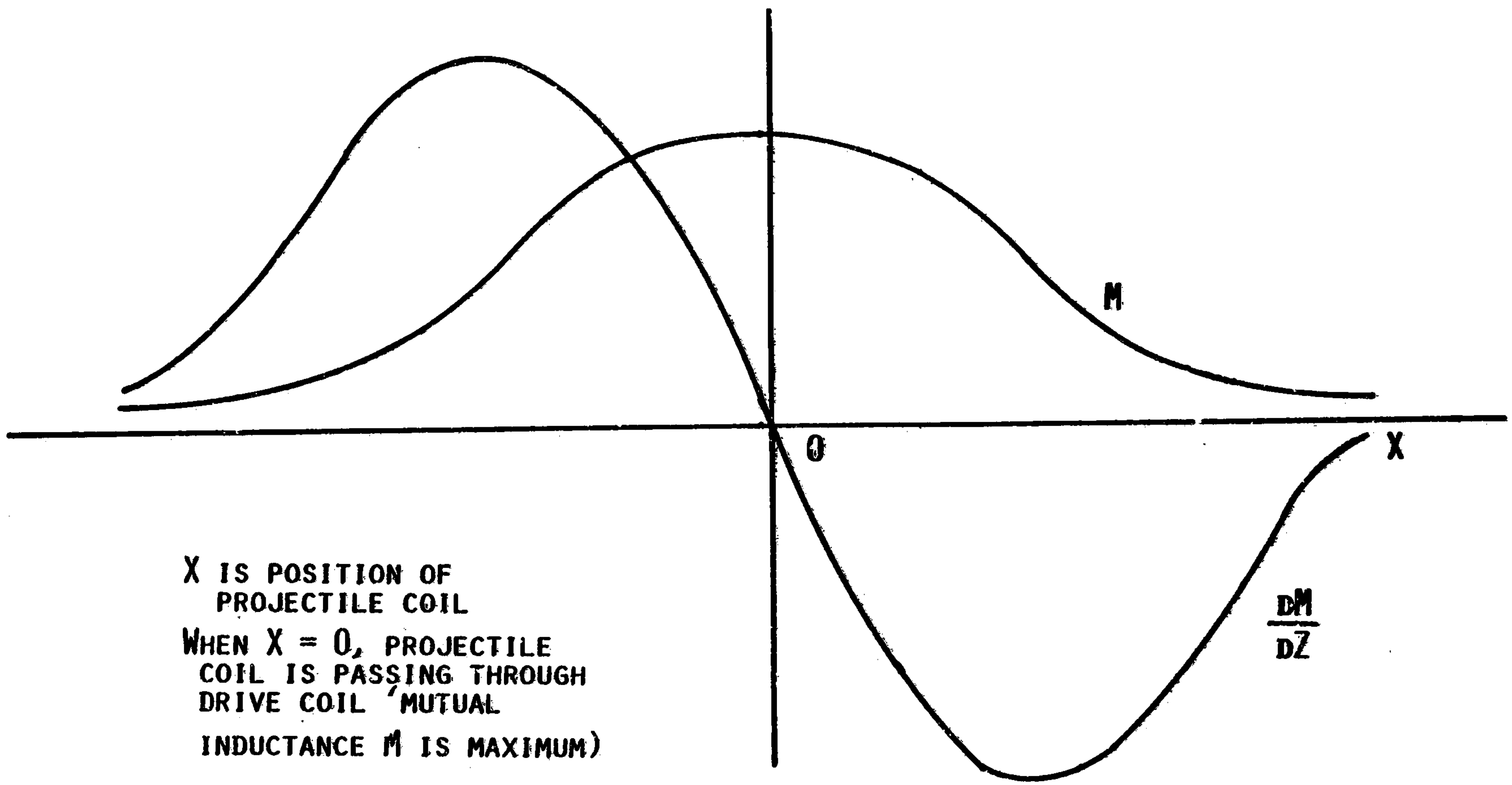


FIGURE 3-2. MUTUAL INDUCTANCE GRADIENT AS A FUNCTION OF PROJECTILE COIL POSITION

where I_p and I_d are the currents in the projectile and drive coils respectively. As seen in the figure, the thrust reverses direction as the projectile coil passes through the drive coil. One method of alleviating this problem is to use a sinusoidally oscillating drive coil current. The current would be synchronized to coincide with the travel of the projectile coil through the drive coil.

TABLE 3-2. SPARK GAP LENGTHS FOR NEEDLE POINT ELECTRODES

Peak Voltage (kV)	Length of Spark Gap for Needle Point Electrodes (cm)
10	0.85
50	5.20
100	15.5
150	26.1
200	35.7
250	45.2
300	54.7

The major advantage of the coaxial launcher over the railgun is that projectile stresses may be reduced by adding multiple projectile coils. This allows the stresses to be distributed along the projectile, so that it may be much longer than a railgun projectile.

The "mass driver" is a discrete-coil coaxial launcher which allows the projectile to be pushed and pulled through the drive coil region. Figure 3-3 shows a section of a discrete coil accelerator. The current is fed into the projectile coil by a set of brushes. Two mass driver prototypes have been fabricated. Mass Driver 1 was built at Massachusetts Institute of Technology in 1976. This launcher accelerated a 0.5 kg projectile to 800 m/s through a 2-m length.

Brush-commutation will not work above certain speeds, so for higher velocities, induction is used to introduce currents into the projectile coils. The pulsed induction accelerator was first demonstrated in the USSR by Bondaletov in 1978. A 2-g projectile ring was accelerated to 5 km/s. Pulsed induction launchers only operate in the "push-mode" since the drive coil itself induces the current. One advantage to the induction methods is that no mechanical contact is necessary between the projectile and the drive coil barrel, so there can be effectively no wear of the system.

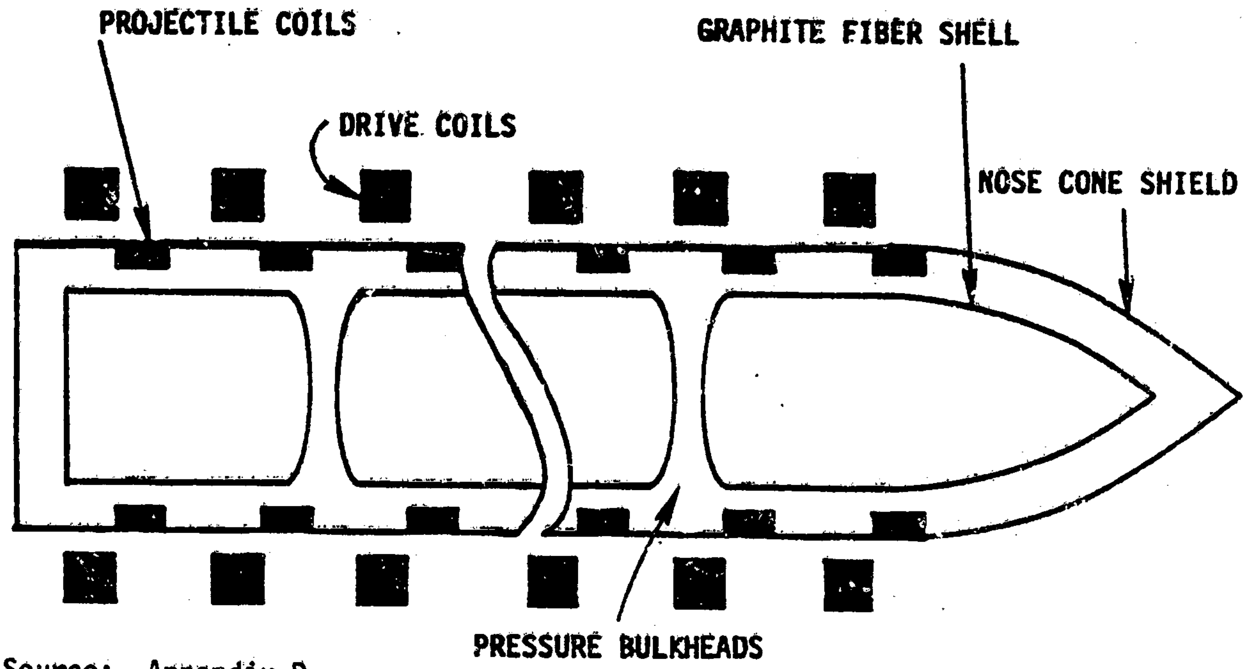


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FIGURE 3-3. DISCRETE COIL LAUNCHER

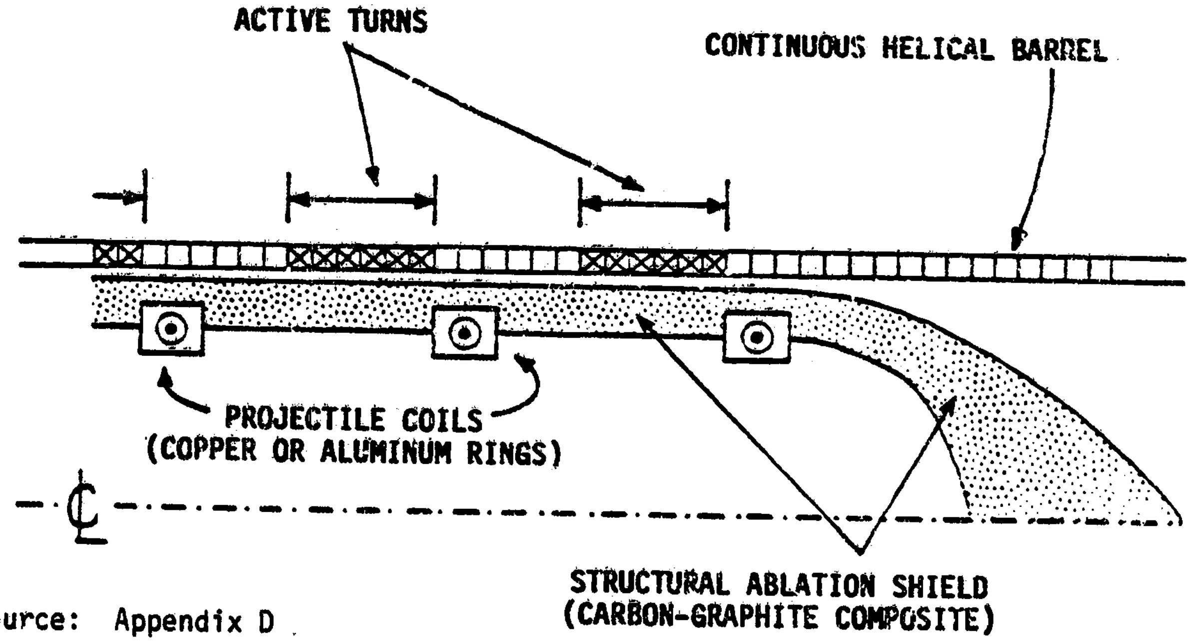

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FIGURE 3-4. SELECTED REFERENCE DESIGN LAUNCHER

Another coaxial accelerator concept is the helical railgun. Instead of discrete drive coils, a continuous helix is used. Brush commutation feeds the excitation current to the projectile coils. MIT demonstrated the concept in 1979 when 4.5 kg gliders were accelerated to 60 m/s.

The coaxial EML concept selected for this study by Henry Kolm and Peter Mongeau (see Appendix D) is illustrated in Figure 3-4. The reference coaxial launcher is a combination of the discrete-coil pulsed-induction accelerator and the helical railgun configurations. As with the induction accelerator, the projectile coils are excited by induction, not by mechanical contact; and like the helical railgun, the concept uses a continuous winding instead of discrete coils, so that most of the commutation energy is stored in the winding rather than in the commutation capacitors. The drive "coils" consist of a continuous helical winding of rectangular copper-alloy wire, instead of discrete coils. Each turn or set of turns is individually commutated by switches, such as solid state or triggered vacuum gap switches, so that an energized segment consisting of several turns (4 to 16 depending upon the mission under consideration) is synchronized to travel with the projectile.

The projectile coils are solid copper or aluminum rings distributed along the projectile at approximately one radius apart. These coils are imbedded in a non-conducting composite, which would also serve as thermal protection and structure, as recommended by Kolm and Mongeau.

The drive barrel and the projectile are in mechanical contact, but no electrical contact occurs. The current is induced from the drive coils to the projectile rings.

A summary of the coaxial launcher concepts for each mission from Appendix D is shown in Table 3-4. The preliminary sizing was done by EML Research, Inc., and is documented in Appendix D.

3.1.3 Earth-to-Orbit EML System

The Earth-to-orbit EML system would launch bulk materials to low-Earth orbit. Materials which were studied include Orbit Transfer Vehicle (OTV) propellants (in water form), Space Station supply requirements (life support consumables, spares, and other miscellaneous items), and raw materials for processing in space. Concept options are illustrated in Figure 3-5.

The operational scenario for this mission begins at the projectile/payload fabrication plant. The projectile, propulsion system, and in some cases, the payload would be integrated before shipment to the launch site. At the site, the projectiles would be stored until time of launch approaches. Water payloads would be supplied by the water plant at the launch site.

TABLE 3-4. COAXIAL LAUNCHER CONCEPT SIZING SUMMARY

Mission	Projectile Mass (km/s)	Exit Velocity (km/s)	Length (m)	Drive Coil Inside Radius (cm)	Force (MN)	Projectile Current (kA)	Drive Coil Current (kA)	Kinetic Energy at Muzzle (GJ)	Back EMF Voltage (kV)
GROUND-BASED									
• Earth Orbital System	3,000	7	2,040	21.8	74	251	154	74	83
• Solar System Escape	1,915	20	2,040	26.0	188	2,380	424	383	1770
• Earth Escape	750	20	2,040	14.3	74	188	387	150	127
• Suborbital	225	5	510	10.2	5.5	149	101	2.8	14
• Electromagnetic Boost									
-Solid Rocket Vehicle	15,000	2	2,040	40.2	15	40	61	30	3
-TAV/SSTO	921,000	0.5	2,550	250.5	45	42	21	115	11
• Lunar Base Supply	3,000	12	2,940	21.8	74	192	202	216	109
SPACE-BASED									
• Satellite Kick System	5,400	2.5	319	28.5	53	224	168	17	29
• Nuclear Waste Disposal	1.5	11	617	2.4	0.15	33	141	0.091	3

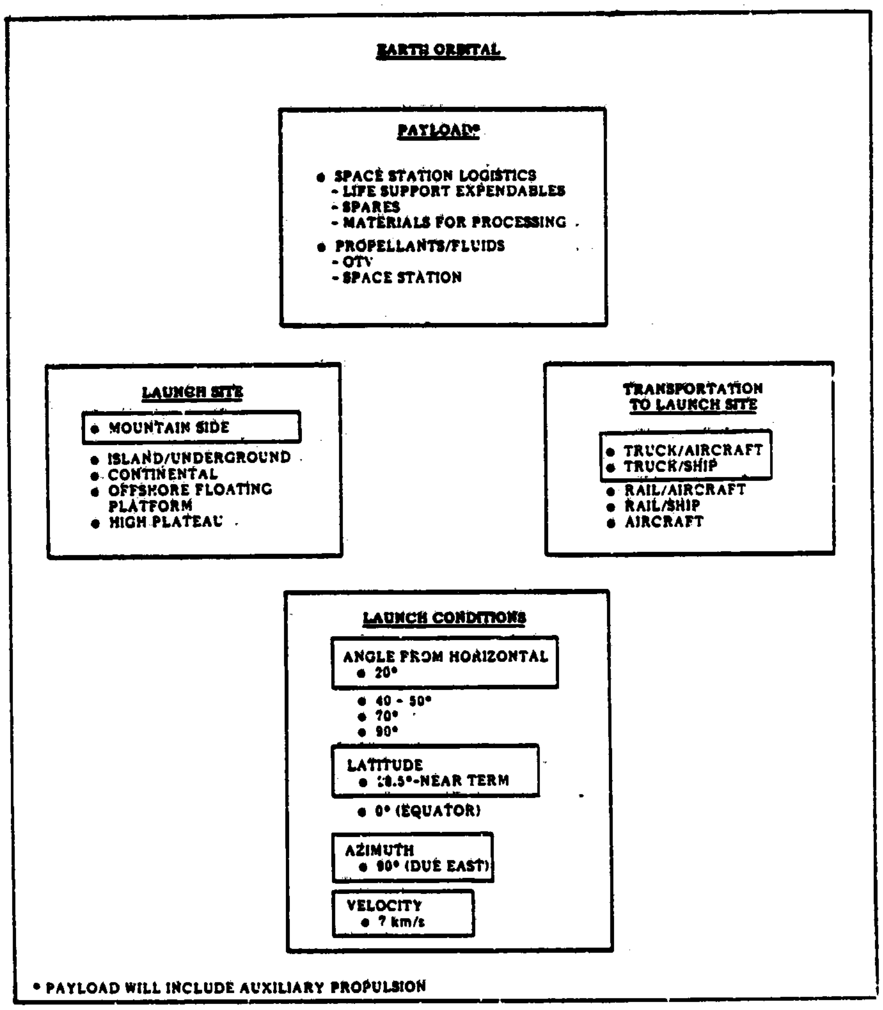


FIGURE 3-5. EARTH-TO-ORBIT OPTIONS

Just prior to launch, the projectile/payload would receive final checkout and then be loaded into the electromagnetic launcher. The projectile would be accelerated to the 6.85 km/s launch velocity before leaving the muzzle of the launcher.

The acceleration limit of launch was set at 2500 g's; however for the 2040-m launcher envisioned the acceleration is approximately 1225 g's.

For a payload of 650 kg, the total projectile mass would be 5900 kg for a railgun system and approximately 3000 kg for a coaxial system. The difference in the two masses is due to the ability of the coaxial accelerator to distribute the acceleration loads into many segments by using multiple projectile coils.

The Earth-to-orbit system would have an elevation angle of 20° from the horizontal. The lends itself well to a launcher system along the side of a mountain, saving construction costs of digging an underground launcher tube 2040 m in length.

One problem for the near-term Earth-to-orbit EML launcher is the Space Station orbit requirement of 28.5° inclination. To supply the Space Station, a launcher at 28.5° latitude could launch once per day to place its payload near the Station. If the system were placed at the equator, two launchers per day could be possible; however, two launcher tubes must be built--one pointed toward the Station on its northward pass and one toward the southward pass. If the Space Station were in a 0° inclination orbit, the station could be supplied on every orbital pass by an equatorial-based EML system. For an orbital altitude of 500 km, the period is approximately 1.6 hours. This corresponds to a maximum launch rate of 16 per day to a single Space Station.

If significant savings could be shown using an EML system based at the equator, NASA might consider a Space Station in an equatorial orbit. This however is not foreseen in the near future.

3.1.4 Earth-to-Orbit Hybrid Launcher

A hybrid EML/rocket launcher was envisioned to launch cargo into low-Earth orbit using the same mission model as the Earth-to-orbit (all EML) launcher. The EML in this case would replace the rocket's first (and largest) stage by providing a velocity boost of 2 km/s from the Earth's surface. Concept options for this and other electromagnetic boost concepts are illustrated in Figure 3-6.

The hybrid system operational scenario is described here. At the solid rocket manufacturing plant, the motor cases would be built and loaded with propellant. The motors would then be transported to the launch site by rail or a combination of rail and ship, depending upon the location of the site. Upon arrival, the motors would be placed in storage.

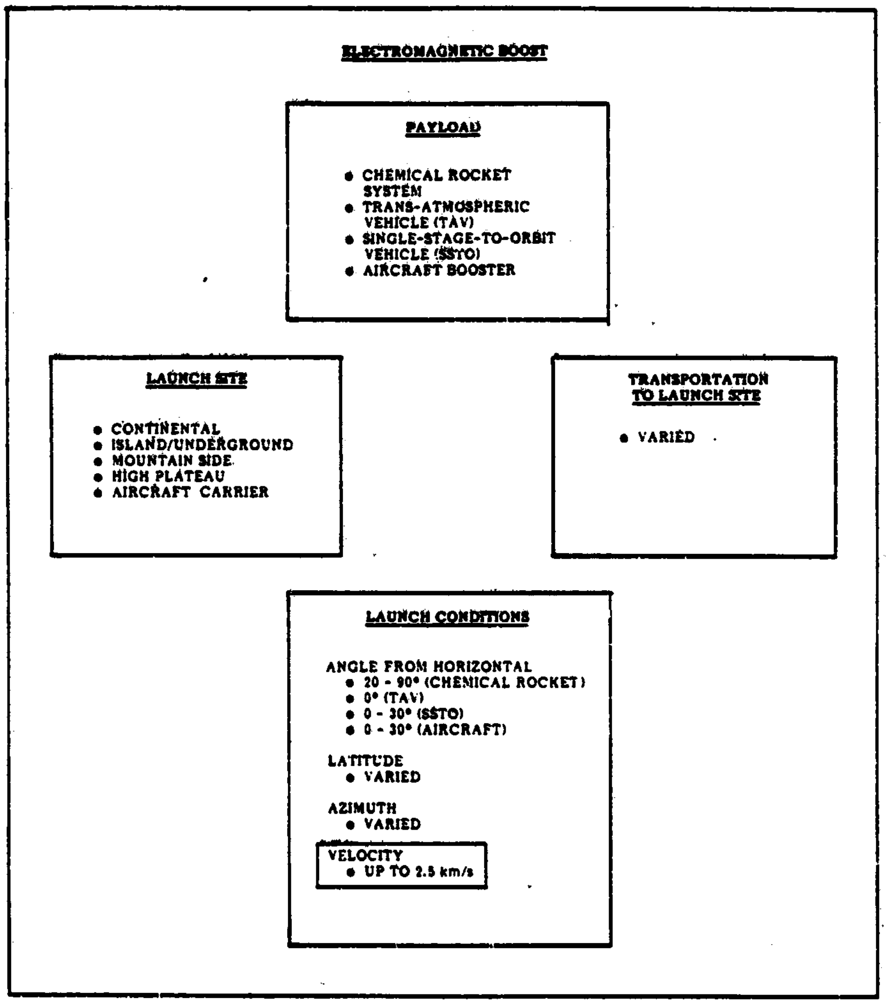


FIGURE 3-6. ELECTROMAGNETIC BOOST OPTIONS

The payloads would be delivered separately to the launch site and stored until launch time. Before launch, the motors would be stacked and the payload integrated. The assembled vehicle/projectile would be loaded into the electromagnetic launcher. All launch systems would be readied, and countdown begun. When the vehicle leaves the muzzle, it would be traveling at 2 km/s. Trajectory simulations indicate that, for maximum performance, the first stage of the rocket would fire after 15 to 20 seconds, and the vehicle would continue along its trajectory to low-Earth orbit.

A mass of 15,000 kg was initially selected for the rocket. Peter Koren of Thiokol was consulted to preliminarily size a three-stage rocket (because of the better performance compared with two-stage rockets). A total ΔV of 9.45 km/s is required for Earth-orbit insertion, with 7.45 km/s of this requirement supplied by the rocket and 2.0 km/s supplied by the EML.

The rocket was initially sized for steel cases with a mass fraction of 0.88. The mass breakdown was as follows:

Stage	Propellant Mass (kg)	Stage Mass (kg)
1	7873	1074
2	3810	519
3	1016	138

Later, a brief analysis was conducted to see if a 15,000 kg rocket was acceptable or if another vehicle size would be more appropriate. A preliminary comparison of various masses of 3-stage solid-propellant rockets was conducted using Battelle's Launch Vehicle Performance Program. This computer program is used for advance planning purposes to analyze launch vehicles. Thrust, I_{sp} or mass flow rate, burn time, and motor jettison masses are entered for each motor. Payload and payload adapter masses are input; payload shroud mass and time of shroud jettison are entered as well. Coast times are assigned by the user to further define the trajectory. The program computes two-dimensional and three-dimentional trajectories and performance reserves. An iterative process is used to calculate the maximum payload for a given vehicle configuration to a defined orbit from a defined launch site.

Battelle's Launch Vehicle Performance Program was run for varying masses at two initial boost velocities (2 and 3 km/s). Coast times between burns were allowed to vary from vehicle to vehicle to determine the maximum payload capability. The results are shown in Figure 3-7 for input conditions of 290 sec I_{sp} (typical solid propellant rocket), 88% stage mass fraction, and final circular orbit altitude of 500 km.

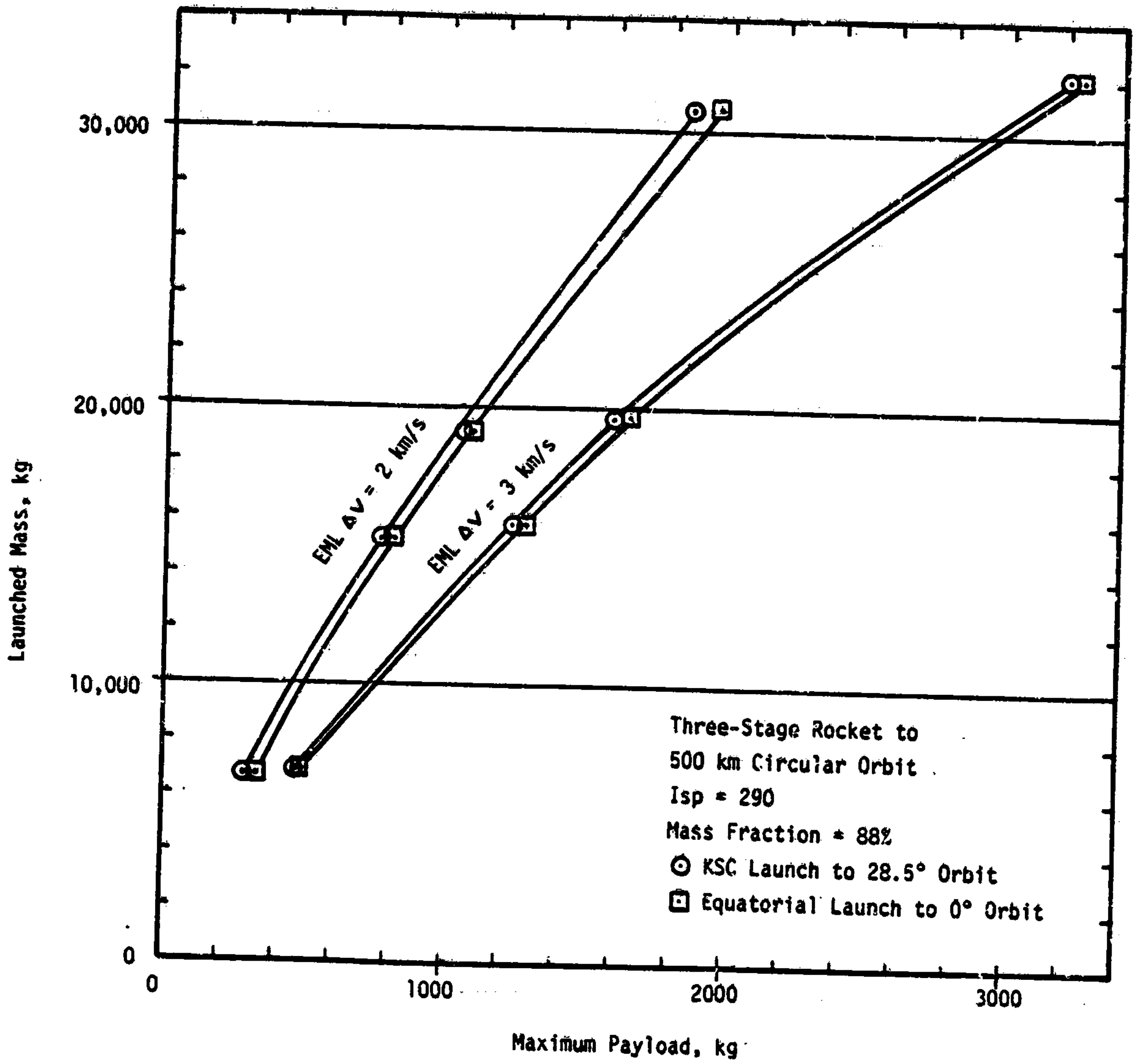


FIGURE 3-7. HYBRID EML/ROCKET MAXIMUM PAYLOAD AS A FUNCTION OF LAUNCHED MASS

Four representative points were chosen off the payload curves to calculate railgun launch parameters as a method of comparison. The four cases are:

(1) 15,000-kg launched mass with EML ΔV = 2 km/s (resulting maximum payload = 800 kg)

(2) 15,000-kg launched mass with EML ΔV = 3 km/s (maximum payload = 1250 kg)

(3) 20,000-kg launched mass with EML ΔV = 2 km/s (maximum payload = 1080 kg)

(4) 20,000-kg launched mass with EML ΔV = 3 km/s (maximum payload = 1650 kg)

For all cases, 100-g acceleration was assumed. The results of the brief investigation are tabulated in Table 3-5.

TABLE 3-5. CALCULATED RAILGUN PARAMETERS FOR SELECTED HYBRID CASES

Parameters	Case Number			
	1	2	3	4
Minimum Length (km)	2.04	4.5	2.04	4.5
Acceleration Time (sec)	2.04	3.0	2.04	3.03
Force (MN)	14.7	14.7	19.6	19.6
Current (MA)	7.67	7.67	8.85	8.85
Launch Kinetic Energy (GJ)	30.	67.5	40.	90.
Peak Power (GWe)	58.8	88.2	78.3	117.
Peak Voltage (kV)	7.67	11.5	8.85	13.3

One of the parameters calculated by the Performance Program is the kick angle (the angle through which the launch vehicle "kicks over", usually during the first stage burn). This angle was used to define the necessary elevation angle of the mountain launch site. For the EML hybrid system, the minimum elevation angle was 35 degrees. This angle is fairly high for most mountains, but it was felt that there may be mountains with this elevation angle over a 2-km length.

At the defined acceleration of 100 g's, the length of an EML system is 2 km. Inside bore diameters are 0.8 m for the coaxial accelerator and 1.2 m for the railgun.

For the coaxial accelerator concept, the projectile coils could be imbedded inside the motor cases during fabrication. For the railgun concepts, a sabot with conducting armature would be placed at the rear of the projectile.

For the same launched mass, increasing the boost velocity from 2 km/s to 3 km/s more than doubled the energy requirement, while only resulting in a 50% gain in payload mass. The energy-to-payload-mass ratios are approximately equal for cases with the same boost velocities; i.e. for Cases 1 and 3 and for Cases 2 and 4. It was decided to continue with the 15,000-kg rocket vehicle. The decision was based in part on the lower mass (the lower the better for near-term EML development). Also, 800 kg was felt to be an acceptable payload capability; for comparison purposes, the Scout launch vehicle is capable of placing 235 kg into a 500 x 500 km orbit when its smaller 0.86-m diameter heat shield is used (Vought, 1980).

3.1.5 Lunar Supply Launcher

The EML lunar supply launcher was envisioned to supply necessary life support consumables and OTV propellants to a manned lunar base. The year 2010 was selected as the initial operating capability for such an EML system. Figure 3-8 presents lunar supply concept options.

For preliminary sizing purposes, the Earth-to-orbit projectiles were used. Maximum acceleration was set at 2500 g's and the launch velocity set at 12 km/s (reference Section 3.6). These requirements correspond to a minimum launcher length of 2940 m.

Ideally, the lunar supply launcher would be located at the equator and would be placed vertically. A 3-km long vertical launcher would be built in an underground shaft similar to that used in the ESRL concept (Rice, et al, 1982).

The projectile would be fabricated in the United States and transported by ship or aircraft to the equatorial launch site. Water payloads would be supplied at the launch site; other payloads (food, for example) would be transported by air.

Launch support facilities would be similar to those defined in Rice, et al (1982) for the ESRL system which was sized for a 10-km/s launch velocity.

One major problem of the lunar supply launcher is the large amount of auxiliary propulsion which must be carried for midcourse correction, lunar orbit insertion, and, if required, lunar landing. Section 3.8.3 describes the propellant requirement necessary and shows that a payload of approximately 500 kg would be possible for lunar-orbit insertion when using the Earth-to-orbit projectiles.

Assuming a launch capability of one launch per hour (one on either side of a daily launch window), 12 launches would be possible per month (144 per year). With a maximum payload mass of 500 kg, this corresponds to a maximum yearly payload delivery of 72 MT. If a launch every half-hour were attainable, 18 launches per month could be possible. This would correspond to a maximum yearly delivery of 108 MT of payload.

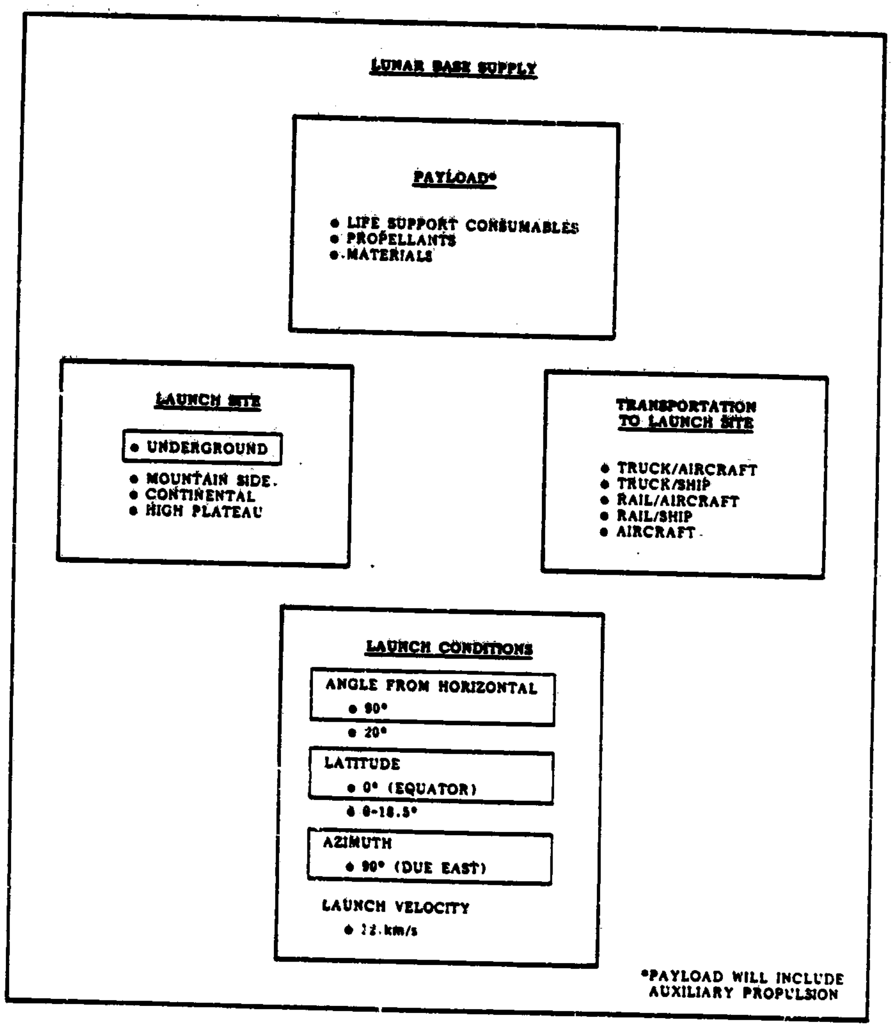


FIGURE 3-8. LUNAR BASE SUPPLY OPTIONS

3.1.6 Solar System Escape Launcher

High-level and transuranic (TRU) nuclear wastes produced by U.S. commercial nuclear power plants would be sent to a waste processing facility (see Section 2.3 for discussion of TRU waste processing and Rice, et. al., 1982 regarding high-level wastes). This facility would be located in the United States and would perform a dual function by processing the waste and fabricating the projectile. The waste would be stored for a number of years before becoming part of the projectile. Concept options for the solar system escape and Earth escape missions are presented in Figure 3-9.

The nuclear waste is encorporated in a stainless steel radiation shield. The shield requirement is thicker for the high-level waste form than for the TRU wastes, because of the higher levels of radiation. Structural strength is also provided by the shield. Carbon-carbon materials are applied to the outside of the projectile for thermal protection. A tungsten nose cone is believed to be required to withstand the 20 km/s launch velocities.

The finished projectiles would be transported to the remote island launch site by railroad, then by an ocean vessel. At the launch site, the projectiles would be placed in a storage and checkout facility.

At launch time, the projectile is loaded into the EML system and, when the launcher is ready for launch, the projectile is launched at 20 km/s to escape the solar system. Figure 3-10 provides an overview of this concept scenario.

Since it was felt that the projectile could withstand high launch accelerations, 10,000 g's was set as the acceleration limit. This corresponds to a launcher length of 2.0 km. Because of the length of the launcher, the vertical launch requirement, and the safety considerations, an underground launch system on a remote location is needed.

To dispose of all the high-level and TRU wastes produced by U.S. commercial nuclear power plants, 26 launches per day are necessary during the 6-hour launch window. Four launcher tubes would be required at a minimum, and the power requirements would need to be increased over the ESRL requirements (Rice, et al, 1982) to provide for one launch every 15 minutes (one launch per hour per tube).

3.1.7 Earth-Escape Launcher

The Earth-escape launcher would be used to launch planetary explorers and probes. A payload mass of 600 kg was believed to be large enough to provide an adequate payload, yet at the same time allow a reasonable projectile mass. The projectile would consist of the 600-kg

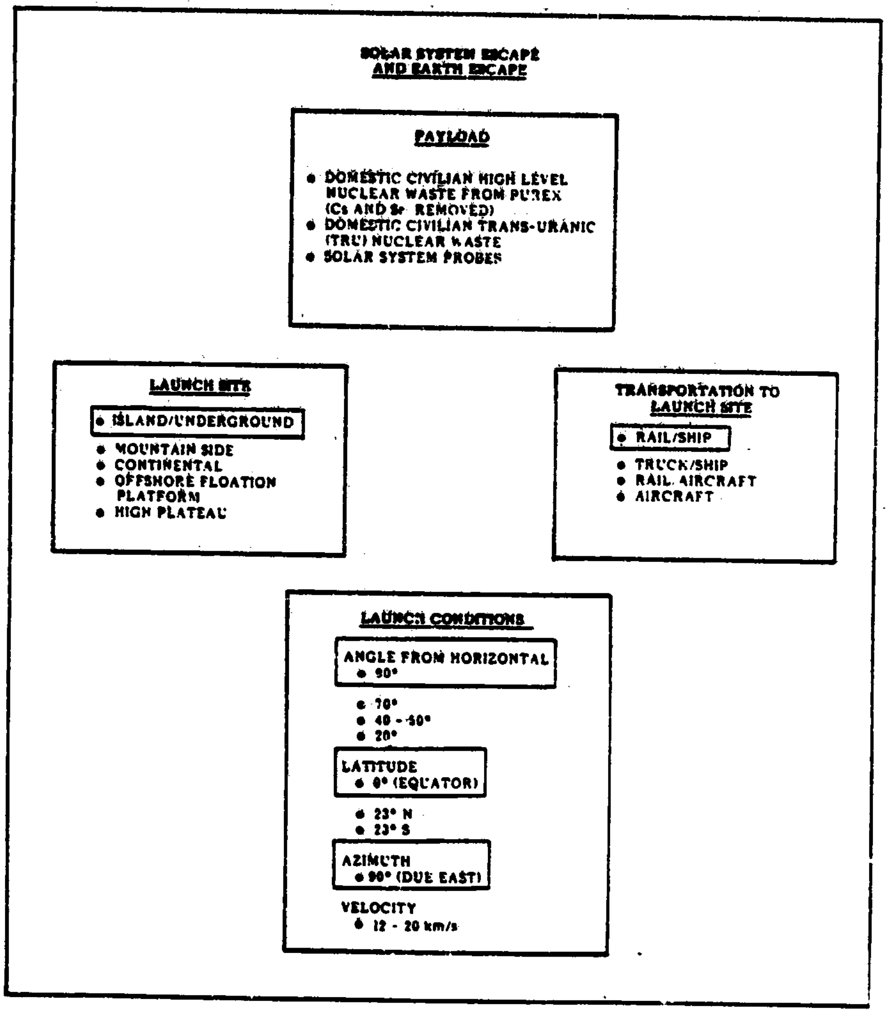


FIGURE 3-9. SOLAR SYSTEM ESCAPE AND EARTH ESCAPE OPTIONS

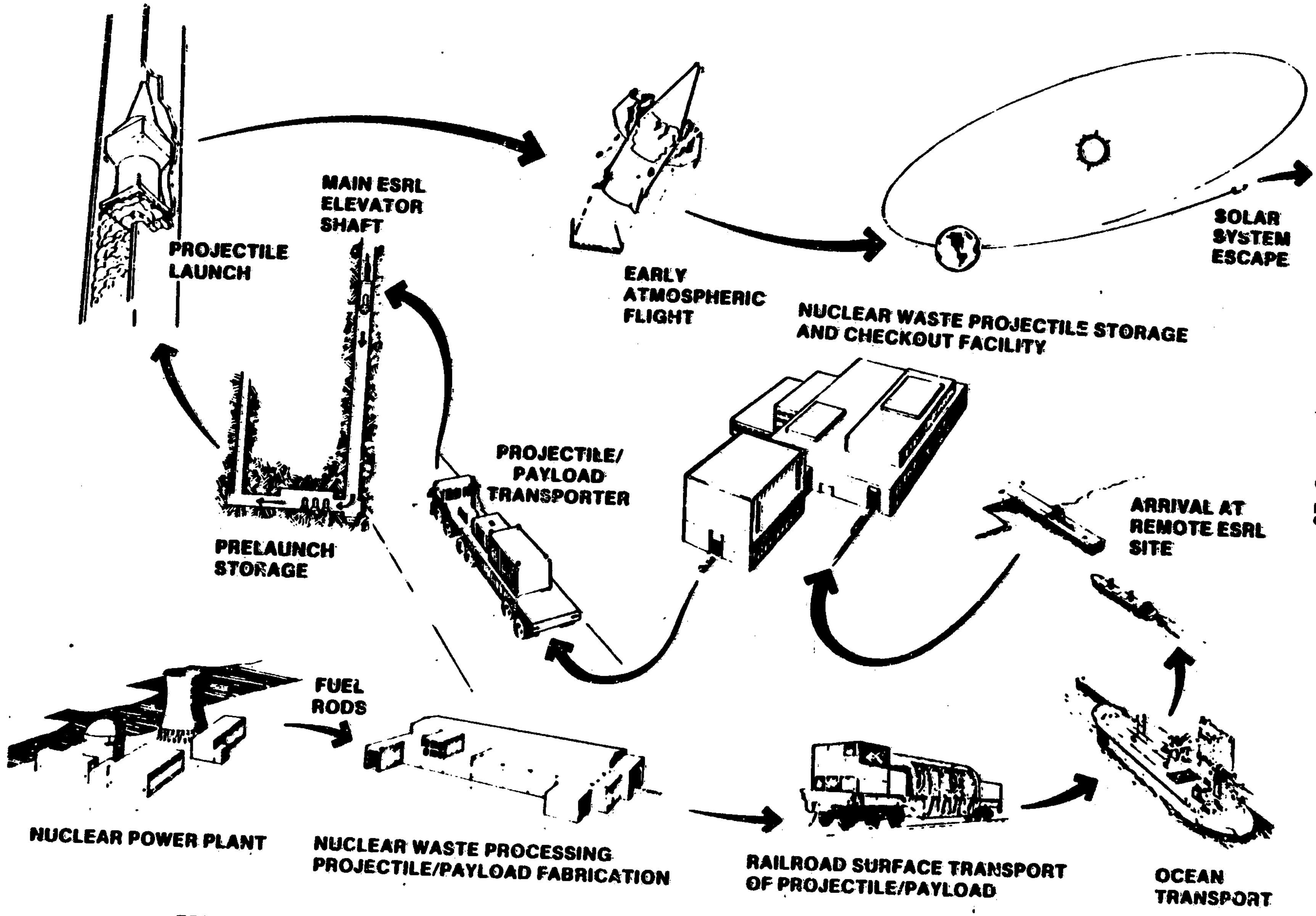


FIGURE 3-10. OVERVIEW OF NUCLEAR WASTE DISPOSAL IN SPACE MISSION SCENARIO (RAILGUN LAUNCH ILLUSTRATED)

payload, a small auxiliary propulsion system for mid-course guidance, structural support for the payload and propulsion system, stabilization fins, and a tungsten nose cone to withstand atmospheric flight during the 20 km/s launch velocity. Figure 3-9 illustrates the concept options.

An acceleration limit of 10,000 g's was set to provide reasonable launcher lengths. At a launch velocity of 20 km/s, the minimum length would be 2040 m. The coaxial accelerator would appear to be more attractive for this case, because the acceleration loads could be distributed along the projectile, saving a significant amount of structural mass. Preliminary conceptualization yielded projectile masses of 2055 kg for a railgun system and 750 kg for a coaxial accelerator system, with the major difference being structural mass.

The Earth-escape launcher would be placed vertically and therefore would require an underground site to support the system. The launch velocity must be variable to accommodate launches to different planets and solar system locations (12-20 km/s velocities were envisioned, see Section 3.4 for details).

3.1.8 SSTO/TAV Booster

The Transatmospheric Vehicle (TAV) and Single-Stage-to-Orbit (SSTO) vehicle are concepts studied by the USAF (Battelle, 1983) and NASA (Jackson, 1983; Eldred, 1984), respectively. Most of these vehicle concepts utilize rocket propulsion, although airbreathing propulsion is being considered. The various concepts range in mass from about 500,000 to 5,000,000 kg. Concept options are shown in Figure 3-6.

The design goals for the TAV are to use a fully reusable, horizontal takeoff, single-stage vehicle (with ground assist) to deliver 9100 kg (20,000 lbs) of payload to a low polar orbit. The vehicle must be ready to launch within 5 minutes following an alert with turnaround time to allow 2 launches per day. The TAV operational requirements are summarized: manned operation; minimal ground support; horizontal takeoff and landing; flexible basing; and adverse weather operations. During the Phase I USAF/ASD study conducted by Battelle, 14 TAV concepts were evaluated. Of those, six were considered in the class of "single-stage-to-orbit" type systems and three used ground-assist sleds of various types. These systems would be excellent candidates for EML use for the initial boost.

The civilian SSTO goals and requirements are similar but less severe. Ground support and quick turnaround time are not as critical and the use of dedicated, vulnerable launch facilities are not critical issues.

The projectile (vehicle) mass was set at 900,000 kg for this study. Since the vehicle will be manned, a 5-g acceleration limit was set. It was felt that an initial velcity boost of 500 m/s was sufficient

to launch the vehicle. This led to EML system lengths which are of the order of existing aircraft runways (2.55 to 3.83 km).

The railgun concept presented here for a TAV/SSTO launch is illustrated in Figure 3-11. The projectile/vehicle sits on top of a solid sliding armature. The vehicle for the coaxial accelerator concept rests upon a support structure containing a saddle-shaped projectile coil (shown in Figure 3-12 and on page 92 of Appendix D). The support structure slides along the length of the drive coils.

3.1.9 Suborbital Launcher

For most of the missions described in Section 2.5 (atmospheric soundings, astronomy experiments, and reentry tests), a vertical or near vertical launch is preferred over a 20-35° launch. A mountainous site is then eliminated, with an underground facility the likely siting. The various concept options are shown in Figure 3-13.

Payloads would be manufactured under the guidance of the principal scientists. They would then be sent to the projectile manufacturing facility for integration. The combined projectile/payload would be sent to the launch site for final checkout before launch.

The railgun projectile was sized as a scaled-down Earth-orbital projectile. A 225-kg projectile was envisioned, with a square sabot 32.5 cm on a side. The projectile itself would be 30 cm in diameter, with a projectile/sabot length of 1.2 m.

The rails would be from 510 to 765 m long depending upon whether distributed or single energy storage was used. With these lengths and a nearly vertical launch angle to suit most mission needs, the launcher tube would most likely be placed underground to provide structural strength.

The coaxial projectile was also sized at 225 kg in mass, with a payload of 160 kg. The projectile would have a diameter of 20 cm and length of 2.8 m.

Although launch site flexibility is highly desired for most types of suborbital missions, 56% of the US sounding rocket launches have been fired from two locations, with another 14% from a third site. An EML suborbital launcher tube would obviously be fixed, because of its large size and many supporting systems.

3.1.10 Space-Based Systems

The scope of this study encompassed space applications for both Earth-based and space-based EML systems. Two orbiting space-based systems were identified and are described in this section: satellite kick system and high-level nuclear waste disposal system. Concept options for both missions are presented in Figure 3-14.

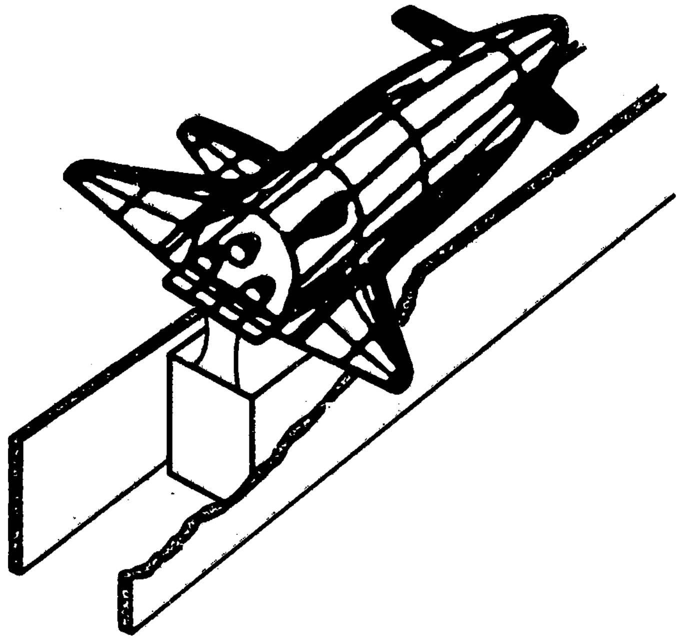


FIGURE 3-11. RAILGUN LAUNCHER FOR TAV OR SSTO VEHICLE

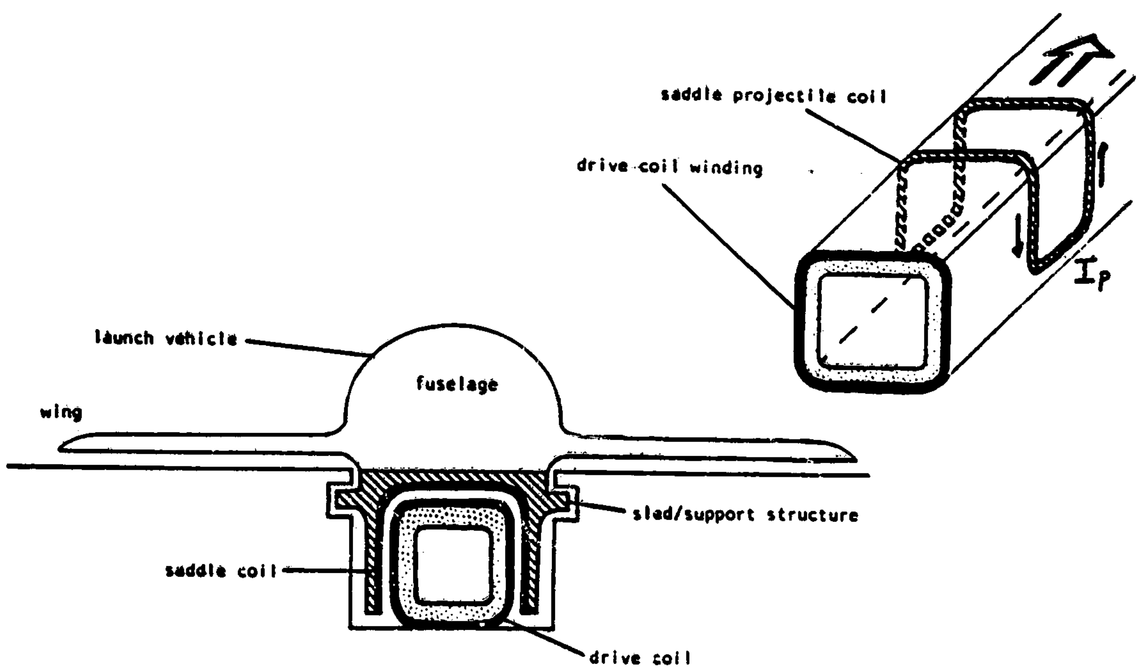


FIGURE 3-12. SADDLE COIL LAUNCHER FOR TAV AND SSTO APPLICATIONS

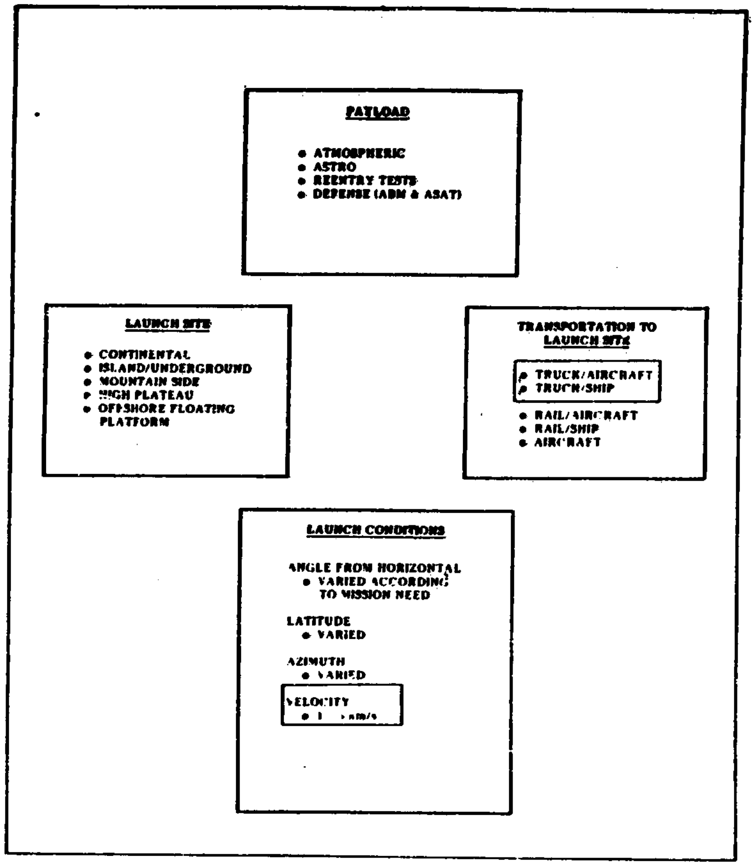


FIGURE 3-13. SUBORBITAL OPTIONS

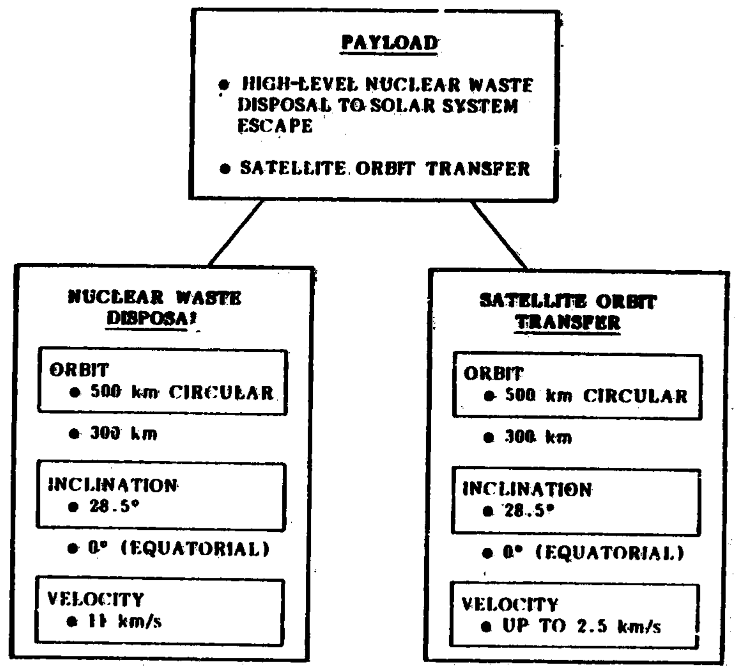


FIGURE 3-14. SPACE-BASED OPTIONS

Required velocity impulses are smaller from orbit than from the Earth's surface. The minimum solar system escape velocity from Earth's surface is 16.7 km/s (no drag included), while this number is reduced to approximately 10 km/s with a low-Earth orbit launch (Section 3.3).

Orbiting space-based EML systems have several inherent disadvantages when compared with Earth-based systems. One disadvantage is construction cost. For near-term operation, the equipment must be built on Earth and delivered to orbit via the Space Shuttle. For long launcher tubes, this may be expensive because of the many Shuttle flights required for assembly (the dimensions of the payload bay are 18.3 m long and 4.5 m in diameter).

Another disadvantage is Newton's third principle: for every action, there is an equal and opposite reaction. For the launcher to maintain the orbit, a sizeable impulse is required to balance every shot or series of shots. Drag makeup and attitude control would also be needed, which would incur additional propulsive maneuvers.

3.1.10.1 Satellite Kick System

The first space-based EML delivery system is a satellite kick system which would replace the need for an OTV or large upper stages. The EML satellite kick system would be based as an orbiting Space Station. This Space Station is envisioned to be in a 500-km circular orbit at 28.5° inclination to facilitate delivery to and from the Station by a Space Shuttle or derivative.

Because the launch rates would be low (one to two launches per month), the Station need not be a dedicated facility. A nuclear power plant could be used both to power the EML system and to supply needed Station power requirements. Thermal radiators would be used to dissipate generated heat into space. A storage module would be required to store the satellites after Shuttle delivery to orbit before the designated launch dates.

The Space Shuttle would deliver the satellite payload to the Space Station, where it would be stored until time of launch. At that time, the satellite and its supporting systems would be checked out. The payload would be loaded into the breech of the launcher and launched to its required velocity (up to 2.5 km/s). When the proper velocity is reached, the payload would be released from its armature/driver and continue towards its destination. To avoid orbital debris problems and to maintain high reusability, a deceleration system is required for the driver mass.

The minimum velocity needed to place the satellite system into a geosynchronous transfer orbit with perigee altitude of 500 km is 2.36 km/s. This minimum velocity was calculated assuming a Hohmann transfer from 500 km altitude. The eccentricity of the transfer orbit is 0.72, leading to a velocity of 9.97 km/s required at perigee to sustain

the transfer orbit. The orbiting EML system is traversing around the Earth at 7.61 km/s, the satellite therefore needs an additional 2.36 km/s. Lower apogee altitudes require smaller velocity increments. Therefore, the system was sized for velocities up to 2.5 km/s; the actual velocity per shot would depend upon the desired orbit. This single impulse system would only provide the ΔV required to put the satellite into an elliptical transfer orbit. The satellite would need to have an attached small propulsion system to circularize at the apogee altitude, if this is required. Again assuming a minimum energy Hohmann transfer, 1.5 km/s would be required for circularization at GEO altitude

The payload consists of the satellite and its supporting systems. Preliminary analysis indicated that to be competitive with other systems, a useful payload mass of 5000 kg and a minimum payload diameter of 2 m was needed. An acceleration limit of 1000 g's was placed on the launcher system. This limit was considered to be high, considering the delicate nature of many satellites, but was needed to limit the length of the system.

Two types of railgun systems were envisioned (Figure 3-15). A large solid metal armature was used in both cases. The first configuration has the satellite riding on a support structure on top of the armature external to the railgun. This concept has some obvious structural problems to overcome. The off-center-of-gravity loads may become severe when traveling at 2.5 km/s, and could damage not only the satellite support structure but the rails themselves. An advantage of this concept is that the bore need not be as large as 2 m in diameter.

The satellite was launched "conventionally" in the railgun bore in the second concept. The armature acts as a pusher plate behind the satellite. This method eliminates the structural problems associated with the external satellite, but increases the diameter of the bore to at least 2 m.

Both railgun concepts assumed a single energy store, for ease of assembly and maintenance. This increased the length of the rails from a minimum of 320 m to 480 m. If only one rail segment at a time could be transported by Shuttle, this corresponds to at least 27 flights to deliver the rails alone, and more would required to deliver the rest of the launch system.

The coaxial accelerator concept consists of 319 m of drive coil. The drive-coil inside diameter is 57 cm, which is very small for a satellite system. For comparison, the Delta launch vehicle shroud is 2.4 m (8 ft) in diameter. The Scout heatshields are 1.03 m and 0.86 m in diameter.

3.1.10.2 Space-Based Nuclear Waste Disposal

The projectile consists of a small unshielded high-level waste (cermet waste form) billet encapsulated in an insulator/sabot. The billet

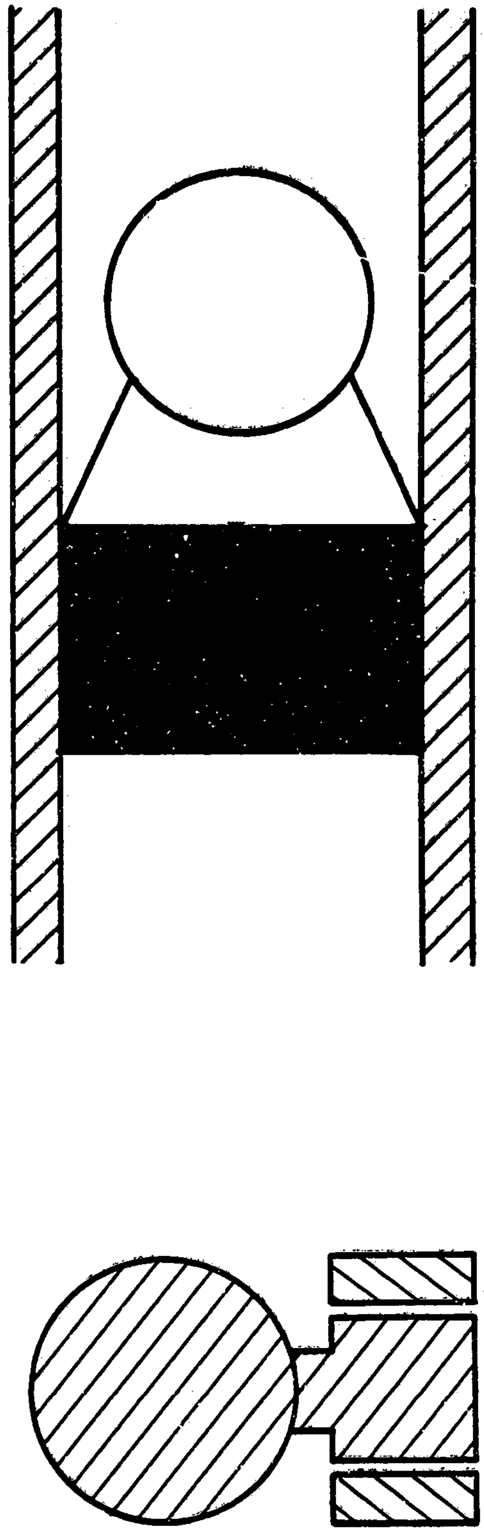


FIGURE 3-15. POSSIBLE RAILGUN CONFIGURATIONS FOR SATELLITE KICK SYSTEMS

is cylindrical in shape, with a height of 5.858 cm and a diameter of 5.852 cm. The mass of the waste billet is 0.95 kg.

The waste billets are transported by an uprated Space Shuttle to the orbiting EML platform in two shielded stainless steel spheres. Each spherical assembly contains 3167 billets (3000 kg of waste), for a total of 6000-kg mass per Shuttle launch.

To completely dispose of all U.S. commercial high-level waste, an average of 500 kg must be launched daily using the EML system (Rice, et al, and Earhart, 1982). Therefore one Shuttle delivery flight is required every 12 days (30 Shuttle launches per year dedicated to nuclear waste disposal).

Because of the very high radiation doses from the unshielded waste billets (see Section 3.2), the EML platform area must be unmanned or shielded during launch, etc. After the Shuttle leaves, the billets would be unloaded from the delivery spheres and placed in the launch sabots. The waste projectile (billet and sabot) would be loaded into the launcher. Because the platform would be orbiting around the Earth, a launch window of about 19 minutes would be available for launch on every orbital pass. This corresponds to a launch every 36 seconds during the window. The entire process from unloading the billets to commencing the launch procedure would be automated.

The platform was assumed to be located in a 500-km circular orbit at 28.5° inclination to facilitate launches from an existing pad at Kennedy Space Center. A nuclear power plant was assumed to provide the necessary power for all platform operations. Large radiators would be used to dissipate the heat generated during the 11 km/s launch. The minimum acceleration length as a function of launch acceleration is shown below:

Acceleration (g's)	Length (m)
1,000	6,170
10,000	617
25,000	247
50,000	123
75,000	82

Obviously, to reduce the cost of Space Shuttle delivery costs for assembly and to reduce the risk of orbital debris hits, the acceleration must be as high as possible.

Kolm and Mongeau selected a 10,000 g acceleration limit for their coaxial accelerator design, but commented that this limit could be increased to 30,000 g's "without much extrapolation of present

technology" (Appendix D). They also modified the shape of the projectile/billet (mass remained at 1 kg of waste form) to allow four projectile coils.

When the preliminary railgun system was conceptualized, it was felt that, because the projectile would be given a large enough impulse to exit the solar system from orbit, the condition of the waste billet upon leaving the EML was not as critical as for the case of an Earth-based launch which has greater probability of an accident impacting the Earth. To keep the launcher at reasonable lengths, the acceleration limit was defined by the launcher materials, not the projectile material. Assuming a safety factor of 2 and AMZIRC rails, the minimum length of the rails is 86 m for an acceleration of 72,000 g's and bore area of 48 cm^2 (3.9-cm radius). A distributed energy storage system would then be 86 m long. For a single energy store, the length would become 138 m (at four times the energy cost for simplicity of system).

3.2 Analysis of Radiation from Orbiting EML (Nuclear Waste Disposal System)

The projectile for the space-based EML nuclear waste disposal system consists of commercial high-level nuclear waste. The waste mix is defined as a high-level Purex (plutonium and uranium extraction) waste with 90 percent of the cesium and strontium removed (Modified PW-4b). The cylindrically-shaped waste form is the Oak Ridge National Laboratory iron-based cermet.

The nuclear waste radiation analysis was performed to determine the dosage obtained from an unshielded cylindrical waste form billet, with a radius (r) of 2.926 cm and length (l) of 5.858 cm. The analysis was conducted at "detector points" at various distances from the waste form, beginning at the billet surface to a distance of 1 km. The source term assumed for the waste mix is the same as that used in Rice, Miller, and Earhart (1982).

Figure 3-16 illustrates the billet geometry used during the radiation analysis. The dose (in rem/hr) was calculated at a number of detector points at varying distances (d) from the projectile. Table 3-6 summarizes the results of these calculations.

3.3 Launch Window Analysis for Solar System Escape from Orbiting EML System

3.3.1 Previous Work

The Battelle ESRL study considered the impulsive launch of payloads from the Earth's surface to interstellar space (Rice, et al, 1982). The launch velocity requirements were estimated as a function of launch latitude, time of day, and time of year. Both vertical and non-vertical launches were considered and atmospheric drag penalties were included. The primary conclusions were as follows:

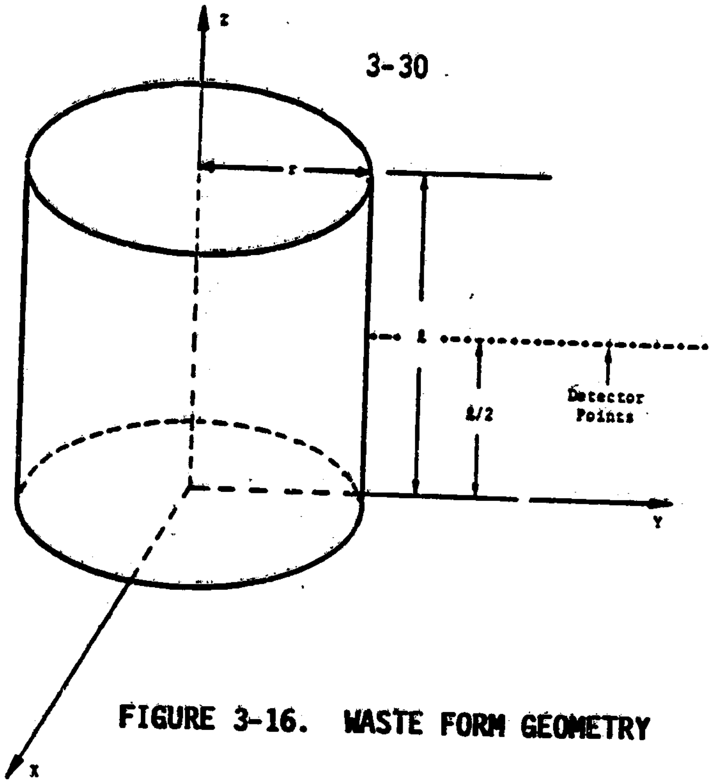


FIGURE 3-16. WASTE FORM GEOMETRY

TABLE 3-6. RADIATION DOSAGE AS A FUNCTION OF DISTANCE FROM THE WASTE FORM PROJECTILE

d (m)	Dose (rem/hr)
0.02926	5.364×10^{5}
0.05	1.420×10^{5}
0.10	3.553×10^{4}
0.20	8.984×10^{3}
0.50	1.433×10^{3}
1.0	3.609×10^{2}
2.0	9.022×10^{1}
5.0	1.442×10^{1}
10.0	3.593
20.0	8.893×10^{-1}
50.0	1.347×10^{-1}
100.0	2.897×10^{-2}
200.0	4.687×10^{-3}
500.0	1.290×10^{-4}
1000.0	1.122×10^{-6}

- The minimal launch velocity (approximately 16.7 km/s for no drag) for solar system escape is independent of latitude up to a latitude of approximately 23.5 degrees.
- The penalty for launching at non-optimal times of the year is reduced with an equatorial launch site.
- The optimum time of day is approximately 6 a.m. for any latitude and launch date.
- For equatorial launches, two optimal launch dates occur each year--90 days after the vernal equinox and the autumnal equinox, respectively.
- For vertical equatorial launches, with drag effects included, a launch velocity of 19 km/s could provide a daily launch window ranging from 3.5 to 4.6 hours. A launch velocity of 20 km/sec could provide a daily launch window of approximately 6 hours.

3.3.2 Orbiting EML System

In the current investigation, the effects of launching from a platform in Earth orbit were examined. From a flight-mechanics standpoint, a launch from orbit differs from a surface launch in the following respects:

- The launch energy requirements are reduced.
- Unlike a surface launch, the projectile would be injected into its trajectory at the perigee of the Earth-escape hyperbola, thereby altering the time-of-day relationship
- Since the launcher is traversing its orbit at a rate of 24 hours (local time on Earth below the vehicle) in about 90 minutes, the daily launch window is compressed.

Figure 3-17 illustrates the combined effects of the factors just discussed for a launch from a 500 km orbit. The remaining projectile velocity is shown at an infinite distance from the Sun for a launch on an optimum date.

As shown in Figure 3-17, the minimum launch velocity has been reduced by approximately 10 km/s as compared to a surface launch; and the best local launch time is near midnight rather than 6 a.m.

If a launch velocity of 10 km/s is assumed, the launch window per orbit would be from about 7 p.m. to 3:30 a.m., local time. On the other hand, because of the orbital velocity of the launcher, this difference in local hour-angle would be traversed in about 24 minutes.

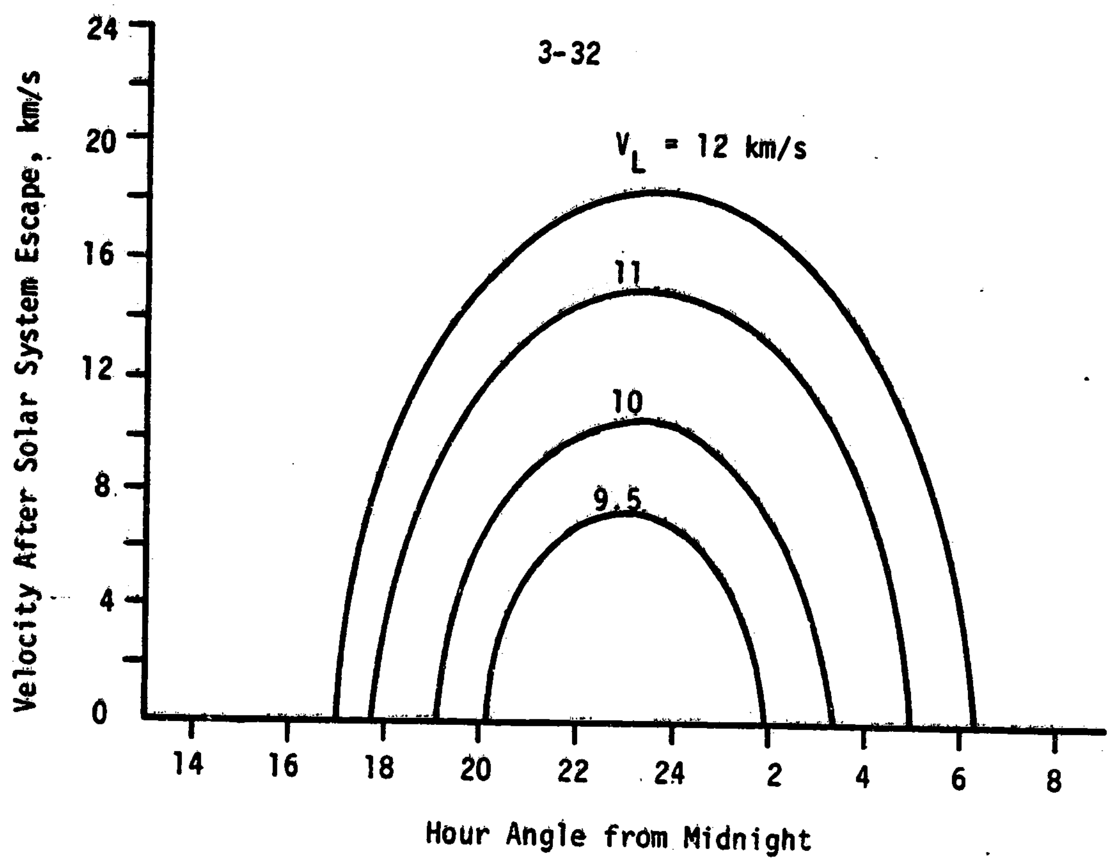


FIGURE 3-17. LAUNCH FROM 500-km ORBIT ON BEST DAY

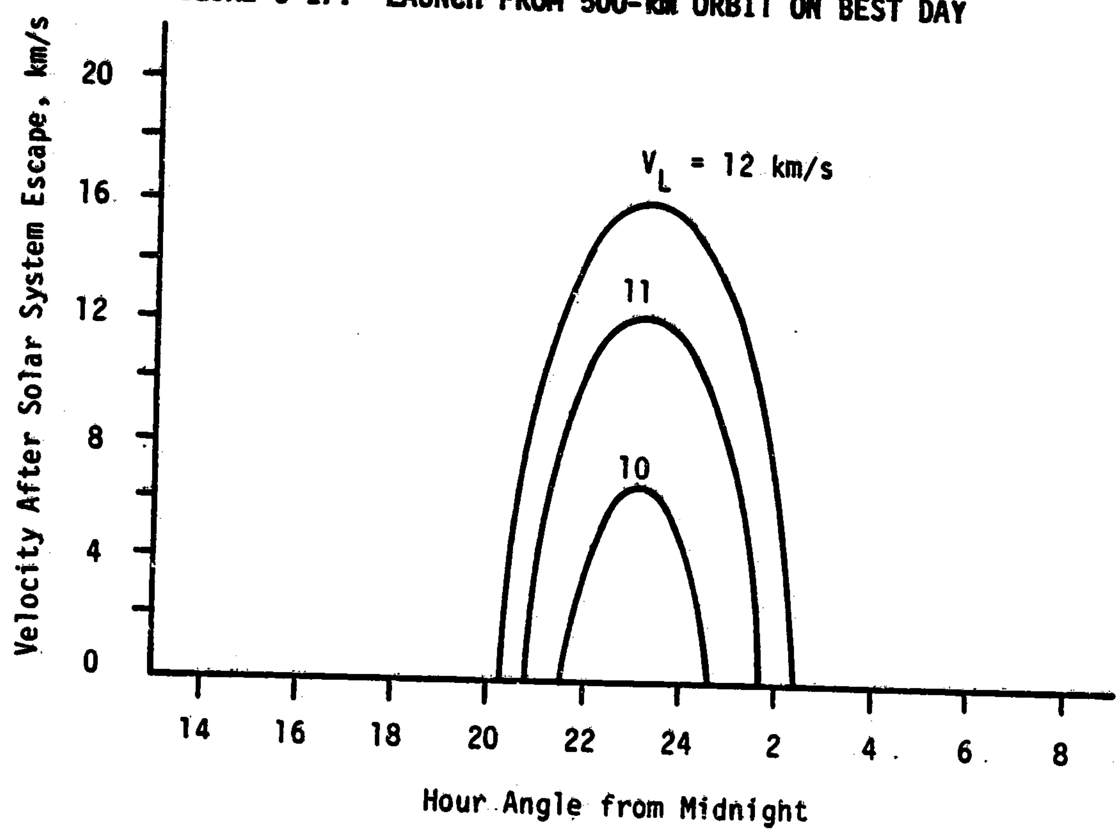


FIGURE 3-18. LAUNCH FROM 500-km ORBIT ON WORST DAY

Figure 3-18 is a similar plot for a worst day of the year (at either equinox). On these dates, the actual launch window would be reduced to about 12 minutes on each orbit, or approximately 3.2 hours per day, for the 10-km/s launch velocity.

Although the launch of payloads to solar system escape appears reasonable from a launch velocity standpoint, consideration must be given to the fact that reactive impulse would be imparted to the launcher, equal and opposite to the impulse applied to the projectile. To maintain the orbit of the launcher, a sizeable corrective impulse would be required after each projectile launch.

3.4 Launch Window Analysis for Solar System Probes

For payloads launched to destinations within the solar system, only vertical launches from the Earth's surface were examined.

To compare solar system probe launches to solar system escape missions, it is obvious that solar system probes will require lower launch velocities. However, the solar system probes are constrained to deliver a payload to a particular destination. For flights to the planets, the geometric constraints are formidable for a fixed launcher. The analysis of launches to specific planets was considered to be beyond the scope of this effort. Instead, the analysis was confined to simply establishing the launch requirements for delivering probes to various distances from the Sun.

For destinations outside of the Earth's orbit, vertical launches occurring near 6 a.m. will make maximum use of the Earth's velocity. For destinations closer to the Sun, launches near 6 p.m. will be properly aligned to subtract most effectively from the Earth's velocity, thereby achieving the smallest perihelion.

Figure 3-19 shows both cases for an assumed launch velocity of 16 km/s (no drag penalty included). The aphelion possible on an optimum day is seen to be over 20 Earth orbit radii from the Sun. As in the case of solar system escape payloads, the optimum days are 90 days after each equinox. On a worst day, the same launch velocity could still produce an aphelion beyond the orbit of Saturn.

For probes nearer the Sun, a worst-day launch would provide a perihelion inside the orbit of Mercury, while an optimum day would permit a perihelion of less than 0.25 Earth radius from the Sun.

Figure 3-20 confines attention to probes to the outer solar system, and shows aphelion as a function of launch velocity for 6 a.m. launches. Although flight times to aphelion are not shown, these times can be quite long for these minimum-energy trajectories. For the cases shown, the flight time to the orbit of Jupiter would be about 2 years

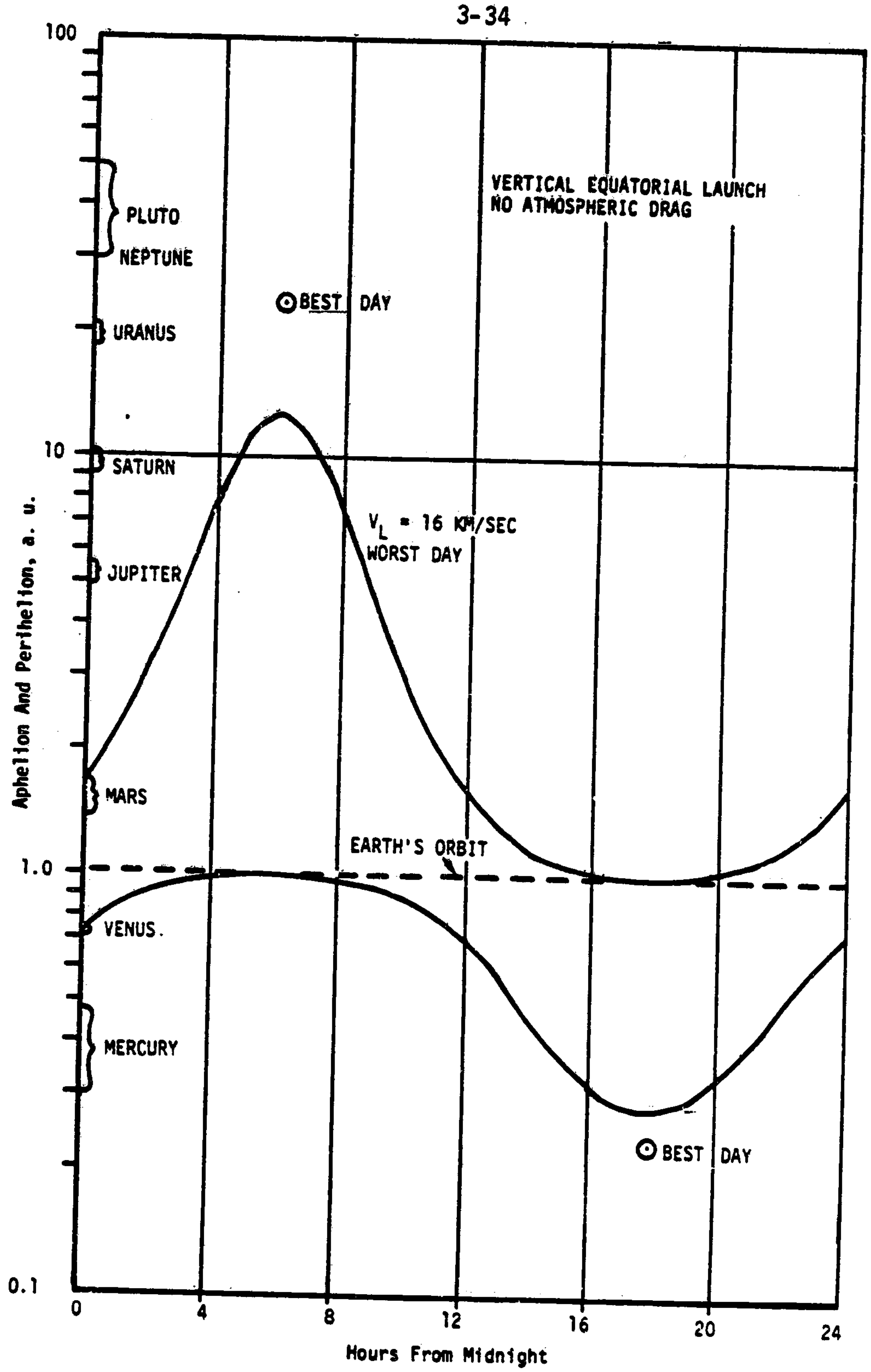


FIGURE 3-19. SOLAR SYSTEM PROBE APHELION/PERIHELION RANGE

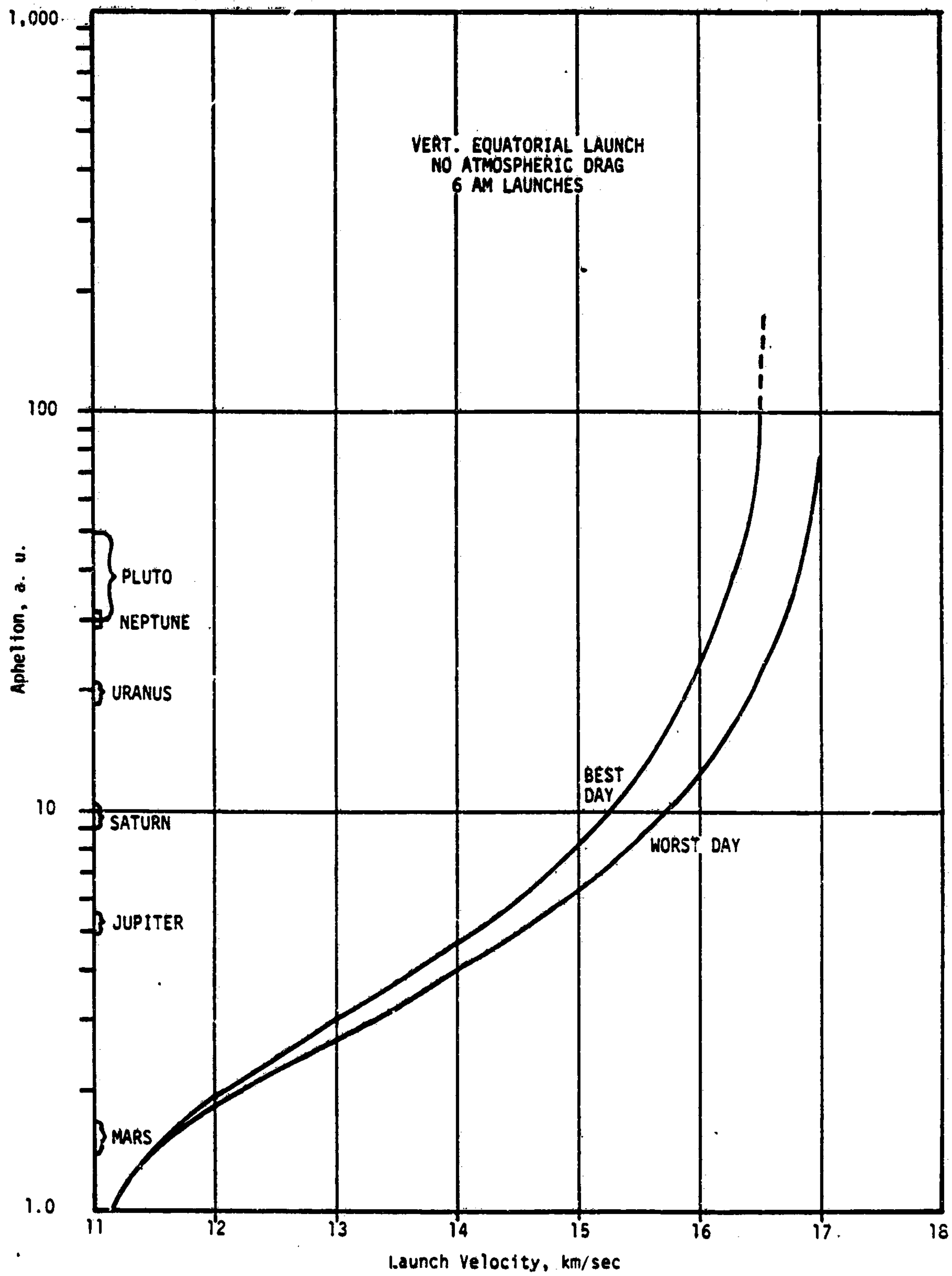


FIGURE 3-20. SOLAR SYSTEM PROBES--MAXIMUM DISTANCE FROM SUN

for the launch velocities indicated. For destinations near the orbits of Saturn and Uranus, the flight times would be about 4 and 11 years, respectively.

3.5 Analysis of Requirements for Launching Payloads to Earth Orbit

3.5.1 Apogee Motor Impulse Requirements

The use of EML concepts for placing payloads in circular equatorial orbits was investigated parametrically to determine the launch velocities and apogee motor impulse requirements as functions of orbit altitude and launch angle from the horizontal.

Figure 3-21 summarizes the results. For simplicity, and in keeping with the exploratory nature of the study, atmospheric drag was neglected in computing the launch velocities.

In Figure 3-21, the solid lines are contours of constant launch velocity (provided by the EML) and the dashed lines are contours of constant apogee motor impulse required to circularize the orbit (provided by an on-board propulsion system). For vertical launches to the lower altitudes, the apogee motor requirements are clearly prohibitive.

3.5.2 Range Safety Limitations

A further limitation appears if range safety aspects are considered. The payload must be protected at launch by a rather massive shield. The shield will then separate from the payload and will remain in the transfer orbit. It will then return to Earth and impact at a location dependent upon the apogee altitude and the launch angle.

The longitude of the shield impact point, relative to the launch site, is shown in Figure 3-22. For vertical launches, the shield would always appear to draft toward the west, to an observer at the launch site.

Figure 3-23 displays the same information, but specialized for an equatorial launch site on the east coast of South America. For vertical launches, the shield would fall on the South American continent for all apogee altitudes up to about 10,500 km. For higher apogees, shield impact would be in the Pacific Ocean. For this particular site, it appears that launches could be made to any altitude (up to geosynchronous altitude), without impacting any land mass, only for a launch angle of about 55 degrees from the horizontal.

Figure 3-24 is specialized for launches from the west coast of South America. This would appear to be the most promising launch site for vertical launches, since it would always result in a shield impact in the Pacific Ocean.

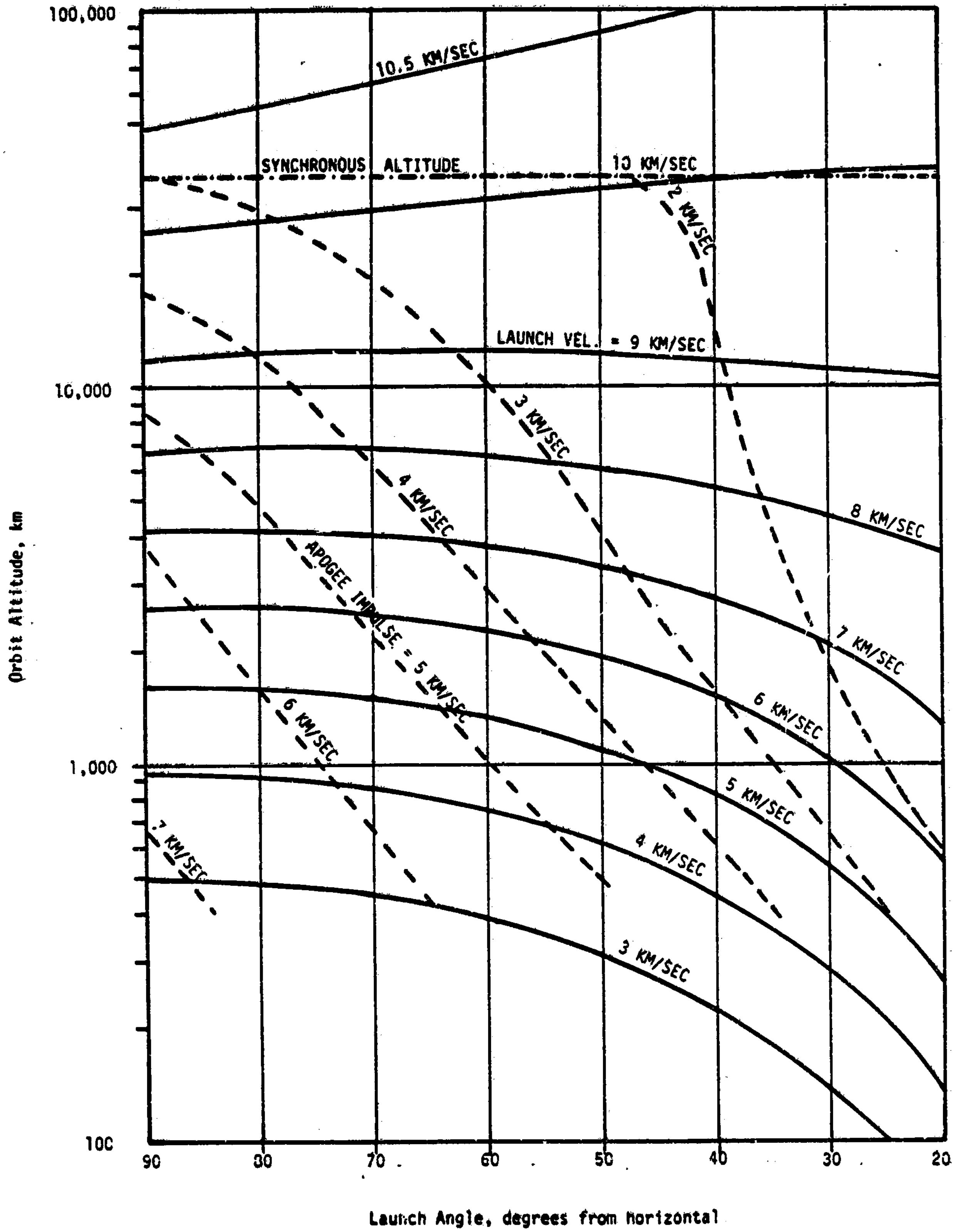


FIGURE 3-21. APOGEE MOTOR IMPULSE REQUIREMENTS FOR CIRCULAR EQUATORIAL ORBITS

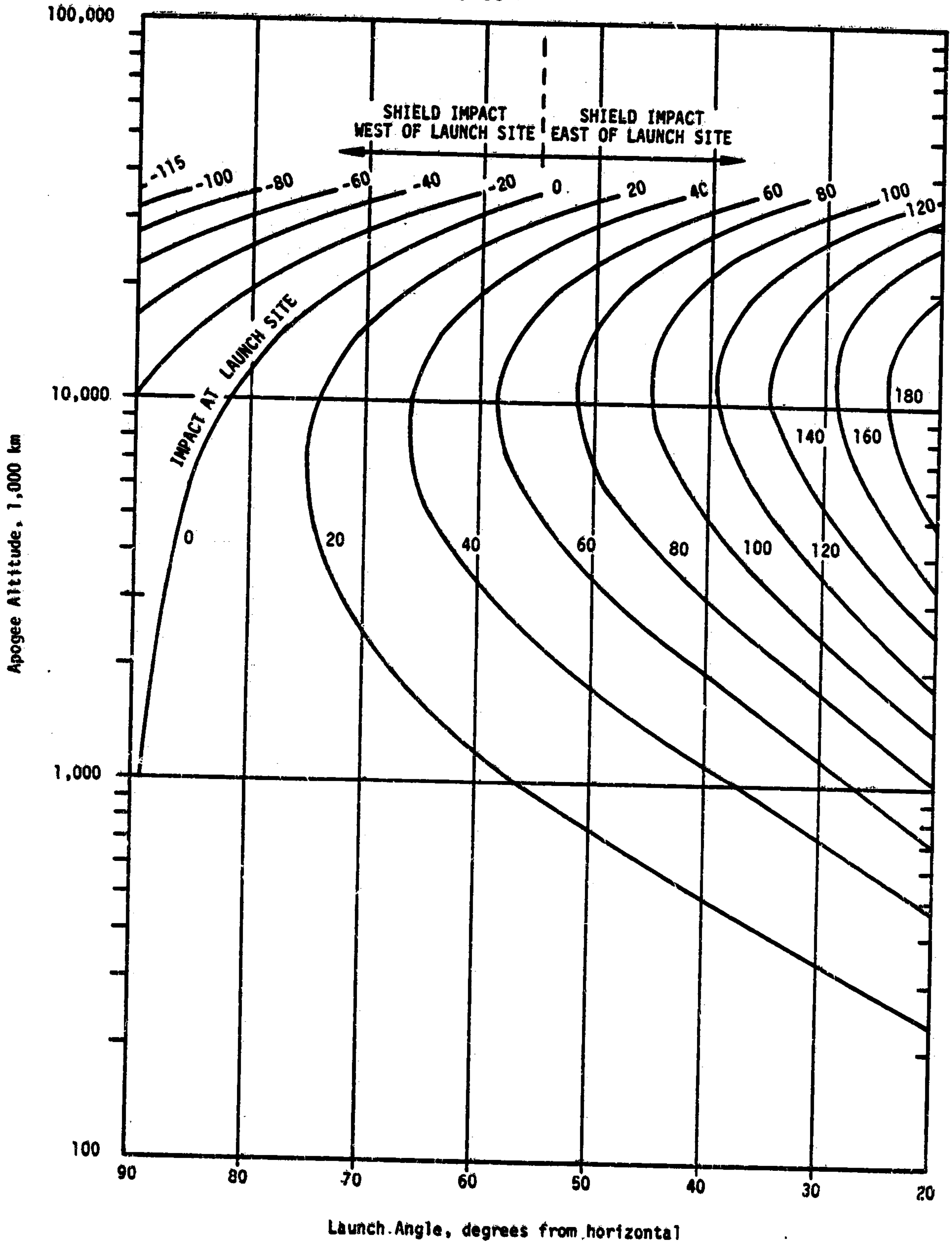


FIGURE 3-22. LONGITUDE OF SHIELD IMPACT RELATIVE TO EQUATORIAL LAUNCH SITE

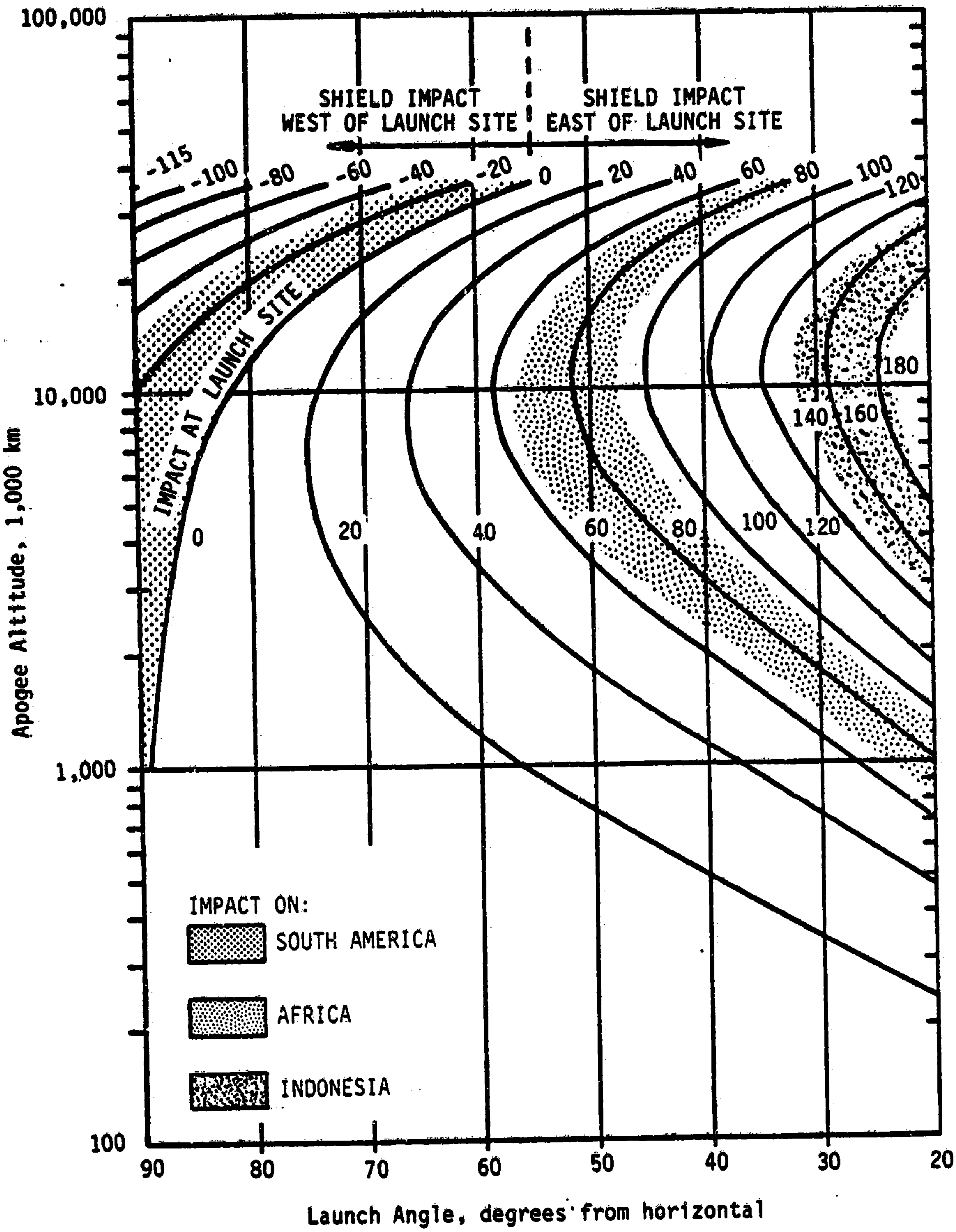


FIGURE 3-23. LONGITUDE OF SHIELD IMPACT RELATIVE TO BELEM, BRAZIL, LAUNCH SITE

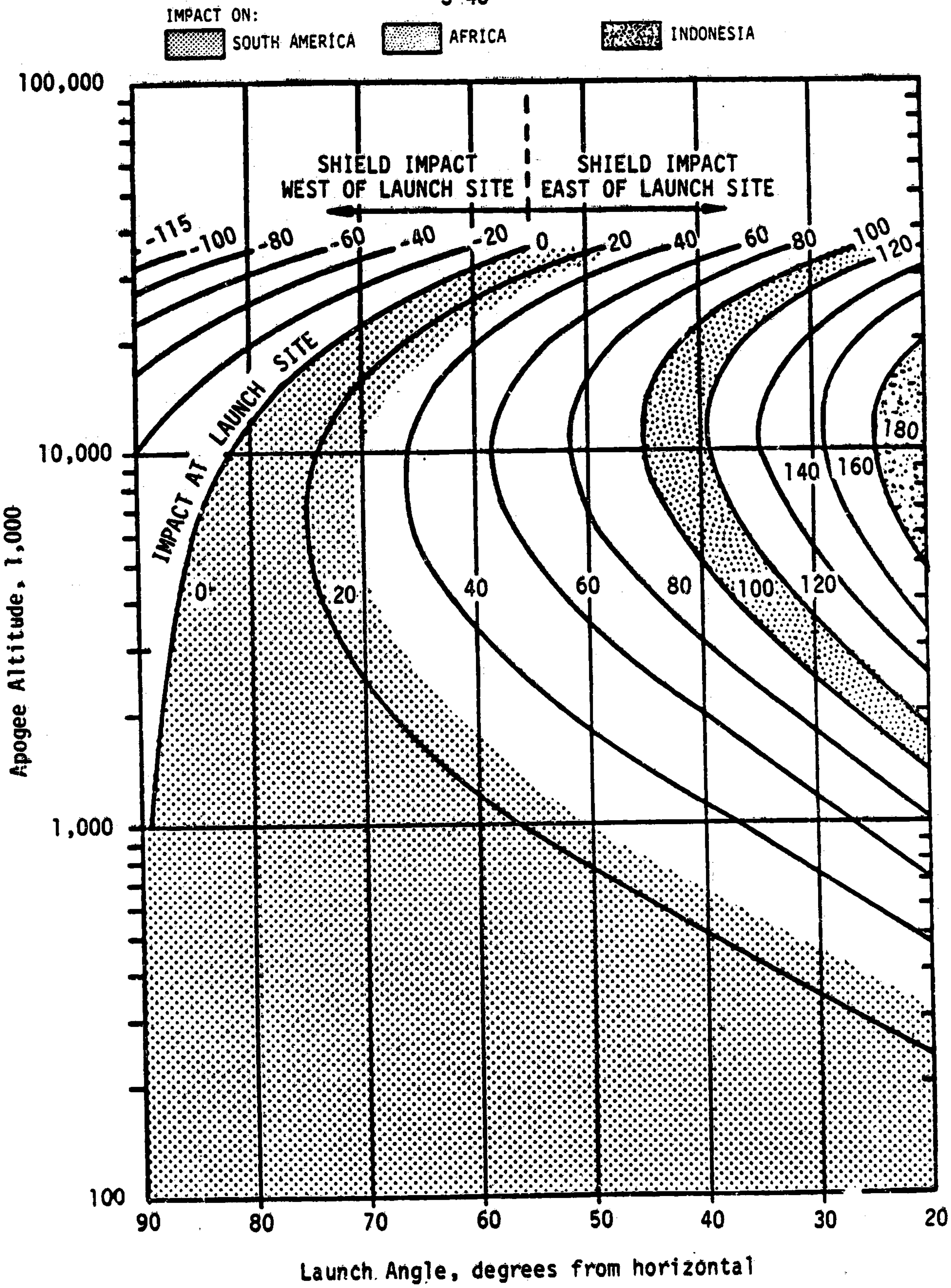


FIGURE 3-24. LONGITUDE OF SHIELD IMPACT RELATIVE TO QUITO, EQUADOR, LAUNCH SITE

3.6 Lunar Base Resupply Mission Requirements for Vertical Equatorial EML Launches

The impulsive velocity requirements for the lunar base resupply mission have been computed for vertical EML launches from the equator. The atmospheric drag penalty has not been considered, but could be included as a multiplying factor based on the projectile ballistic coefficient as discussed in Section 3.1.4.3 of Rice, Miller, and Earhart (1982).

For this analysis, the Earth-Moon trajectory data were obtained from direct numerical integration of the three-dimensional equations of motion of the projectile, under the simultaneous influence of both the Earth and the Moon. In contrast, the retro-impulse maneuvers for lunar capture or landing were computed by considering only the gravitational attraction of the Moon itself.

3.6.1 Frequency of Launch Opportunities

For vertical equatorial launches, the projectile trajectory will lies in the equatorial plane, except for small out-of-plane perturbations in the near-vicinity of the Moon. Consequently, the arrival of the projectile at the Moon must occur when the Moon is very close to the intersection of the plane of the Moon's orbit and the Earth's equatorial plane. Since the Moon passes through the equatorial plane twice a month, there will be two opportunities per month for a projectile fired from a fixed launcher.

3.6.2 Launch Velocity Requirements

Because of the distance to the Moon, the minimum launch velocity for lunar capture is only slightly less than Earth-escape velocity. The distance to the Moon varies from about 356,400 km to 406,700 km. While this variation has only a slight effect on the minimum launch velocity requirement, it has a pronounced effect on flight time.

Figure 3-25 illustrates the relationships between launch velocity, flight time, and the impact velocity on the lunar surface for direct-impact flights. As seen in the figure, the flight time is most sensitive to variations in the launch velocity when the launch velocity is near the minimum value. The use of these lower-energy trajectories would not be appropriate for EML launches because of the flight time sensitivity to launch velocity uncertainties. The time of flight must be known with precision to launch at the appropriate time to ensure the simultaneous arrival of the projectile and the Moon at the line of intersection of lunar plane and the equatorial plane.

As seen in Figure 3-25, the launch velocity requirements for lower flight times (and lower flight time sensitivity) are only slightly larger than the minimum. A greater penalty is suffered in the impact velocity as the flight time is reduced. The variation in the Earth-Moon distance has a pronounced effect on the flight time, but a negligible effect on the impact velocity.

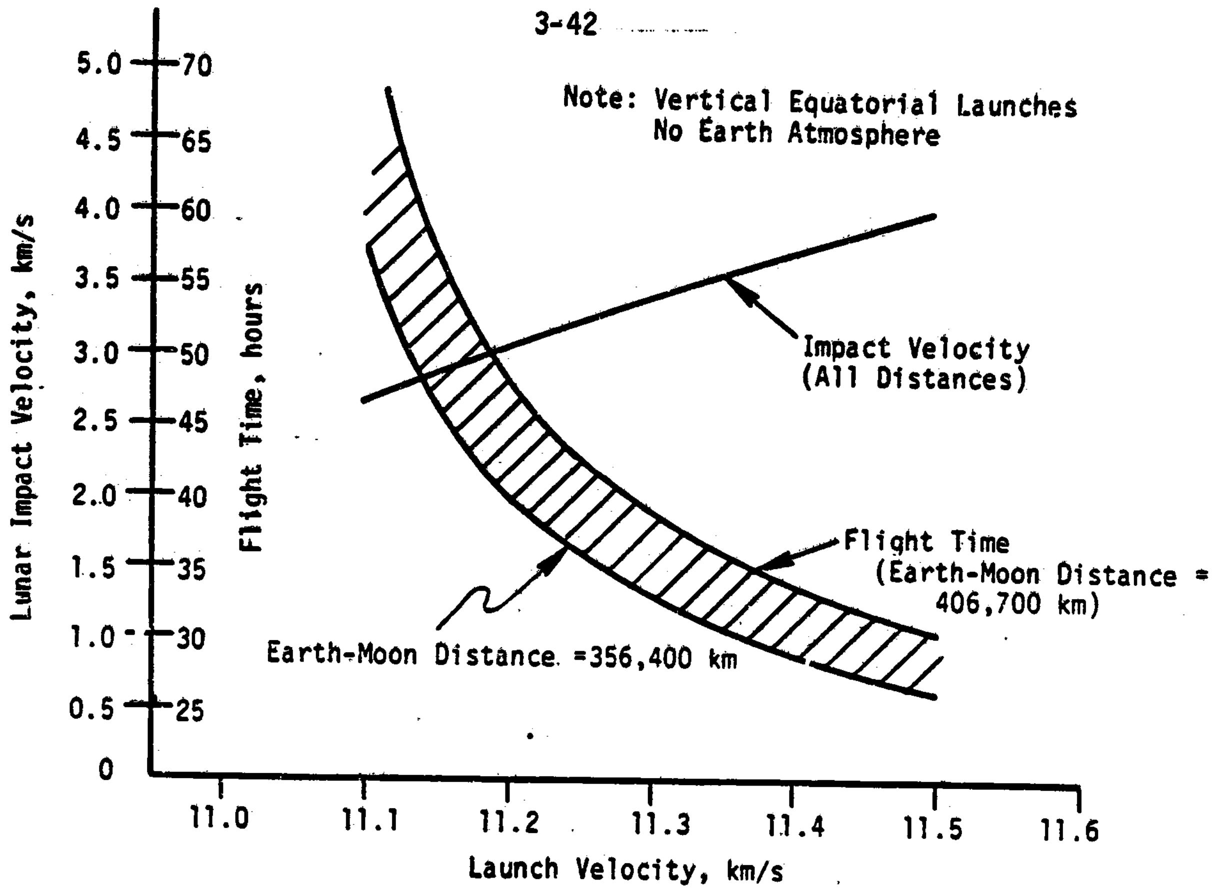


FIGURE 3-25. FLIGHT TIME AND IMPACT VELOCITY VERSUS LAUNCH VELOCITY

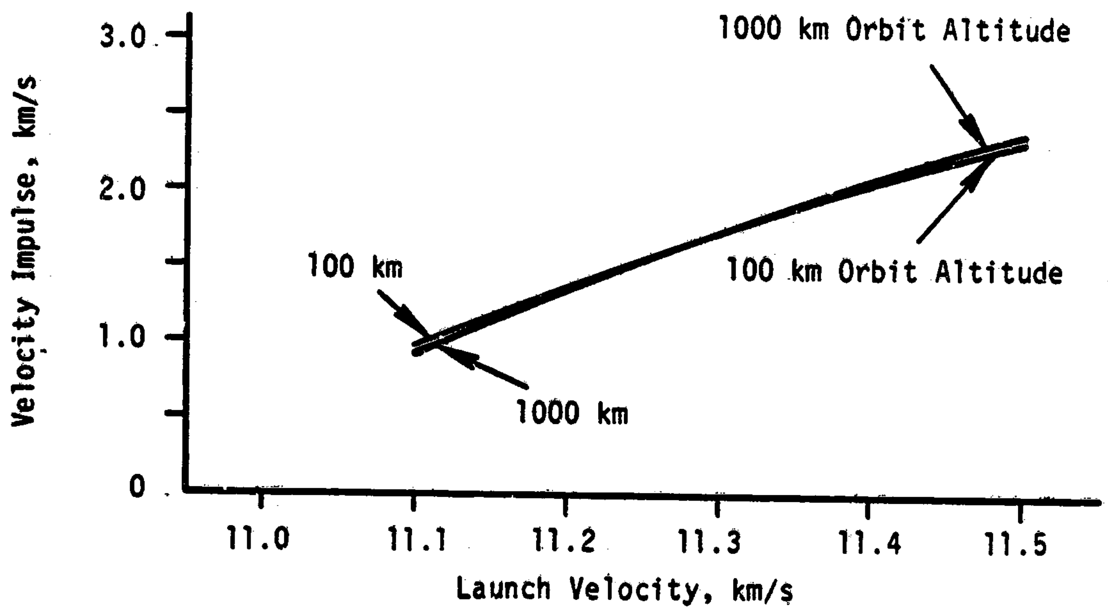


FIGURE 3-26. VELOCITY IMPULSE REQUIRED TO ESTABLISH
CIRCULAR PARKING ORBIT ABOUT MOON

Another parameter which was investigated is the dihedral angle between the orbit plane of the Moon and the plane of the Earth's equator. This angle varies between about 18.8 and 28.59 degrees, but the effect was found to be insignificant for the vertical equatorial launches. All data shown are for the larger angle, but the effect of a smaller angle would not be discernible in the plots.

3.6.3 Lunar Landing Retro Requirements

For direct descent to the lunar surface, the impact velocities shown in Figure 3-25 are roughly equal to the retro velocity changes required for soft landings. The actual requirements would be somewhat larger, but the difference would be small for high-acceleration burns close to the lunar surface.

An alternative scheme would invoke the use of an intermediate parking orbit about the Moon. Although the parking orbit mode will require a greater total retro capability than the direct descent mode, the intermediate orbit may be a practical necessity to achieve a precise landing at the site of the lunar base.

Figure 3-26 shows the retro impulse required to establish a circular orbit about the Moon. For a reasonable range of orbit altitudes, the retro requirement is almost independent of orbit altitude, but depends primarily on approach velocity to the Moon which, in turn, is a function of launch velocity. The variation of the Earth-Moon distance is not significant.

To achieve a landing from the parking orbit, two techniques could be used. In the first, one retro impulse would be used to leave the parking orbit and the projectile would be permitted to impact the lunar surface in a hard landing. Figure 3-27 shows the relationship between the de-orbit impulse and the resulting velocity at the surface. The minimum de-orbit impulse would produce a grazing, horizontal impact while the maximum impulse would permit the projectile to fall vertically to the lunar surface.

For a soft landing, the alternative scheme would use an additional retro burn just prior to impact. The magnitude of this third retro impulse is approximated by the velocity at the surface as shown in Figure 3-27.

The total retro requirements for a soft landing are summarized in Figure 3-28 as a function of initial launch velocity at the Earth, the parking orbit altitude, and the relative magnitude of the de-orbit impulse used to depart the parking orbit. For each orbit altitude, the upper limit of the band represents the vertical fall to the lunar surface (maximum de-orbit impulse), while the lower bound corresponds to the grazing approach. The accuracy of the landing would be greatly enhanced by using a steep descent to the surface.

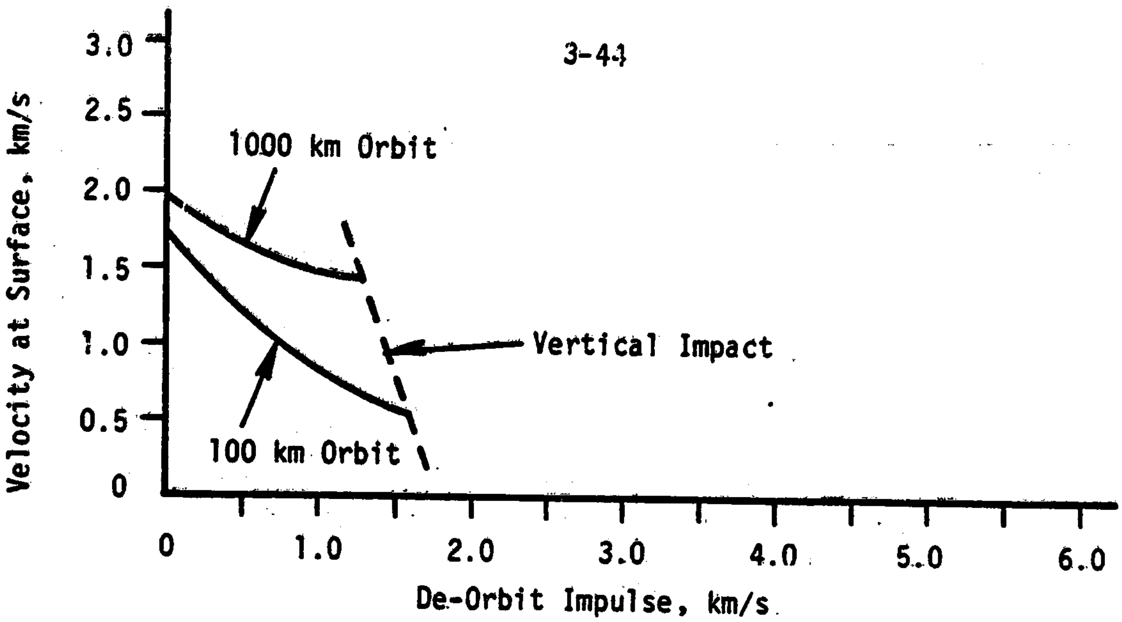


FIGURE 3-27. VELOCITY AT LUNAR SURFACE VERSUS DE-ORBIT IMPULSE

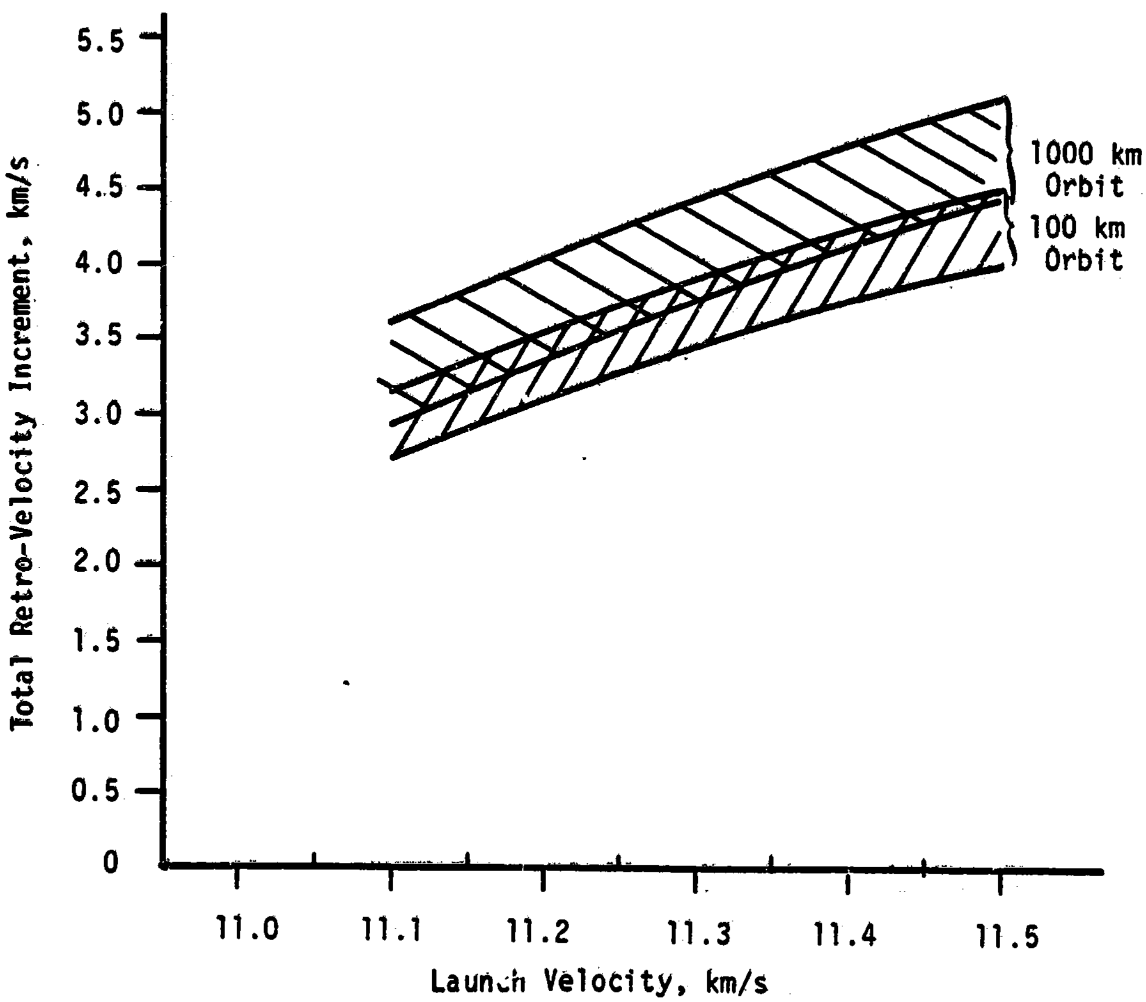


FIGURE 3-28. TOTAL RETRO IMPULSE FOR SOFT LUNAR LANDING

3.6.4 Launch Windows

As mentioned in Section 3.6.1, there will be two occurrences each month in which the Moon passes through the Earth's equatorial plane. In each event, the establishment of a satisfactory trajectory demands close control of two angles, measured at the time of launch.

The first angle is measured in the plane of the Moon's orbit, and specifies the position of the Moon, at launch time, from the line of intersection with the Earth's equatorial plane. The desired angle is directly proportional to the flight time and depends, therefore, on the launch velocity.

The second control angle is measured in the Earth's equatorial plane, and corresponds to the latitude of the launch site, at launch time, relative to the line of intersection with the lunar orbit plane. This angle is related to the time of day, whereas the former corresponds to the day of the month.

The second angle (relative to time of day) is more critical than the first, in that the Earth rotates about 15 degrees per hour and the tolerance is relatively small. In fact, the launch window could be as small as 4½ minutes without midcourse correction, even if it is assumed that the launch velocity could be adjusted precisely to satisfy the flight time requirement imposed by the first angle at the time of launch.

Since the launch window appeared to be prohibitively small, a brief analysis was conducted to explore the possibility of expanding the window by the use of a midcourse velocity impulse.

For the computations, the launch velocity (without atmospheric loss correction) was fixed at 11.4 km/sec, the minimum distance to the Moon (356,000 km) was used, and the angle between the lunar orbit plane and the Earth's equator was set at the maximum value of 28.59 degrees. To further reduce the number of free parameters, the midcourse correction was applied 5 hours after launch in all cases. The nominal flight time for these conditions is about 28.5 hours, so the midcourse increment was applied relatively early in the trajectory. In general, the earlier the correction, the smaller the velocity requirement but the greater the required precision.

The results of the analysis are summarized in Table 3-7. A midcourse correction of 0.92 km/s expands the launch window to six one-hour opportunities per month (two opportunities of three days length). The midcourse ΔV of 0.92 km/s is in addition to the 3.5 to 5.0 km/s lunar landing requirement.

TABLE 3-7. REPRESENTATIVE MIDCOURSE VELOCITY INCREMENT REQUIREMENTS

Window Width (hours) Each Day	Consecutive No. of Days	Required Increment (km/sec)
0.5	1	0.37
0.5	2	0.55
0.5	3	0.76
1.0	1	0.61
1.0	2	0.75
1.0	3	0.92

3.6.5 Conclusions

The following conclusions are drawn for vertical equatorial launches to the Moon:

(1) With a midcourse correction of 0.92 km/s, six launch windows of one-hour length are available each month for launching to the Moon. Other windows of various durations are available by changing the midcourse correction capability.

(2) The launch velocity requirement, ignoring atmospheric losses, ranges upward from about 11.1 km/s.

(3) A launch velocity of about 11.3 km/s (no drag penalty included) would be appropriate to reduce the sensitivity of the flight time to uncertainties in the launch velocity. Assuming a ballistic coefficient of 93,000 kg/m^2, the launch velocity accounting for atmospheric drag becomes 11.94 km/s.

(4) For practical launch velocities, and for direct impact hard landings, impact velocities of 3.5 to 4 km/s would be experienced.

(5) For soft lunar landings, the total retro impulse requirements could be in the 4.5 to 5 km/s range if an intermediate lunar parking orbit is used. The parking orbit may be a practical necessity for accurate placement of the payload on the lunar surface.

(6) For a lunar-orbit destination, the total impulse requirements could be in the 1.5 to 2.5 km/s range.

3.7 Earth-to-Geosynchronous Orbit Launch Velocity Requirements

This section presents the launch velocity requirements for placing a projectile launched from an Earth-based EML into geosynchronous orbit (GEO). The basic approach used to develop these velocity requirements is outlined in Section 3.1.3.2 of the previous ESRL Final Report (Rice, et al, 1982). This approach incorporates the following assumptions:

- Due-east launch from the equator
- Orbit altitude of 35,800 km with 0 degrees inclination.

Using the laws of conservation of energy and conservation of angular momentum, launch velocities and angles were calculated for several trajectory angles at GEO altitude. These numbers were then corrected for the rotation of the Earth and for atmospheric drag. The results are shown in Figure 3-29. Launch velocities and corresponding angles from horizontal (θ_L) are drawn for several trajectory angles at GEO altitude (θ) measured from the local horizontal. For the recommended configuration having a launch angle of 20 degrees from the horizontal, the resulting launch velocity is 11.7 km/s.

A propulsion system is necessary to circularize the projectile into the required orbit. The law of cosines determines the necessary velocity increment of the propulsion system aboard the projectile. Figure 3-30 shows the velocity increment (ΔV) necessary for the circularization as a function of projectile velocity at GEO altitude (35,800 km). With a launch velocity of 11.7 km/s and launch angle of 20 degrees, the projectile velocity at GEO altitude is determined by the method described above and found to be 1.4 km/s. This corresponds to a velocity increment of 1.7 km/s for which the on-board propulsion system must be sized.

3.8 Projectile Concepts

To properly study certain missions, useful payloads must be defined. The projectile concepts for each of the four reference concepts are defined in this section. Also, two other revised projectile concepts are derived in this section. The TRU waste projectile is derived from the ESRL Mission A projectile, while GEO payloads are found by revising the Earth-orbital projectile.

3.8.1 LEO Projectiles

The basic requirements for the Earth-orbital projectiles were defined in Rice, et al (1982) and are summarized here:

- Maximum payload = 650 kg
- Propellants consist of hydrazine and chlorine trifluoride
- N_2H_4 = 300 kg
- ClF_3 = 850 kg
- Dry propulsion system = 450 kg
- Instruments, ACS, and astrionics = 105 kg.

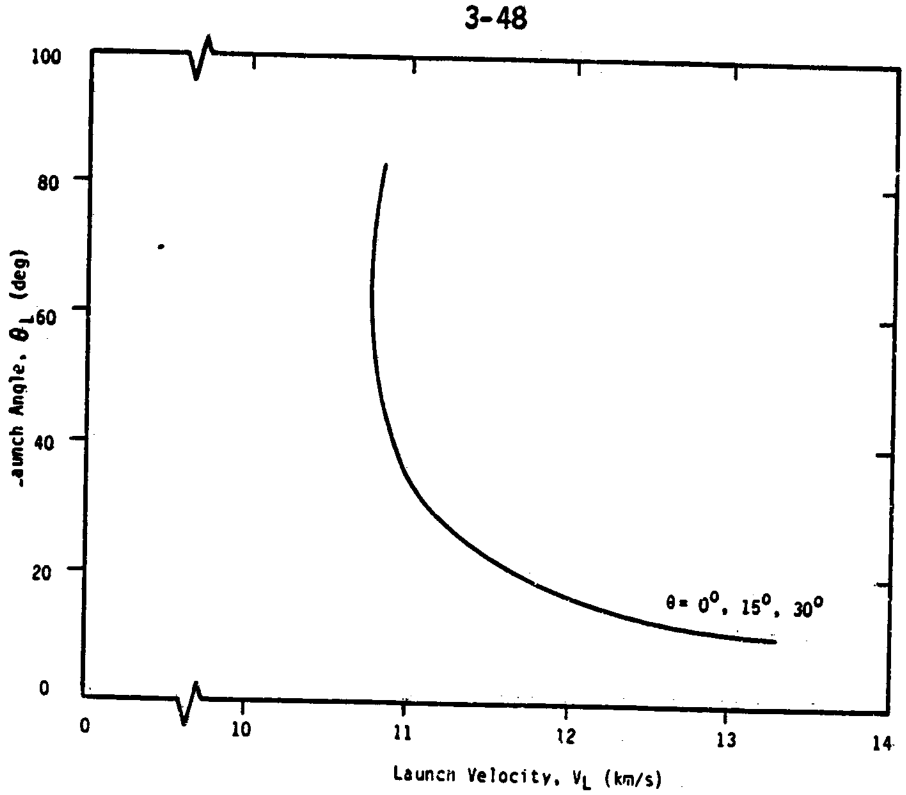


FIGURE 3-29. LAUNCH VELOCITY AND ANGLE REQUIREMENTS FOR 500-km CIRCULAR ORBIT

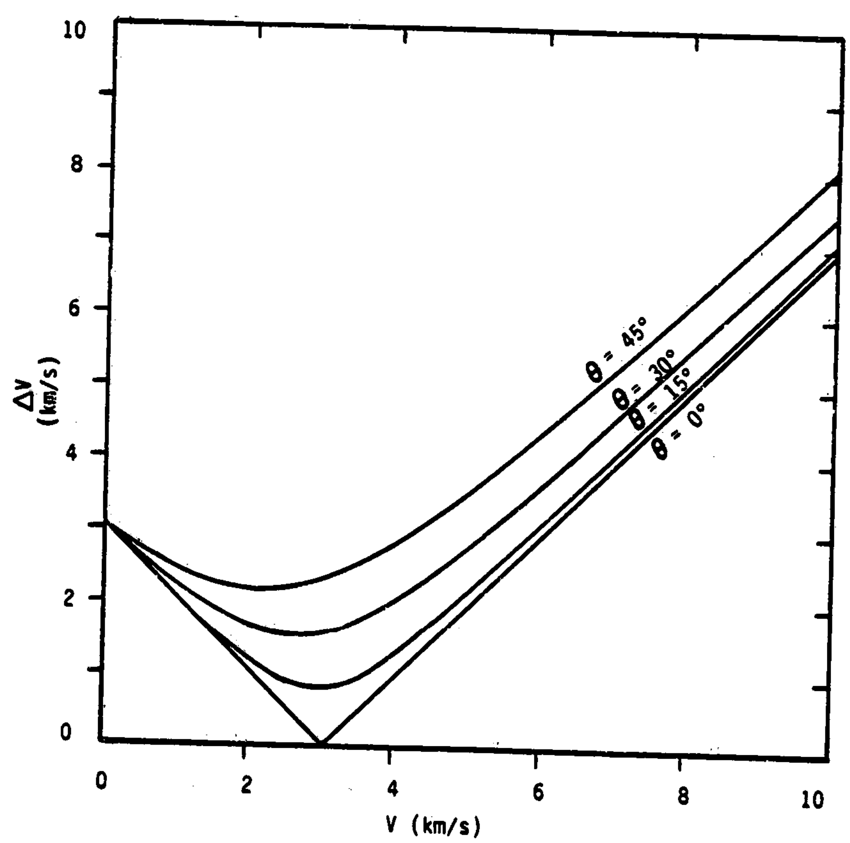


FIGURE 3-30. DELTA-V REQUIREMENTS FOR 500-km CIRCULAR ORBIT

This means that a minimum of 2300 kg must be launched. Remaining projectile elements would include structure, thermal protection, nose cone, and fins, as well as projectile coils for coaxial launches and sabots for railgun launches.

Figures 3-31 and 3-31 illustrate the Earth-orbital projectile concepts used in this study. The LEO projectiles are more fully described in Sections 4.2.2.1.3 and 4.3.2.1.3.

3.8.2 Hybrid EML/Rocket Projectiles

A useful payload of 800 kg would be available for a three-stage solid-rocket vehicle launched at 2 km/s by an EML. The total launched mass for this payload requirement would be approximately 15,000 kg. Section 3.1.4 described the methods used to select the vehicle mass.

Besides the payload, other projectile elements would include structure, three solid-propellant stages, nose cone/shroud, stabilization fins, projectile coils (for coaxial system), and rear sabot/armature (for railgun system). The two hybrid EML/rocket projectile concepts are illustrated in Figures 3-33 and 3-34. Sections 4.4.2.1 and 4.5.2.1 describe the projectiles in more detail.

3.8.3 Lunar-Supply Projectile

The Earth-to-orbit projectile (Section 3.8.1) was used as the baseline projectile to estimate payload mass for the lunar supply mission. Available payload and projectile mass is 2300 kg. Figure 3-35 indicates the fraction of the available mass which is payload mass for different values of V/I and propellant mass fraction f. The figure was plotted using the following equations:

$$\Delta V = I \ln \frac{m_{pl} + m_{ps} + m_p}{m_{pl} + m_{ps}}$$

and

$$f = \frac{m_p}{m_p + m_{ps}}$$

where

ΔV = required velocity increment, in m/s
I = specific impulse, in m/s
m_{pl} = payload mass, in kg
m_p = propellant mass, in kg
m_{ps} = propulsion system (dry) mass, in kg.

Approximately 4.5 km/s must be supplied by the projectile to soft-land on the Moon (3.5 km/s for lunar landing and 1.0 km/s for midcourse correction). If the desired payload destination is lunar orbit, instead of lunar impact, the propulsion system must supply about 2.75 km/s. From Figure 3-34, assuming an Earth-storable propulsion system with a specific impulse of 310 s (3000 m/s), the following table shows maximum payload mass for different mass fractions with an available (payload and propulsion system) mass of 2300 kg.

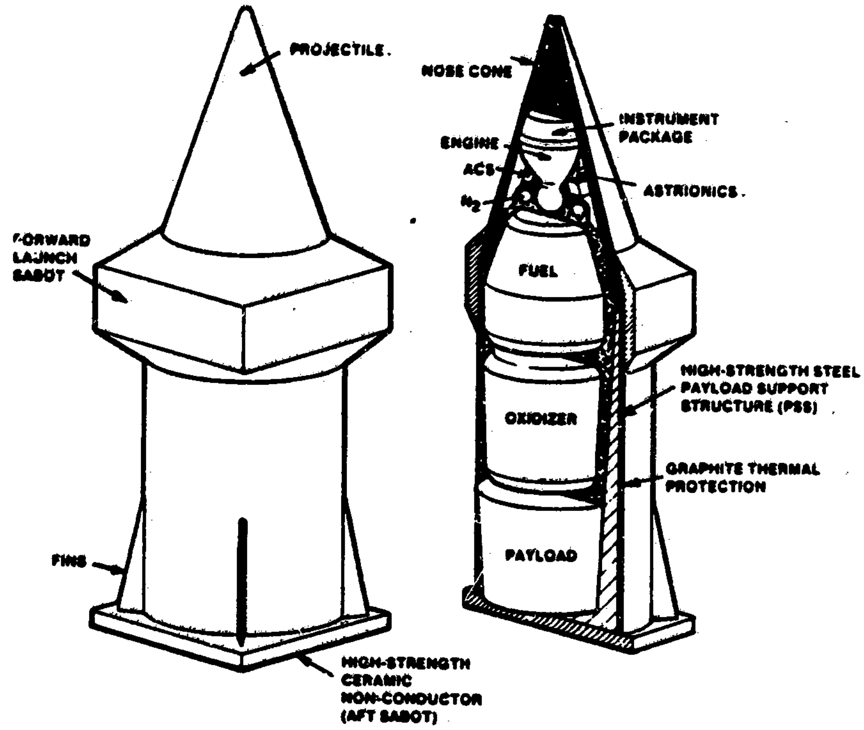


FIGURE 3-31. EARTH-ORBITAL PROJECTILE (RAILGUN)

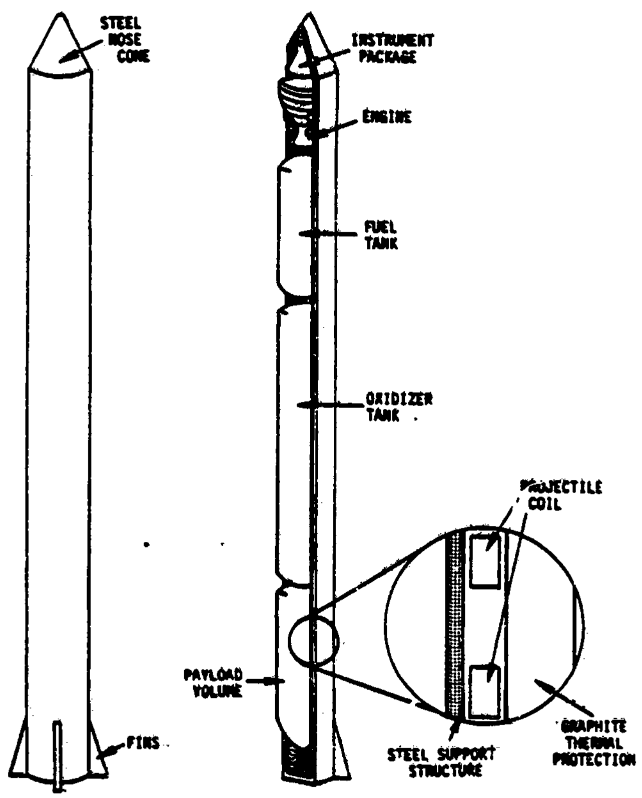


FIGURE 3-32. EARTH-ORBITAL PROJECTILE (COAXIAL)

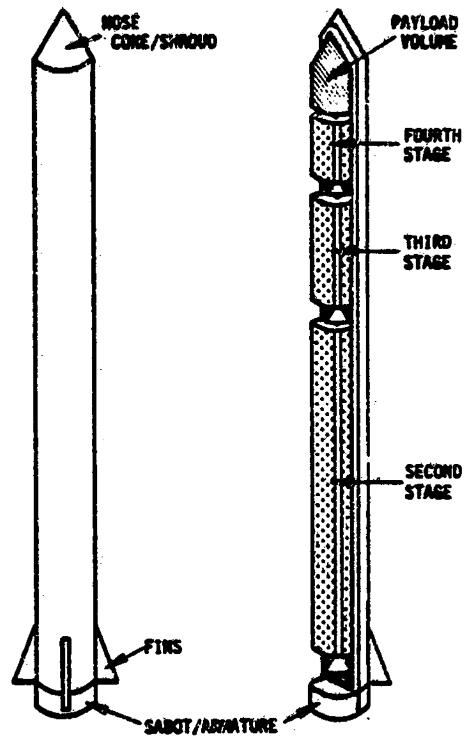


FIGURE 3-33. HYBRID RAILGUN/ROCKET PROJECTILE

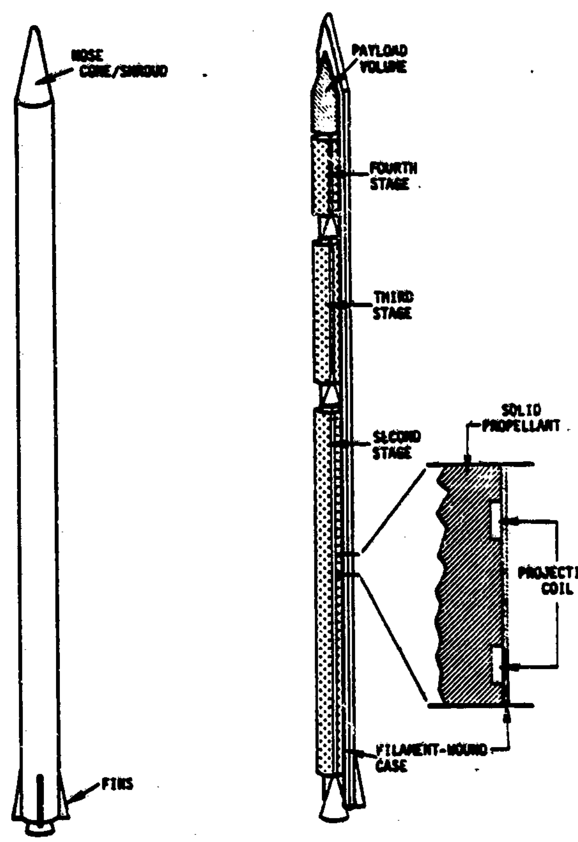


FIGURE 3-34. HYBRID COAXIAL/ROCKET PROJECTILE

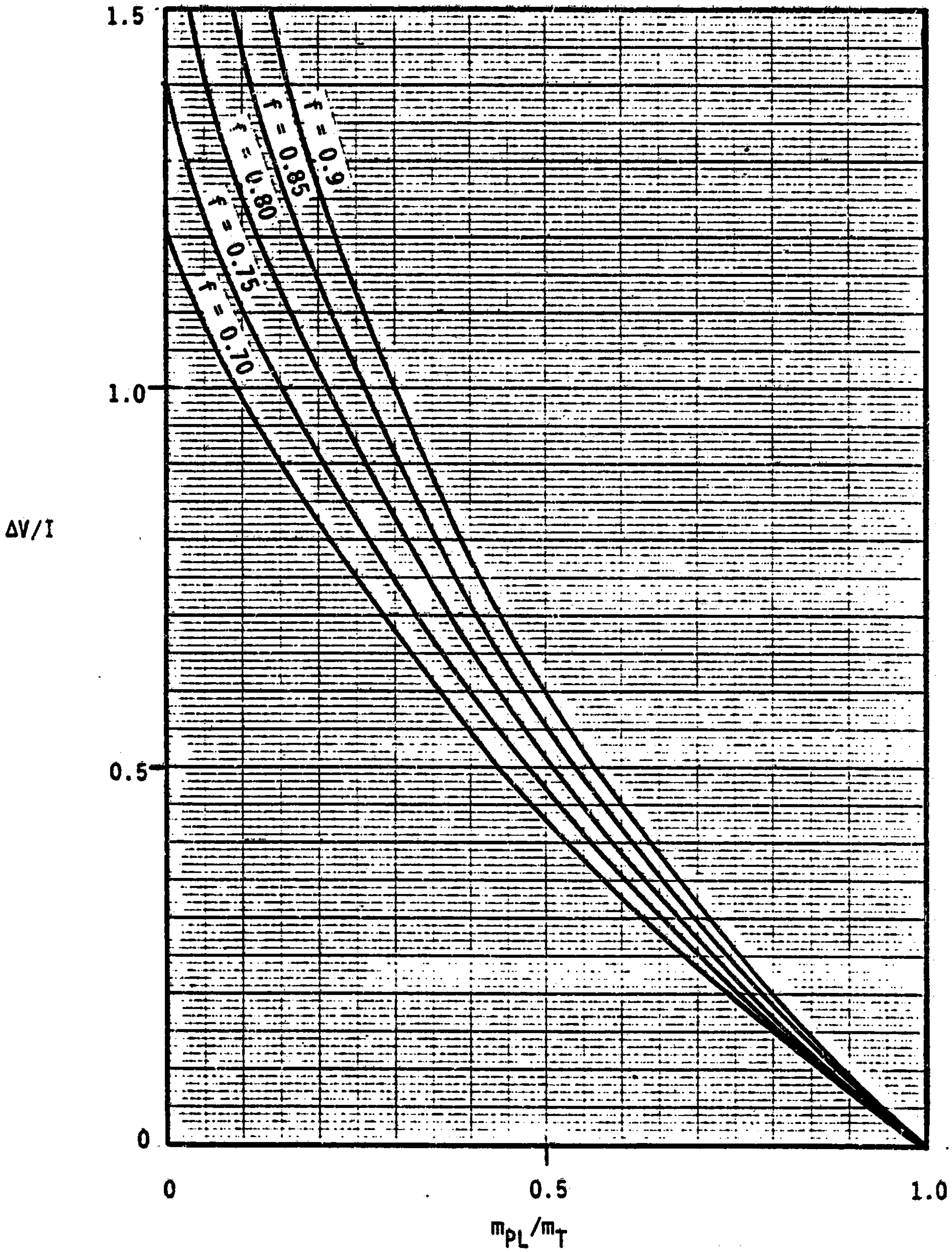


FIGURE 3-35. PAYLOAD MASS AS A FUNCTION OF ΔV/I AND PROPULSION SYSTEM MASS FRACTION

Mass Fraction, f	Payload (kg)	
	Lunar Impact	Lunar Orbit
0.7	--	325
0.75	--	460
0.8	70	565

Lunar orbit insertion appears to be the most feasible mission option. If a mass fraction of 0.75 is selected, the mass breakdown then becomes:

- Payload = 460 kg
- Propellants = 1340 kg
- Dry propulsion system = 460 kg.

The remaining projectile elements are identical to those defined for the Earth-to-orbit projectiles, which were designed for accelerations of 2500 g's.

3.8.4 TRU Waste Projectile

The ESRL Mission A projectile (Rice, et al, 1982) was used as a baseline for determining the useful payload for a TRU waste launch. Treated TRU wastes are less dense than HLW in cermet form (2.03 g/cm^3 vs. 6.5 g/cm^3) so the original projectile must be lengthened to have a meaningful payload. At the same time, however, TRU wastes have less stringent shielding requirements, so the shield thickness may be reduced, leaving more volume available for the TRU waste payload.

The revised projectile was calculated to have the same diameter as the HLW projectile (51 cm), but is lengthened by 40 cm. The radiation shielding is reduced by half to 6 cm. With these alterations, the TRU waste payload is found to be 285 kg per projectile.

3.8.5 GEO Projectile

A launch from Earth to geosynchronous orbit requires less orbit-circularization propulsion than a launch from Earth to 500 km because of smaller orbital velocities (3.1 km/s compared with 7.6 km/s). This directly results in a larger GEO payload capability for the same projectile mass. The payload capability was calculated as follows. Using the same basic projectile that was defined for the ESRL Study (see Section 3.1.5.2 of Rice, et al, 1982), 2300 kg is again available for the projectile's payload and propulsion system. From Section 3.7, the required propulsion system ΔV is 1.7 km/s. The specific impulse is assumed to be 3000 m/s, with the following system definitions:

- ClF_3 and N_2H_4 propellants
- Oxidizer-to-fuel ratio of 2.8

- Nozzle expansion ratio of 14.0
- Chamber pressure of 100 N/cm^2 (150 psi).

The propellant requirement was determined from

$$\Delta V = I \ln \left(\frac{m_t}{m_t - m_{prop}} \right)$$

where

m_t = $m_{ps} + m_{prop} + m_{pl}$
m_{ps} = mass of dry propulsion system
m_{prop} = mass of propellants
m_{pl} = payload mass.

The required propellant mass is then 995 kg. Thus, 735 kg of ClF_3 and 260 kg of N_2H_4 are necessary for orbit circularization. For a 50-second burn time, the thrust level is defined to be 60,000 N (14,000 lb_f).

The mass fraction of the propulsion system was assumed to be 0.7, which corresponds to a total propulsion system mass of 1420 kg and a dry system mass of 425 kg. Therefore, the resulting payload mass to GEO is then 880 kg which is 1.35 times greater than the baseline ESRL payload capability of 650 kg.

3.9 Nose Cone Material Selection

This section addresses several issues which are critical to the selection of a material for the nose cones of the EML projectiles. Two major concerns are discussed here: ablation effects and economics.

3.9.1 Ablation Considerations

If the EML projectile is to survive its flight through the atmosphere, the nose cone material must not substantially ablate away. The ablation effects are especially severe for the solar system escape and planetary missions, where launch velocities exceed 15 km/s.

The ESRL projectiles used tungsten for the nose cone material (Rice, et al, 1982). This was determined to be costly, and it was recommended that, if at all possible, steel should be used instead.

Estimates of nose cone recession due to ablation were made by three aerodynamics experts in the previous ESRL study. These estimates ranged from several centimeters to half the projectile diameter for a launch velocity of 18 km/s.

For a final honors thesis at the Ohio State University, A. Kerslake (Kerslake, 1982) investigated this problem, studying the effects of ablation on steel, tungsten, and graphite materials. Her results indicated that all three materials were acceptable for the ESRL projectile with total nose cone recession of less than 10 cm, for launch velocities

below 17 km/s. Above 18 km/s launch velocity, all of the materials show increasingly large recessions. Steel proved to be the least desirable material and graphite the most desirable.

Early in this study, it was decided to use steel for the nose cones of the EML projectiles. This decision was made with some concern regarding the accuracy of ablation estimates for the high-speed projectiles (solar system escape and planetary). As the study progressed, the reference concepts were chosen. None of these concepts used launch velocities of greater than 12 km/s. Therefore, it was decided to continue using steel as the material of choice for the projectile nose cones.

3.9.2 Economic Considerations

The tungsten nose cones for the ESRL nuclear waste disposal and Earth-orbital missions (Rice, et al, 1982) have masses of 440 kg and 1150 kg, respectively. In 1982, tungsten metal cost approximately $33.00 per kilogram, and in May 1983, a pure powder from which tungsten metal would be made cost $11.14 per pound giving an implicit price of approximately $25.00 per kilogram for tungsten metal. The nuclear waste disposal nose cone was estimated to cost in the range of $30K to $80K (nominal of $50K) based upon labor factors of 2, 3.5, and 5 to manufacture the nose cone. The Earth-orbital nose cone was costed at $40K to $250K (nominal $185K) with similar factors. The low estimate was then adjusted to be slightly more than the materials cost of $38K under the assumption that additional uses would be found for the metal, including potential reuse.

Since the drop in the price of tungsten is indicative of the state of the economy rather than in new production techniques, there is no basis for major revisions to the estimates for tungsten nose cones.

If a high strength, low alloy, structural shaped steel is adequate for Earth orbital missions, it would have a significant materials cost advantage ($0.239 per pound or $0.55 per kilogram). In addition, it is more easily worked than tungsten. A steel nose cone with a mass of 1150 kg has a materials cost of $632.50. If it were batch-produced to 0.01-cm tolerances, it quite likely would cost under $1,000 and almost certainly under $2,500. If a stainless or other high-alloy steel were required, the price variance would be great. The lowest-priced stainless steel is currently approximately $1.25 per kilogram and T-15 high-speed tool steel is quoted at $21.65 to $25.57 per kilogram. Thus, T-15 steel costs approximately the same as tungsten.

It is difficult to determine costs for carbon-carbon graphitic materials under consideration for the nuclear waste disposal and other Earth-escape missions in the time under consideration (beyond the year 2000). At the present time, carbon-carbon graphitics can cost thousands of dollars per kilogram in final shapes. The cost lies not in the raw materials (which are comparable to low-priced steels), but in the processing and quality control labor necessary to make the final product.

To achieve the ultimate properties associated with advanced carbon-carbon materials, it is necessary to control the quality of the raw materials from which the fibers are made, and then to control, at each step, the spinning of fibers, weaving, shaping, pyrolizing, and final finishing. In the time under consideration, it is reasonable to expect that carbon-carbon graphitics, or other materials with similar properties, will be available at prices less than tungsten for equivalent applications. Thus, carbon-carbon graphitic nose cones are estimated to cost about the same as tungsten nose cones.

3.10 Projectile Stress Analysis

Calculations were made to determine the strength and stability of the ESRL Earth-orbital projectile. The compressive strength and the buckling stability were analyzed.

The critical cross-section of the payload support structure (PSS) for the compression and buckling analysis was selected as the PSS was just below the lower end of the forward sabot. The carbon-carbon thermal protection and the oxidizer tank wall were assumed to be nonstructural (a conservative assumption). The applied force across this section was calculated by summing the individual component masses ahead of the critical cross-section (instruments, astrionics, ACS, propulsion system with propellants, nose cone, and forward sabot, which total 2950 kg) and multiplying by a 2500-g acceleration. This results in an applied load of 71×10^6 N. The applied axial compressive stress is calculated as:

$$\sigma_c = \frac{F}{A}$$

where F is the applied force and A is the area over which the force is applied. The applied compressive stress was then calculated to be 8.11×10^8 N/m^2 (117,700 psi) at the critical location. This stress value is below typical yield stresses for stainless steels (typically on the order of 1.035×10^9 N/m^2); the ratio of yield-to-applied stress is then 1.3.

The critical buckling compressive stress level at the selected section was based upon Euler's column formula for a column with one fixed end:

$$\sigma_{cr} = \frac{\pi^2 E}{(L/r)^2}$$

where

$$r_x = r_y = 1/4\ (D_o^2 + D_i^2)^{1/2}.$$

E is the modulus of elasticity (for steels, $E = 2.07 \times 10^{11}$ N/m^2), r is the radius of gyration about the bending or buckling axis, D_o is the outside diameter of the PSS, and D_i is the interior diameter of the PSS.

The effective column was treated as fixed end, because of the large steel mass in the lower section of the PSS. The critical elastic buckling stress was computed to be 15.9×10^9 N/m^2 (2.31×10^6 psi), assuming the Euler formula for column instability. The calculated buckling stress was found to be 19.6 times greater than the applied compressive stress.

The local buckling stress was then calculated from Shanley (1957):

$$\sigma_{cc} = \frac{K\sqrt{EE_t}}{(D_o/t)}$$

In this equation K is approximately 0.5 for this case, E_t is the tangent modulus, and t is the wall thickness. The local buckling stress was calculated to be 3.7×10^9 N/m^2 (5.4×10^5 psi), which is 4.6 times greater than the applied stress. These results indicate that the structure would be unstable at a level well above the compressive ultimate strength of most stainless steels, so buckling loads do not appear to be a problem.

It was recommended that further detailed design should include a structural analysis which would include composite structural effects, structural fastener effects, thermal effects, and pressure loading.

4.0 REFERENCE CONCEPTS

The four reference concepts are described in this section. The reference EML concepts were selected from those described in the previous section and are listed below:

- Railgun for Earth orbital launches
- Coaxial accelerator for Earth orbital launches
- Hybrid railgun/rocket launcher
- Hybrid coaxial accelerator/rocket launcher.

The concepts described in Section 3.1 are very preliminary and would require further analysis to define the optimal detailed system designs. The concepts are sufficiently detailed, however, to perform a comparative assessment based upon costs, performance, technology risk, and other factors to select the most promising concepts for further analysis.

This section is composed of five subsections. The first subsection describes the evaluation process which led to the reference concept selections. The reference concepts are detailed in the following four subsections.

4.1 Preliminary Evaluation of EML Concepts

The evaluation was based upon a combination of ten quantitative and qualitative factors. The relative importance of each of these factors was established so that weighting factors could be assigned to each. The EML concepts described in Section 3.1 were then rated for each of the screening issues. EML concepts were given scores of 1 to 3 based upon comparison with "conventional" launch methods planned for the 2020 timeframe. A score of 1 indicated the concept was deficient when compared to conventional methods. Concepts rated equal to conventional methods for a particular screening issue were given a score of 2. A score of 3 indicated that the concept was ranked better than conventional launch methods. A total score was computed for each concept, by multiplying the score for each screening criterion by its respective weighting factor and then summing across the board. The concepts were ranked according to their total scores, with selection following based upon these results.

The screening criteria and their relative weights were:

- Development cost (5)
- Operational cost (5)
- Total cost (30)
- Launcher technology risk (10)
- Payload technology risk (5)

- System flexibility (5)
- Performance (25)
- Logistics (5)
- Safety (5)
- Environmental impact (5)

Figure 4-1 displays the results for railgun concepts, while coaxial accelerator results are shown in Figure 4-2. Based upon these rankings, it was decided by Battelle and NASA/LeRC to proceed with the Earth-orbital mission and the hybrid EML/rocket concept for both the railgun and coaxial accelerator concepts. These concepts were then more fully developed, and are described in this section of the report. (The lunar base supply mission was added to the list late in the study, so it was not fully developed into a reference concept, despite its high score.)

4.2 Earth-to-Space Rail Launcher Concept

4.2.1 Concept Definition

The Earth-to-space rail launcher concept for launching cargo into low-Earth orbit consists of five major pre-launch, launch, and post-launch activities which are discussed in this section. The five activities are:

(1) Projectile/payload fabrication

(2) Surface transport of projectile/payload

(3) Projectile/payload preparation at the launch site

(4) Launch operations

(5) Trajectory monitoring.

Definitions of the individual rail launcher system elements are presented in Section 4.2.2.

4.2.1.1 Projectile/Payload Fabrication

The payload, projectile, and the orbit-circularization propulsion system are manufactured and checked-out prior to system integration and transport to the launch site.

4.2.1.2 Surface Transport of Projectile/Payload

The method of surface transportation would depend upon the selection of the launch site. If the selected site were remote, ship or air transportation would likely be used in areas where roads or railroads were inaccessible. Otherwise surface transportation would likely be accomplished using trucks or railroad cars.

MISSIONS	DEVELOPMENT COST (5)	OPERATIONAL COST (5)	TOTAL COST (30)	LAUNCHER TECHNOLOGY RISK (10)	PAYLOAD TECHNOLOGY RISK (5)	SYSTEM FLEXIBILITY (5)	PERFORMANCE (25)	LOGISTICS (5)	SAFETY (5)	ENVIRONMENTAL IMPACT (5)	TOTAL SCORE
EARTH ORBIT	1	3	3	1	1	1	3	1	3	3	240
SOLAR SYSTEM ESCAPE	2	3	3	1	1	2	1	1	2	3	195
EARTH ESCAPE	1	1	1	1	1	1	1	2	2	3	120
SUBORBITAL	1	1	1	1	1	1	3	2	2	2	165
BOOST											
- HYBRID	1	3	3	1	2	1	3	2	1	3	240
- TAV	1	3	1	1	2	1	2	1	3	3	160
SATELLITE KICK SYSTEM	1	1	1	1	1	1	2	1	1	1	125
SPACE-BASED NWDS	1	1	1	1	2	2	3	2	2	3	180
LUNAR BASE SUPPLY	1	3	3	1	1	1	3	2	3	3	245

SCREENING ISSUES

SCORES

1 Worse than Conventional Methods
2 Comparable to Conventional Methods
3 Better than Conventional Methods

FIGURE 4-1. PRELIMINARY SCREENING OF RAILGUN SYSTEMS

MISSIONS	DEVELOPMENT COST (5)	OPERATIONAL COST (5)	TOTAL COST (30)	LAUNCHER TECHNOLOGY RISK (10)	PAYLOAD TECHNOLOGY RISK (5)	SYSTEM FLEXIBILITY (5)	PERFORMANCE (25)	LOGISTICS (5)	SAFETY (5)	ENVIRONMENTAL IMPACT (5)	TOTAL SCORE
	SCREENING ISSUES										
EARTH ORBIT	1	3	3	1	1	1	3	1	3	3	240
SOLAR SYSTEM ESCAPE	2	3	3	1	1	2	3	1	2	3	245
EARTH ESCAPE	1	1	1	1	1	1	1	2	2	3	120
SUBORBITAL	1	1	1	1	1	1	3	2	2	2	165
BOOST											
- HYBRID	1	3	3	1	2	1	3	2	1	3	240
- TAV	1	3	1	1	2	1	2	1	3	3	160
SATELLITE KICK SYSTEM	1	1	1	1	1	1	2	1	1	1	125
SPACE-BASED NWDS	1	1	1	1	2	2	3	2	2	3	180
LUNAR BASE SUPPLY	1	3	3	1	1	1	3	2	3	3	245

SCORES

1 Worse than Conventional Methods
2 Comparable to Conventional Methods
3 Better than Conventional Methods

FIGURE 4-2. PRELIMINARY SCREENING OF COAXIAL EML SYSTEMS

4.2.1.3 Projectile/Payload Preparation at Launch Site

Upon arrival at the launch site, the projectile/payload would be placed in a storage facility. The projectile and payload would be removed from storage as the scheduled launch time approached. The payload would receive a prelaunch checkout, if necessary. Propellants would be loaded just prior to launch.

4.2.1.4 Launch Operations

At launch time, the projectile is loaded into the preaccelerator and all launch systems are checked out prior to launch. The launch tube is evacuated and the homopolar generators are started.

Weather and wind conditions would be checked before final countdown begins. Launch clearance would be requested from the proper authorities and all pilots in the area warned. An alarm would be sounded in the area of launch, so that all persons are cleared from a designated danger area.

When all systems are ready, the launch sequence and final countdown begins. All launch systems are computerized and fully automated. The preaccelerator is initiated, and the projectile is accelerated to 1 km/s through the preboost section. As the projectile passes through the preaccelerator into the railgun section of the launch tube, current is automatically dumped into the first rail section. A plasma armature is formed behind the projectile. Switching of the current into the segments is performed automatically as the projectile passes through each section. The projectile is accelerated to 6.85 km/s when it leaves the railgun bore. Tracking systems on the ground would be used to verify the trajectory after the projectile has left the railgun system.

4.2.1.5 Trajectory Monitoring

The projectile is tracked throughout its atmospheric flight with a small radar system located near the launch site. Before and after the orbit-circularization maneuver, the projectile's three-axis attitude control system would ensure proper projectile attitude on orbit. The auxiliary propulsion system of the projectile would provide the 2.1 km/s necessary to insert the payload into the proper orbit. The payload would be taken to a space station by the Orbital Maneuvering Vehicle (OMV), if it were available, or another system of this type.

4.2.2 System Element Definition

This section presents the Earth-to-space rail launcher system element definitions. The major system elements which are discussed here are:

- Projectile/payload characteristics
- Surface transport systems

- Launch site support facilities
- Launcher system
- Monitoring systems
- Space destination.

4.2.2.1 Projectile/Payload Characteristics

4.2.2.1.1 Payload. The payloads which are envisioned for the Earth-to-space rail launcher are bulk-type payloads to support manned orbiting space stations. These payloads include:

- Orbit Transfer Vehicle propellants
- Space station supply items
- Materials for space processing facilities.

Liquid hydrogen and liquid oxygen propellants required for Orbit Transfer Vehicles (OTVs) could be transported to space stations in the form of water. An orbital electrolyzer would then be used to transform the water payload into the oxygen and hydrogen propellants. The rail launcher system could also be used to supply other propellants to orbit. An example of this would be a water or hydrazine payload to support space station attitude control and drag make-up systems; this would however be a much smaller supply requirement than are the OTV propellants.

Space station resupply items which could be launched by a railgun system might include life support requirements (food, oxygen, and nitrogen, for example), oxygen and hydrogen (in the form of water) to supply fuel cell make-up requirements, spare parts for station maintenance and emergency repairs, and other miscellaneous supply items, such as crew personel equipment, hygiene supplies, and ship stores. The supply items must be able to withstand the high accelerations of launch (1225 g's to low-Earth orbit); other delicate materials would still require a launch by Space Shuttle.

When materials-processing facilities are operational in space, the raw materials necessary for product manufacture could be transported to the orbital facilities by the rail launcher system.

The payload was determined to have a maximum mass of 650 kg in the analysis done in the previous ESRL study. The 650 kg maximum figure is for materials with the density of aluminum (2700 kg/m^3) or greater. For payloads with the density of water (1000 kg/m^3), the corresponding mass is 320 kg due to volume constraints.

4.2.2.1.2 Propulsion System. A propulsion system is required for the Earth-orbital mission to place the payload into orbit. The propulsion system assumed here is the same one defined in the ESRL study (Rice, Miller, and Earhart, 1982). The basic characteristics of the system are summarized here. A simple hypergolic, high-propellant-density

propulsion system was indicated; hydrazine and chlorine trifluoride propellants with an oxidizer-to-fuel ratio of 2.8 were selected. Propellant mass would be approximately 1150 kg (300 kg of N_2H_4 and 850 of ClF_3) with a dry propulsion system mass of 425 kg. System attitude control and astrionics requirements amounted to 75 kg. The specific impulse of the auxiliary propulsion system was estimated at 310 s (3000 m/s).

The cold-gas attitude control system (ACS) would perform the maneuvers required to place the payload and propulsion system in the proper position for the orbital insertion burn. The propulsion system would provide the 2100 m/s necessary to circularize at 500 km altitude, after which the ACS would again be used to ensure proper attitude of the payload for rendezvous with the Orbital Maneuvering Vehicle or other vehicle used to transport the payload to the space station.

<u>4.2.2.1.3 Projectile Elements.</u> The Earth-to-space rail launcher projectile would consist of the following subsystems:

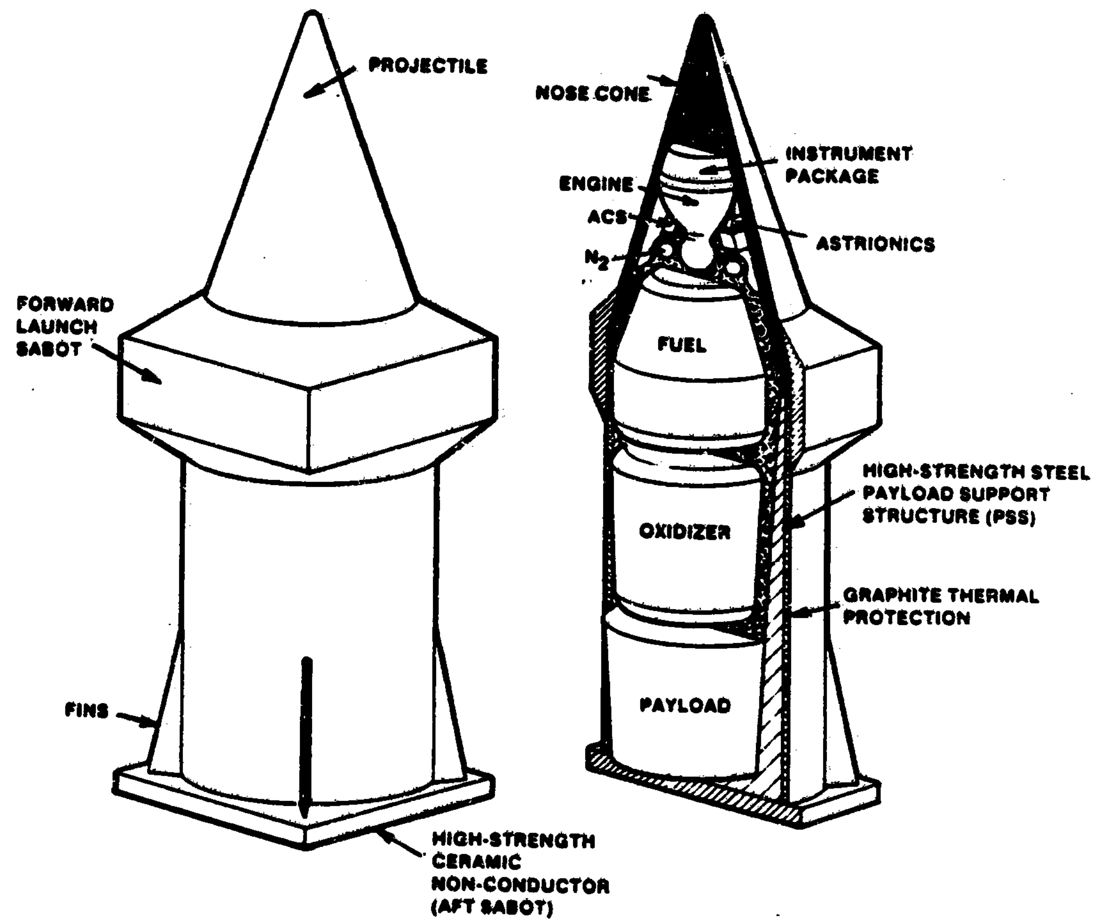
- Forward and aft sabots
- Nose cone
- Instrument package
- Liquid propulsion system, including ACS and astrionics
- Payload
- Payload support structure (PSS)
- Thermal protection system (TPS)
- Fins.

The proposed projectile is illustrated in Figure 4-3. Table 4-1 presents the projectile mass summary.

The forward and aft sabots are required to fit the round projectile to the square bore. The aft sabot also protects the rear of the projectile from excessive heating from the plasma armature. The sabots are jettisoned immediately after launch so as not to detract from the aerodynamic characteristics of the projectile. High-strength, non-conducting ceramic materials would be used to construct the sabots.

The nose cone would be constructed of steel. The tip would be slightly blunted so that the steel would evenly and smoothly melt during atmospheric flight. As discussed in Section 3.9, the amount of erosion during flight is not expected to be significant for the 7 km/s launch velocity, given the dimensions of the nose cone.

A small instrument package would be located beneath the nose cone. The package would include a radio transmitter for trajectory verification after leaving the rail launcher.



Length = 3.6 m
Projectile Diameter = 0.9 m
Sabot Diameter = 1.0 m

FIGURE 4-3. EARTH-ORBITAL PROJECTILE FOR RAIL LAUNCHER SYSTEM

TABLE 4-1. EARTH-TO-SPACE RAIL LAUNCHER PROJECTILE MASS SUMMARY

Projectile Subsystem	Mass (kg)
Payload	650
Propellant	1150
Dry propulsion system	425
ACS	50
Astrionics	25
Instruments	30
PSS	2730
TPS	100
Nose cone	420
Fins	20
Forward sabot	200
Aft sabot	100
Total	5900

The payload support structure (PSS) made of high-strength steel would provide structural support for the payload and propulsion system. The propulsion system would be located in the forward part of the PSS; the nozzle would point toward the nose cone. The PSS would also support the propellant tanks during the high-acceleration launch. The payload would be located in the aft portion of the PSS, attached to the propulsion system. The PSS would be jettisoned before the orbit-insertion burn.

Another advantage of using the square bore is that the fins can be attached more simply to the projectile, no "pop-out" mechanism is required. Four fins are attached to the rear of the projectile to stabilize the projectile during the atmospheric flight.

4.2.2.2 Surface Transportation Systems

The payload, propulsion system, and projectile would be manufactured and assembled in facilities away from the launch site. The systems would be transported to the launch site by conventional surface transportation systems: truck, rail, aircraft, or ship. Aircraft could be used to transport required personnel and high-priority materials to the launch site directly.

4.2.2.3 Launch Site Support Facilities

Launch support facilities to be located on the launch site include: power plant, projectile storage and check-out facilities, general storage facilities, administration and engineering facilities,

industrial area, community living area (if the site is remote), liquid gas and water plant, and other miscellaneous facilities. The launch site should be selected in an area where there are no large population areas within a radius of approximately 50 km. Figure 4-4 is an artist's concept of the launch site.

<u>4.2.2.3.1 Power Plant</u>. A dedicated nuclear power plant was envisioned to supply the electrical power requirements of the rail launcher system and its supporting systems. The power plant would consist of four 50-MWe nuclear reactors similar to those used on Navy ships. The number of power plants is higher than nominally required, because redundancy has been considered for maintenance and unscheduled shutdowns.

<u>4.2.2.3.2 Projectile Storage and Checkout Facilities.</u> A projectile storage and checkout facility would be required to store at least 70 projectiles (one week supply at a launch rate of ten per day) from time of arrival at the site until the projectiles are moved to the storage facility located at the breech of the rail launcher.

<u>4.2.2.3.3 General Storage Facilities</u>. General storage facilities are needed to support the activities of the staff and the operations of the launcher system. Office supplies, gasoline, and rail launcher spares are among the items which would be stored.

<u>4.2.2.3.4 Administration and Engineering Facilities</u>. Office space would be required for the administration and engineering staff. These buildings would be located near the industrial area.

<u>4.2.2.3.5 Industrial Area.</u> Various industrial facilities would be required to support the launch activities, including receiving areas, repair and refurbishment shops, vehicle maintenance, and other facilities as deemed necessary. Aircraft runways would be built to support incoming aircraft traffic of people and supplies. The industrial area would be located in an area appropriate to the activity.

<u>4.2.2.3.6 Community Living Area</u>. If the launch site were located in a remote area with no nearby towns, a community living area would have to be built at a practical distance from the rail launcher facilities. The living area would likely include apartment-type housing, schools, and some shopping and entertainment facilities to support the launch crew and their families.

<u>4.2.2.3.7 Liquid Gas and Water Plant</u>. Liquid nitrogen is required to cool the 3600 inductors and the preaccelerator requires liquid hydrogen and liquid oxygen. A water supply is necessary for the hydraulic operation of the homopolar generators and for space station supply payloads, as well as for launcher base and personnel supplies. A plant is needed to support these requirements, and would likely be located close to the power plant. Liquid nitrogen lines would be used to send the LN_2 to the launcher system, while LH_2 and LO_2 would be transported by truck. Water would be distributed throughout the launch site by an underground plumbing system.

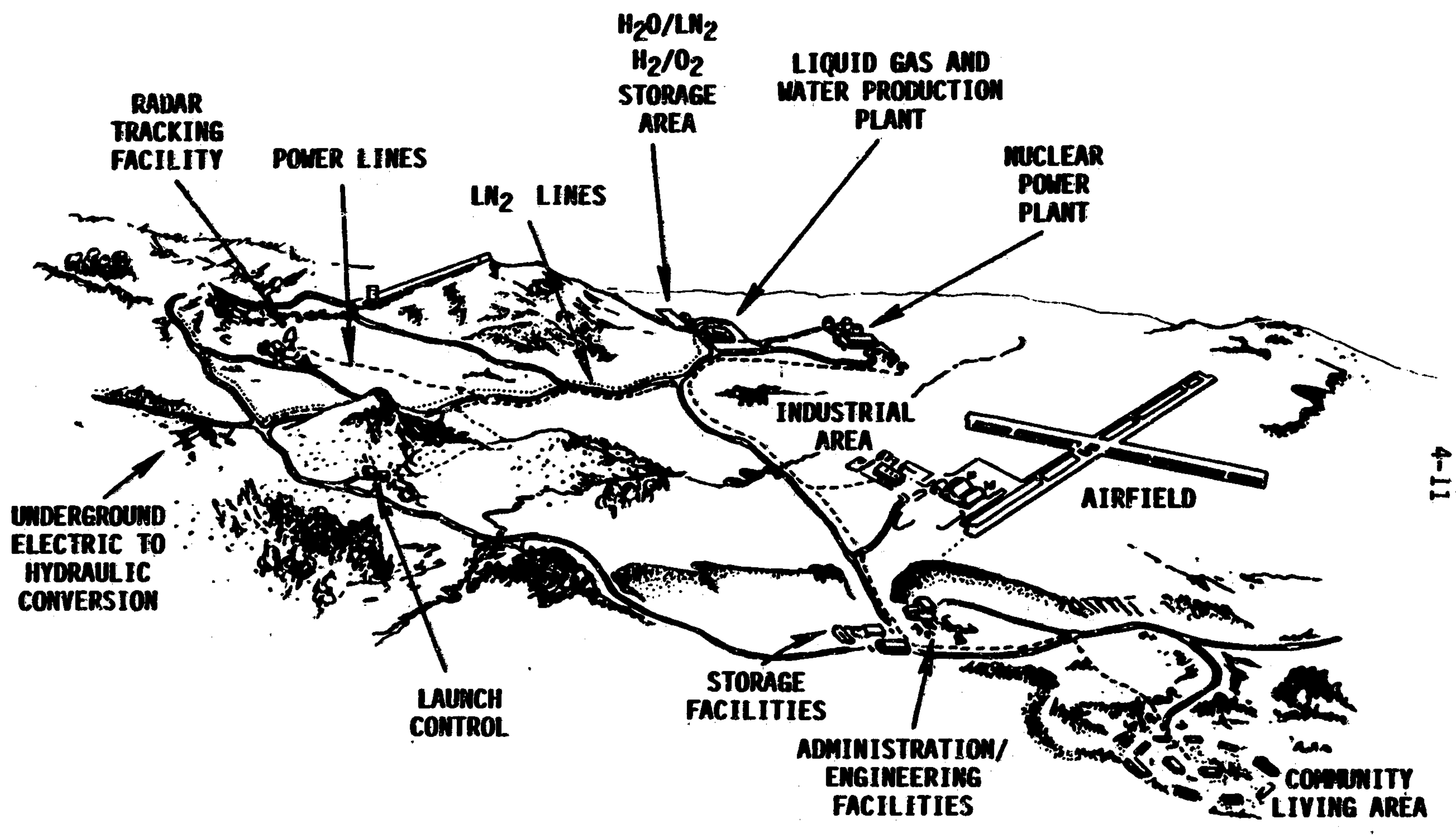


FIGURE 4-4. EML LAUNCH SITE

4.2.2.3.8 Other Facilities. Other necessary support facilities include:

- Electric-to-hydraulic facility
- Launch control center
- Radar tracking facility.

4.2.2.4 Launcher System

The Earth-to-orbit rail launcher system would accelerate the projectile to a 6.85 km/s launch velocity. The launcher is based on a mountain side at an elevation angle of 20 degrees from the horizontal. Access for maintenance and repair has been provided. Figure 4-5 shows a cross-sectional view of the rail launcher system, while Figure 4-6 presents a top view. The rail launcher would have a square bore, 1 m across. The rails would be fabricated of AMZIRC (a copper-zirconium alloy) and would be electrically insulated and spaced with a non-asbestos fiber-reinforced insulator material. A containment tube constructed of Kevlar would confine the rails and insulator materials.

The rail launcher system would be powered with 3600 homopolar generators with corresponding inductors, which would be evenly distributed alon the length of the launcher. Self-activated switches would control the current distribution from the inductors into the rails.

The following subsections briefly discuss the rail launcher systems which have been conceptualized. Launcher systems include:

- Bore and rails
- Energy storage
- Launcher support structure
- Preaccelerator system
- Switching and control
- Temporary projectile storage facilities.
- Service and access systems.

Launcher concept options are provided in Figure 4-7.

4.2.2.4.1 Bore and Rails. The rail launcher would have a square bore which is 1 m across. The rails would be 2040 m in length and would be constructed of AMZIRC, a copper alloy which is approximately 99.9 percent copper and 0.1 percent zirconium. AMZIRC was selected for the rails because its strength and conductivity are better than copper alone.

To calculate the minimum distance between the rails, the pressure applied at the base of the projectile is equal to the pressure

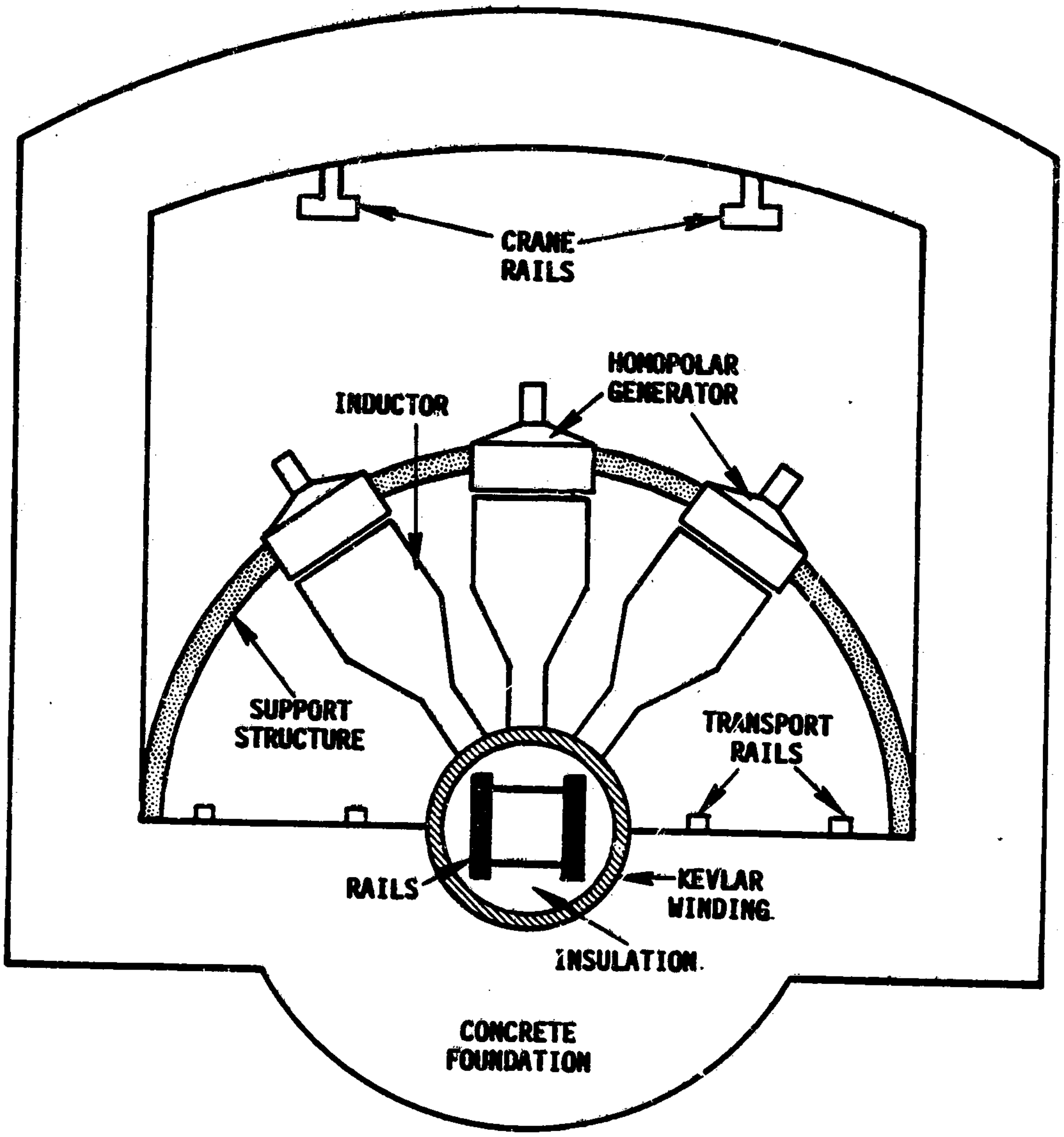


FIGURE 4-5. RAILGUN SYSTEM CROSS-SECTIONAL VIEW

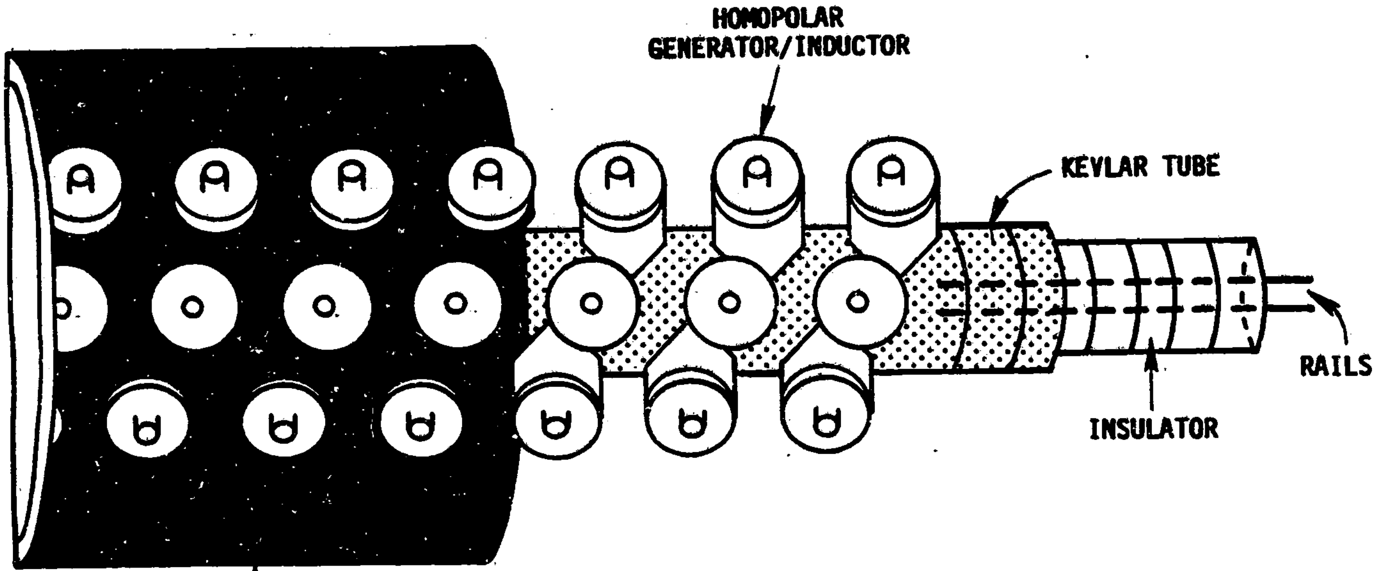


FIGURE 4-6. EARTH-TO-ORBIT RAILGUN SYSTEM SIDE VIEW

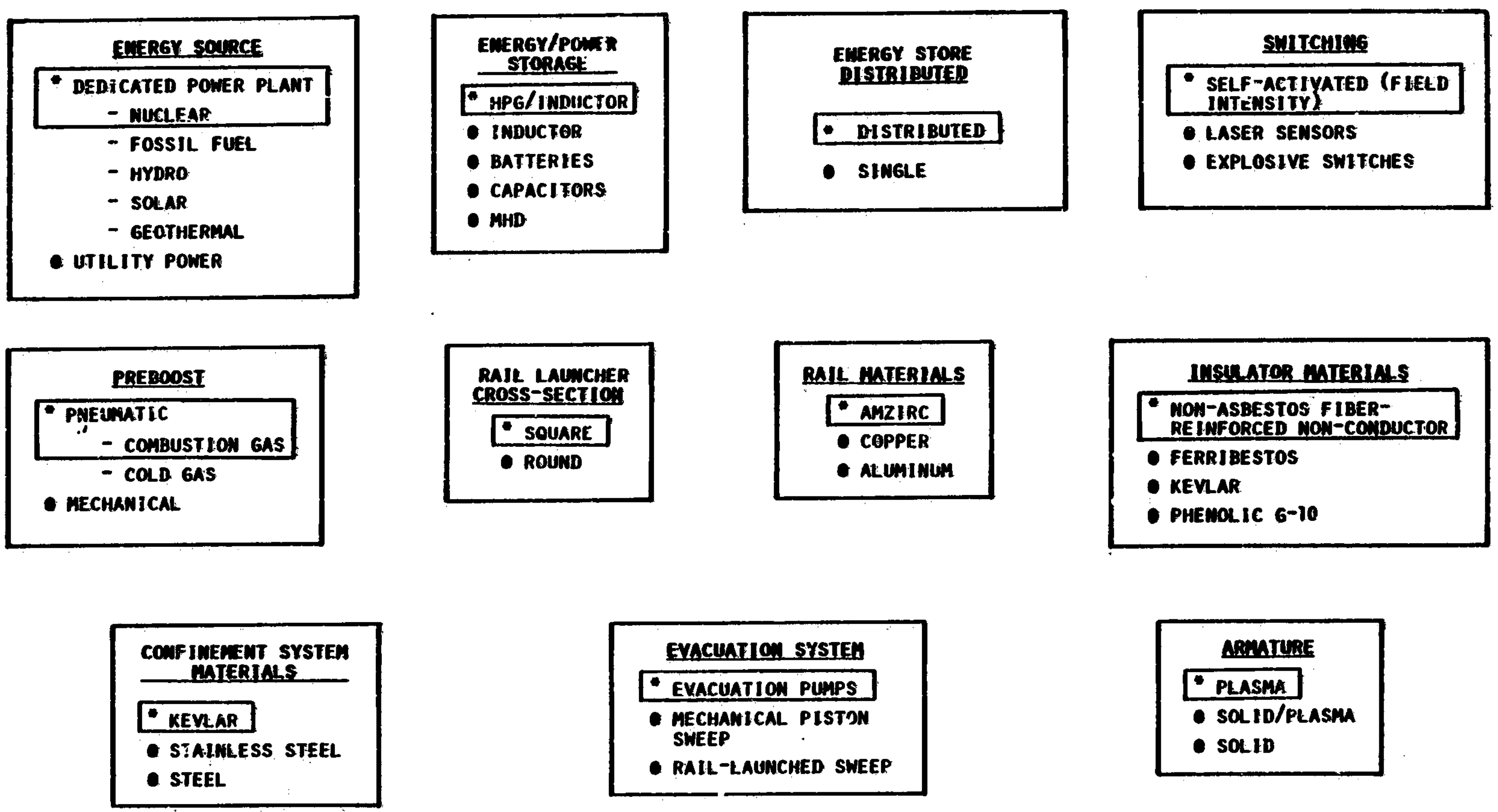


FIGURE 4-7. OPTIONS CHART - EARTH-TO-ORBIT RAIL LAUNCHER

exerted on the rails. The yield strength of AMZIRC is 42,100 N/cm^2 (Engineering Alloys Digest, 1961). Because the force on the projectile is 70.9 MN, the resulting minimum bore area is 1684 cm^2. With a square bore, the minimum rail height is 41 cm. This is not a problem since the bore must be 1 m across to accommodate the projectile. The larger bore size is desirable for several reasons. For stress reasons, the rail-launched projectile should be as short as possible, which widens the projectile diameter. Also, payloads are not as size-constrained when the projectile is widened. Another reason for making the bore larger is the lower resulting pressure on the rails which makes it easier to hold the rails flat during launch. There is a disadvantage to the larger bore diameter, however, in that the magnetic field between the rails decreases as the distance between them increases.

The rails are separated by insulator materials. The tube is wrapped in Kevlar to contain the tube stresses. Figure 4-6 illustrates the bore, rails, insulation, and spacers.

<u>4.2.2.4.2 Energy Storage</u>. Several energy storage devices were considered for use on the rail launcher system. Homopolar generators in the 50 to 60-MJ range should be available in the next five to ten years (telephone conversation with Dale Pryor, OIME, 1984). A 50-MJ HPG should have a mass of 13,000 kg (Marshall, 1984). A single HPG of the size required (200 GJ) would not be available in the near future and was not considered further in this study.

The HPG/inductor unit developed by Richard Marshall (Section 3.2 of Rice, et al 1982) was selected as the energy storage device and is illustrated in Figure 4-8. The device consists of a 56-MJ HPG coupled with an inductor which stores 48 MJ at a current of 4 MA. The aluminum inductors should have a mass of approximately 1 to 1.5 MT each. The inductors would be cooled to liquid-nitrogen temperatures because of mass and volume efficiencies.

The energy required per unit length is 70.9 MJ/m. Assuming a 72 percent energy transfer efficiency (85 percent from HPG to inductor and 85 percent from inductor to railgun), 3585 HPG/inductor units would be required for the desired 7 km/s launch velocity. The HPG/inductor units would be placed at 0.6 m spacing along the launcher tube.

<u>4.2.2.4.3 Launcher Support Structure</u>. The launcher support structure as envisioned is shown in Figure 4-5. The launcher tube would be partially imbedded in a concrete foundation to contain a forces during launch operations and to maintain the tube's position along its 2 km length. The massive HPG/inductors must be supported as well. They are shown in the figure supported by a half-cylinder steel structure. No structural analysis has been conducted to support this preliminary concept.

<u>4.2.2.4.4 Preaccelerator System</u>. The preaccelerator concept has not been changed from the previous ESRL study (Rice, Miller, and

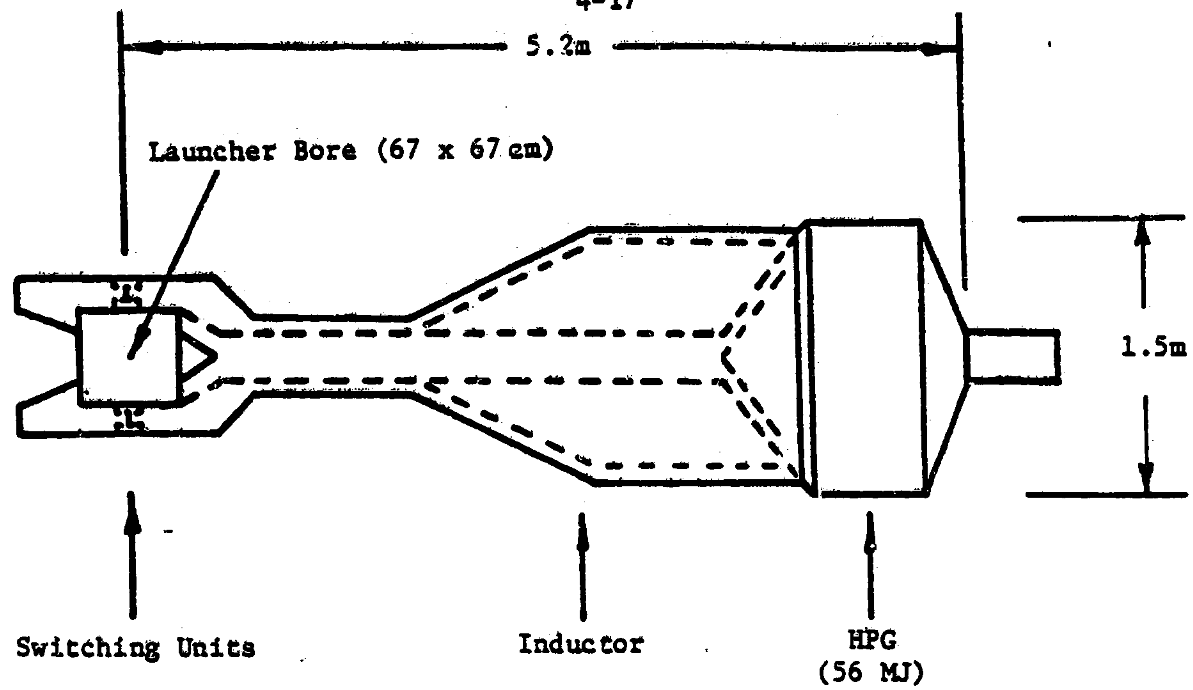


FIGURE 4-8. HOMOPOLAR GENERATOR AND INDUCTOR

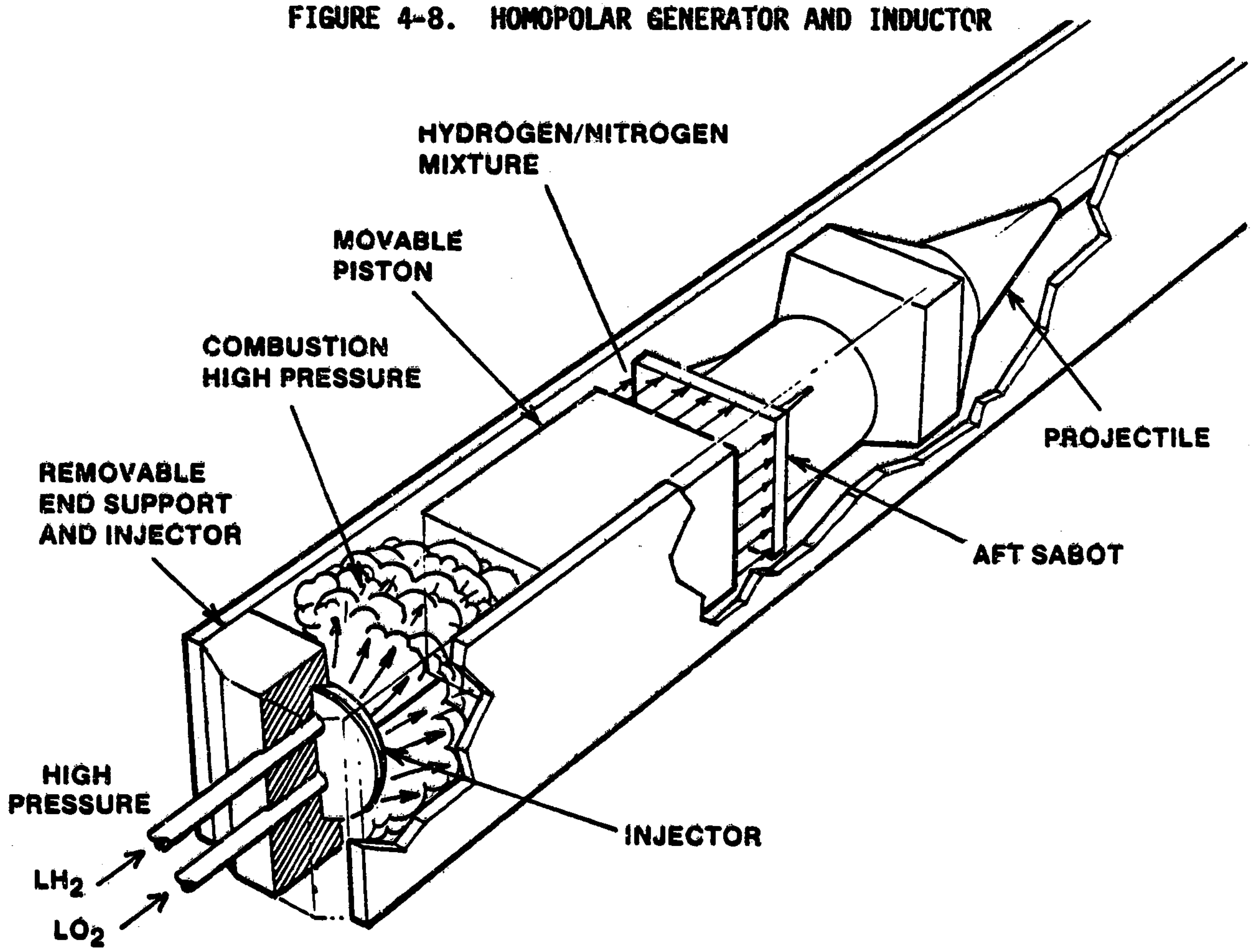


FIGURE 4-9. PROPOSED PREBOOST CONCEPT

Earhart, 1982). The system is required to prevent erosion of the rails caused by the dwell time of the plasma armature during the initial acceleration. Preboost systems currently used accelerate projectiles to 0.5 to 1.0 km/s with a high-pressure (3000-5000 psi) helium gas injector (Hawke, et al, 1984 and D'Aoust, et al, 1984). Helium is generally used because of its high speed of sound (approximately 1 km/s at room temperature).

The system suggested here (Figure 4-9) uses liquid hydrogen and liquid oxygen in continuous high-pressure combustion forcing a piston against a gaseous mixture of hydrogen and nitrogen in between the piston and sabot of the projectile. The concept is similar to a gas gun, except that it is continuously driven by the LH_2/LO_2 combustion process. The preboost system would accelerate the projectile to 1 km/s before entering the rail launcher.

4.2.2.4.5 Switching and Control. Switching of the large currents required for the rail launcher system is a major issue which must be resolved before development of the system is begun. When the HPG is fully charged, the energy is transferred to the inductor. Switching from the inductor to the rails would be activated by the projectile movement down the launcher. When discharged, the energy store must be electrically disconnected from the launcher to prevent reverse flow of energy. One possible switching mechanism using chevron-shaped rail segments insulated from each other is discussed in Marshall (1984) and Rice, et al, (1982).

4.2.2.4.6 Temporary Projectile Storage Facilities. A temporary storage facility would be located near the base of the launcher. Projectiles would be moved daily from the storage and checkout facility to support the day's scheduled launches. The facility would need to store at least 15 projectiles.

4.2.2.4.7 Service and Access Systems. Crane and transport rails are included in the reference concept to allow servicing of the launcher systems. A rail system was chosen to facilitate movement along the 20-degree incline of the system.

4.2.2.5 Monitoring Systems

Trajectory monitoring would be needed after the projectile leaves the muzzle of the rail launcher for range safety purposes. Telemetry would be included as part of the projectile package so that the payload could be tracked on orbit for retrieval by vehicles from the space station.

4.2.2.6 Space Destination

The destination for projectiles launched from this rail launcher system is low-Earth orbit. The primary destination would be a 500-km circular orbit in which the space station system is based, but other

orbits are possible. If the system is upgraded to provide for 12 km/s launch velocities (see Section 3.7), launches to geosynchronous orbit would be possible.

4.3 Earth-to-Space Coaxial Accelerator Concept

4.3.1 Concept Definition

The Earth-to-space coaxial accelerator concept for launching materials to Earth orbit consists of five major operational activities:

(1) Projectile/payload fabrication
(2) Surface transport of projectile/payload
(3) Projectile/payload preparation at the launch site
(4) Launch operations
(5) Trajectory monitoring.

The activities are the same as described in Section 4.2.1 for the Earth-to-space rail launcher, except for the launch operations activities which are described here. Individual coaxial accelerator subsystem definitions are given in Section 4.3.2.

The projectile is removed from the storage facility just prior to its scheduled launch time. All launcher systems are checked out and the projectile is loaded into the small preaccelerator required to initiate projectile motion. The Brooks coil inductor is charged and the launcher tube is evacuated.

Final launch countdown begins after weather conditions are checked and launch clearance is obtained from the proper authorities. All persons are cleared from the immediate launch area.

4.3.2 System Element Definition

Six major system elements have been identified and are discussed in this section. These system elements are:

- Projectile/payload characteristics
- Surface transport systems
- Launch site support facilities
- Launcher system
- Monitoring systems
- Space destination.

4.3.2.1 Projectile/Payload Characteristics

4.3.2.1.1 Payload. The payload would be identical to that for the Earth-to-space rail launcher, described in Section 4.2.2.1.1.

4.3.2.1.2 Propulsion System. Propulsion system characteristics are the same as those described in Section 4.2.2.1.2 for the Earth-to-space rail launcher.

4.3.2.1.3 Projectile. The proposed projectile is shown in Figure 4-10 and consists of the following subsystems:

- Projectile coils
- Nose cone
- Instrument package
- Liquid propulsion system, including ACS and astrionics
- Payload
- Payload support structure (PSS)
- Thermal protection system (TPS)
- Fins.

The projectile subsystems are similar to those described in Section 4.2.2.1.3 for the Earth-to-orbit rail launcher, except where noted below.

Forty projectile coils (actually copper rings) would be spaced every 16 cm along the projectile. The rings would be 2-cm thick with an outer diameter of 49 cm. Each ring would have an axial length of 3 cm. The coils would be imbedded in the carbon-carbon TPS material.

Because of the longer and narrower shape of the projectile, the nose cone mass is smaller than that of the railgun projectile. The steel nose cone would have a mass of 110 kg (versus 420 kg for the rail launcher projectile). The projectile mass summary is shown in Table 4-2.

4.3.2.2 Surface Transportation Systems

Surface transportation systems would be the same as those described in Section 4.2.2.2 for the Earth-to-orbit rail launcher concept. Depending upon the launch site location, trucks, rail, aircraft, or ships could be used to transport projectiles, payloads, personnel, and equipment.

4.3.2.3 Launch Site Support Facilities

Launch site support facilities would be similar to those described in Section 4.2.2.3 for the corresponding rail launcher launch site. The facilities include: power plant, projectile storage and general storage facilities, administration and engineering offices, industrial area, community living area for a remote launch site, liquid gas and water plant (LN_2 to cool Brooks coil inductor; water for supply and launch site use), and other necessary facilities. An artist's concept

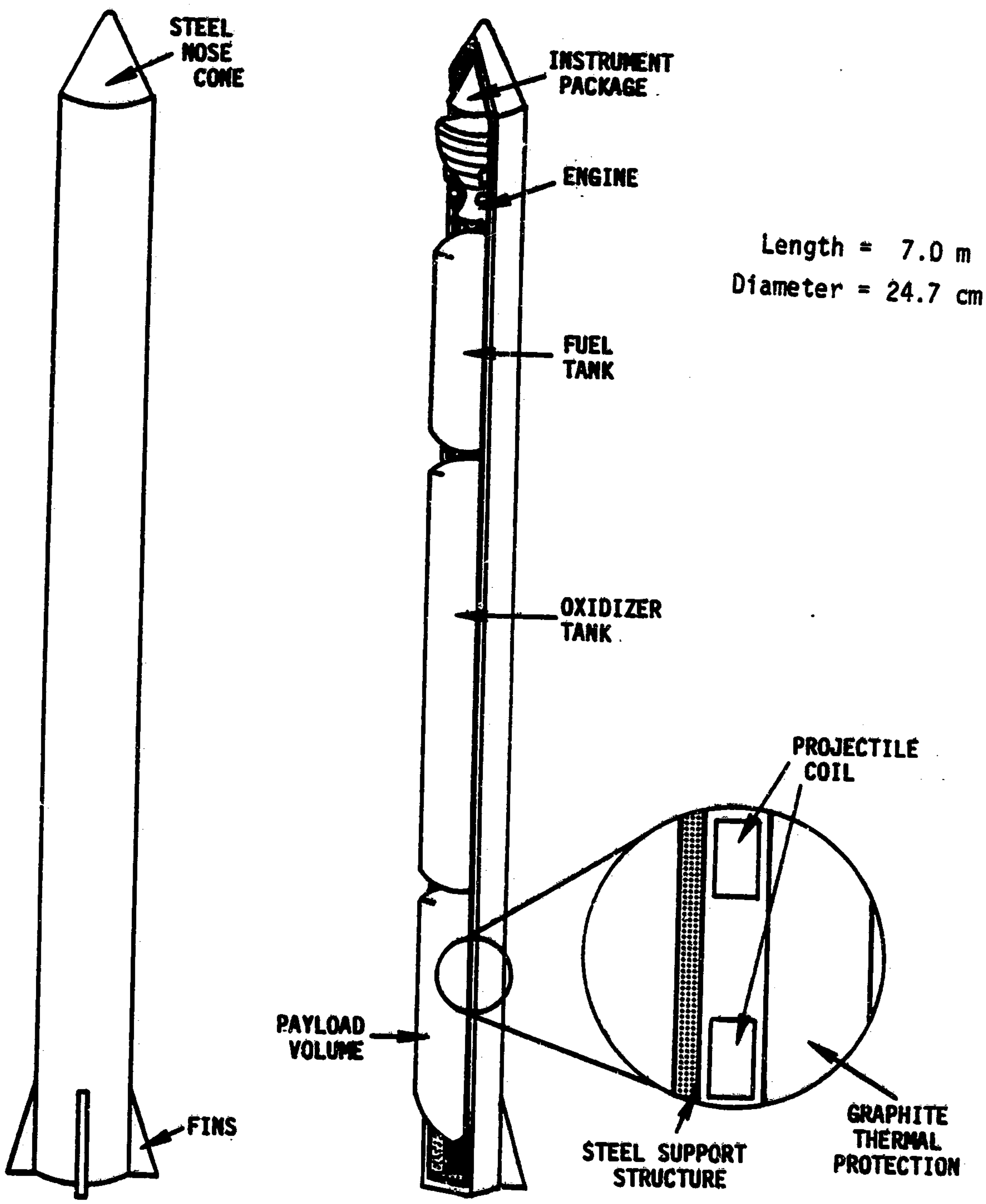


FIGURE 4-10. EARTH-TO-ORBIT COAXIAL PROJECTILE

of the launch site facilities is shown in Figure 4-4.

TABLE 4-2. EARTH-TO-SPACE COAXIAL PROJECTILE MASS SUMMARY

Projectile Subsystem	Mass (kg)
Payload	650
Propellant	1150
Dry propulsion system	425
ACS	50
Astrionics	25
Instruments	30
PSS	300
TPS	200
Nose cone	110
Fins	20
Coils	300
Total	3250

4.3.2.4 Launcher System

This coaxial accelerator system would launch projectiles at 6.85 km/s to supply materials to a 500-km Earth orbit. The launcher would be located along a 2-km stretch of mountain side inclined at 20 degrees from the horizontal. Figure 4-11 is a cross-sectional view of the launcher system.

The drive coils are a continuous helical winding (2040 m long) made of a copper-alloy material. Electrical insulation and structural tube confinement are provided by encasing the drive coils in Kevlar.

Energy would be supplied to the coaxial launcher system from a single large Brooks coil storage inductor. Energy would be distributed to the drive coils such that ten turns behind each projectile ring would be active at all times. The turns would be switched in and out as the projectile moves toward the launcher muzzle.

Launcher systems discussed in this section include:

- Drive coils
- Energy storage
- Launcher support structure

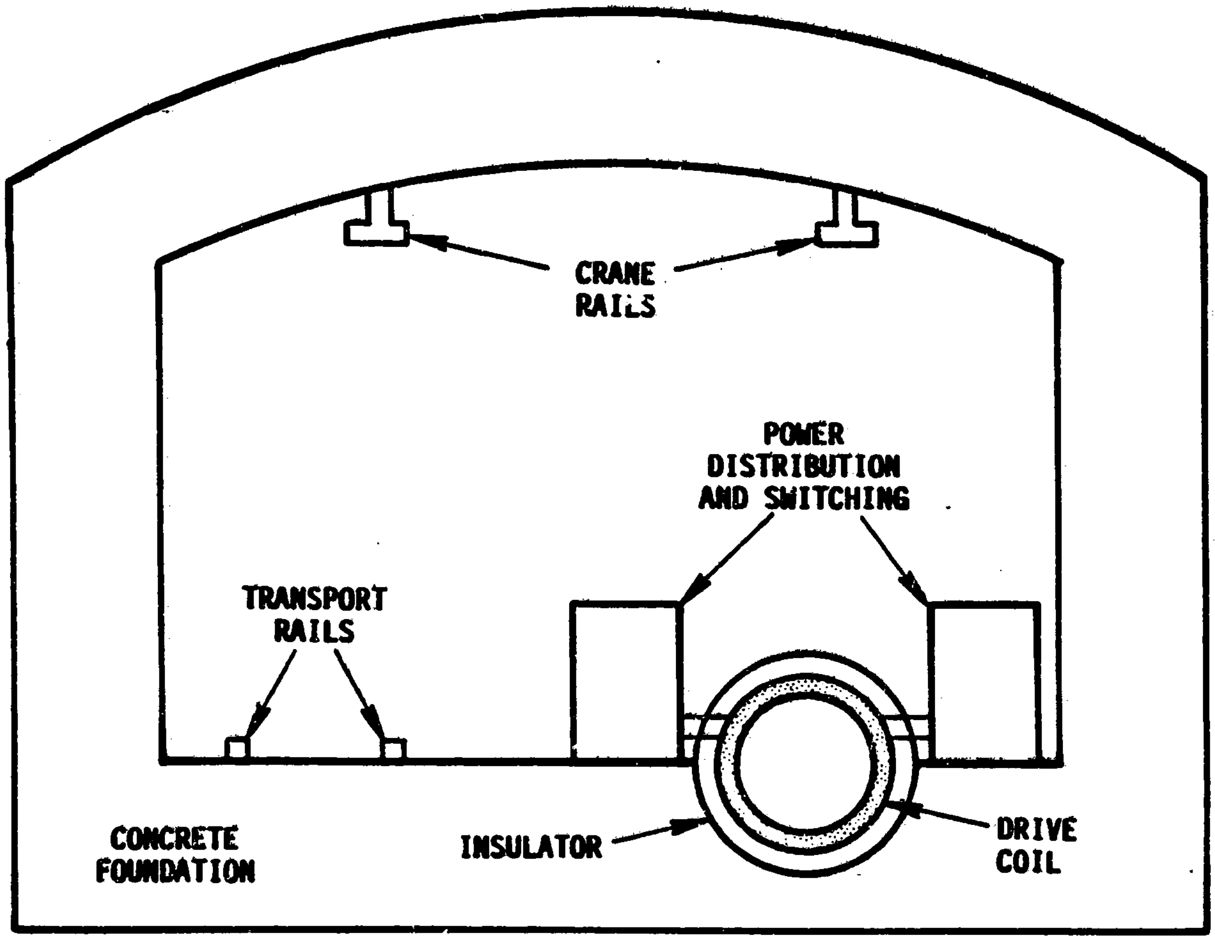


FIGURE 4-11. COAXIAL ACCELERATOR SYSTEM
CROSS-SECTIONAL VIEW

- Preaccelerator system
- Switching and control
- Storage facilities
- Service and access systems.

Earth-to-orbit coaxial concept options are presented in Figure 4-12.

4.3.2.4.1 Drive Coils. The drive coils would be constructed from one layer of rectangular copper-alloy wire in a continuous helical winding. The inner radius would be 25.3 cm and the outer radius 26.3 cm. The length of the drive coil section is 2040 m. An active segment of 8-cm length consisting of ten turns would follow each projectile ring down the length of the launcher tube.

The drive coils are encased in Kevlar for structure and for insulation between the windings. The tube structure does not need to be as strong as that for the corresponding rail launcher system. This is because tube hoop stresses are lower due to the induction of the projectile coils which adds counterbalancing stress.

4.3.2.4.2 Energy Storage. A single large Brooks coil inductor would be used for the energy storage device for this concept. A Brooks coil configuration offers the maximum inductance for a given length of wire used. The dimensions of a Brooks coil are shown in Figure 4-13. The kinetic energy of the projectile at the muzzle of the launcher tube would be approximately 80 GJ. Kolm and Mongeau quote efficiencies of 98.9 percent found by dividing the kinetic energy of the projectile at launch by the supplied energy to the launcher tube. Assuming an 85 percent transfer efficiency from the Brooks coil inductor to the launcher itself, this corresponds to an energy storage requirement of approximately 95 GJ. Information supplied by EML Research, Inc. implies that energy stored is a function of outside diameter (Appendix D). This relationship is shown below:

$$E = 0.0207\ D_o^3 \qquad (4\text{-}1)$$

where the energy E is in GJ and the outside diameter D_o is in meters. Therefore a Brooks coil which would store 95 GJ would have an outside diameter of approximately 36 m. The Brooks coil would be made from aluminum wire. Liquid nitrogen would be used to cool the inductor.

4.3.2.4.3. Launcher Support Structure. The launcher tube would be partially imbedded in a concrete foundation as shown in Figure 4-10. The purpose of the structure is to help contain the hoop stresses of launch and to add rigidity to the 2-km long tube to keep it in alignment.

4.3.2.4.4 Preaccelerator System. A preaccelerator system was added to the reference concept to provide a small initial velocity (up to 100 m/s) to get the projectile moving through the accelerator up the 20-degree incline. A large system, such as that required for

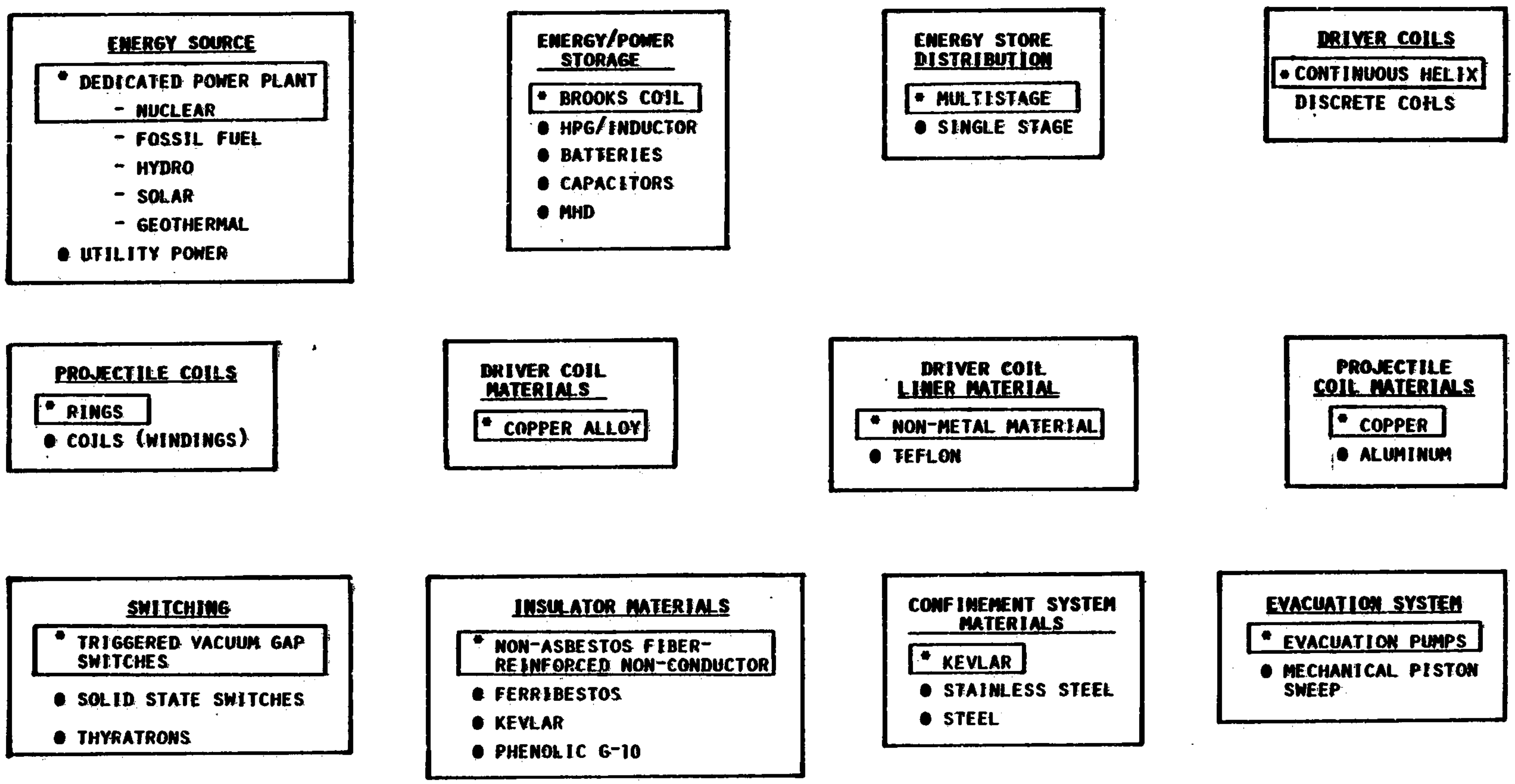


FIGURE 4-12. OPTIONS CHART - EARTH-TO-ORBIT COAXIAL LAUNCHER

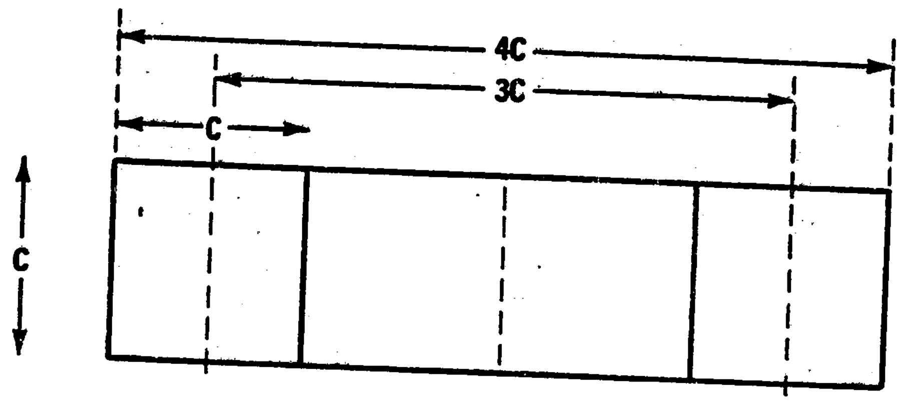


FIGURE 4-13. BROOKS COIL WITH DIMENSIONAL RELATIONSHIPS

the rail launcher system to prevent rail erosion, is not required for the coaxial accelerator system.

4.3.2.4.5 Switching and Control. The drive current needs to be synchronized with the motion of the projectile. Ten drive coil turns are active behind each projectile coil and must be switched in and out as the projectile passes through the launcher. Switching issues are discussed in more detail in Appendix D and in Section 7.1.

4.3.2.4.6 Temporary Projectile Storage Facilities. The projectile storage facility would be similar to that described in Section 4.2.2.4.6 for the Earth-to-orbit rail launcher concept.

4.3.2.4.7 Service and Access System. As in the corresponding rail launcher concept, a transport rail system would be used to provide access for maintenance and servicing of the launcher system.

4.3.2.5 Monitoring Systems

Trajectory monitoring and orbital tracking systems would be identical to the Earth-orbital rail launcher systems (Section 4.2.2.5).

4.3.2.6 Space Destination

The primary destination for projectiles launched from the coaxial accelerator system would be a 500-km circular orbit. Other low-Earth orbits would be possible, with different projectile auxiliary propulsion systems. Should the system be upgraded to allow velocities of 12 km/s, geosynchronous altitude destinations would be possible.

4.4 Hybrid Railgun/Rocket Concept

4.4.1 Concept Definition

This section discusses the hybrid railgun/rocket concept for delivering materials to low-Earth orbit. The five major pre-launch, launch, and post-launch activities of the reference concept are:

(1) Projectile payload fabrication

(2) Surface transport of projectile/payload

(3) Projectile preparation at launch site

(4) Launch operations

(5) Trajectory monitoring.

Individual hybrid system element definitions are provided in Section 4.4.2.

4.4.1.1 Projectile Payload Fabrication

The solid-rocket motors and certain payloads would be manufactured away from the launch site. The systems would receive factory checkout prior to delivery to the launch site. Water payloads would be supplied at the site.

4.4.1.2 Surface Transport of Projectile Payload

Conventional methods of surface transportation (truck, rail, aircraft, and ship) would be used to transport the payloads and rocket motors. Trucks and railroad cars would be the likely candidates over ground. Ships or air transport could be used if the site were inaccessible by other methods.

4.4.1.3 Projectile Preparation at Launch Site

When the payloads and motors arrive at the launch site, they would be placed in a storage facility. When the scheduled launch time approached, the motors would be removed from storage and stacked. The payload would be integrated and the projectile assembled. The projectile would then receive a prelaunch checkout before being transported to the launcher facility.

4.4.1.4 Launch Operations

At the scheduled launch time, the projectile is loaded into the breech of the launcher. All launcher systems are checked out and the launch tube is evacuated prior to launch. Charging of the homopolar generator begins.

Weather and wind conditions would be checked and launch clearance given from the proper authorities before the final countdown

begins. A launch site alarm would be sounded warning all persons to be cleared from the area.

The final launch countdown begins after all systems are readied. The launch systems are computerized and fully automated. A solid armature is attached to the rear of the projectile behind the protective sabot. Current flows through the armature providing the force required for launch. Switching of the current into each segment is activated by the projectile movement through the launcher tube. When the projectile leaves the railgun, it is travelling at 2 km/s. After a brief coast period, the second stage (rocket first stage) is ignited. The stage is jettisoned following burn-out. The rocket second stage is then ignited and is also jettisoned after burn is completed. The rocket third stage is fired at orbital altitude and places the payload in its proper orbit.

4.4.1.5 Trajectory Monitoring

A telemetry system was assumed to be included as part of the projectile system. The payload could then be tracked during its trajectory and on orbit to facilitate retrieval by the OMV for transport to a space station.

4.4.2 System Element Definition

Definitions of the hybrid railgun/rocket system elements are provided in this section. The six major system elements discussed here are:

- Projectile/payload characteristics
- Surface transport systems
- Launch site support facilities
- Launcher system
- Monitoring systems
- Space destination.

4.4.2.1 Projectile/Payload Characteristics

4.4.2.1.1 Payload. The payloads to be delivered to low-Earth orbit by the hybrid railgun system would include:

- Orbit Transfer Vehicle propellants
- Space station supply items
- Materials for space processing activities.

These payloads are described in more detail in Sections 2.1 and 4.2.2.1.1.

Despite the lower accelerations of the hybrid railgun/rocket launch compared to the Earth-to-orbit rail launcher (100 g versus 1225 g),

the mission model remained the same because certain supply items and sensitive equipment would still be transported by the Space Shuttle with a maximum acceleration of 3 g.

The analysis documented in Section 3.1.4 indicated a maximum payload of 800 kg for the total projectile mass of 15,000 kg. It was assumed, however, that 660,000 cm^3 of the payload volume was available. Therefore propellant-supply payloads (i.e., water) would have a mass of 660 kg. This was the basis for the hybrid EML/rocket traffic model shown in Table 3-18.

<u>4.4.2.1.2 Projectile Elements</u>. The hybrid railgun/rocket projectile would consist of the following subsystems:

- Rocket motor stages
- Solid propellant
- Payload
- Nose cone
- Instruments
- Sabot/armature
- Fins.

The proposed projectile concept is depicted in Figure 4-14. Table 4-3 presents the mass summary of the hybrid railgun/rocket projectile.

TABLE 4-3. HYBRID RAILGUN/ROCKET PROJECTILE MASS SUMMARY

Projectile Subsystem	Mass (kg)
Payload	800
Propellant	12,700
Propulsion system/casing	1,100
Instruments	20
Nose cone	180
Fins	100
Sabot/armature	300
Total	15,200

The rocket motor structure would have filament-wound cases, which allows a lower structural mass than steel would, since filament-wound cases can be "stressed to values several times the permissible limit of stress in an equal weight of steel" (Hill and

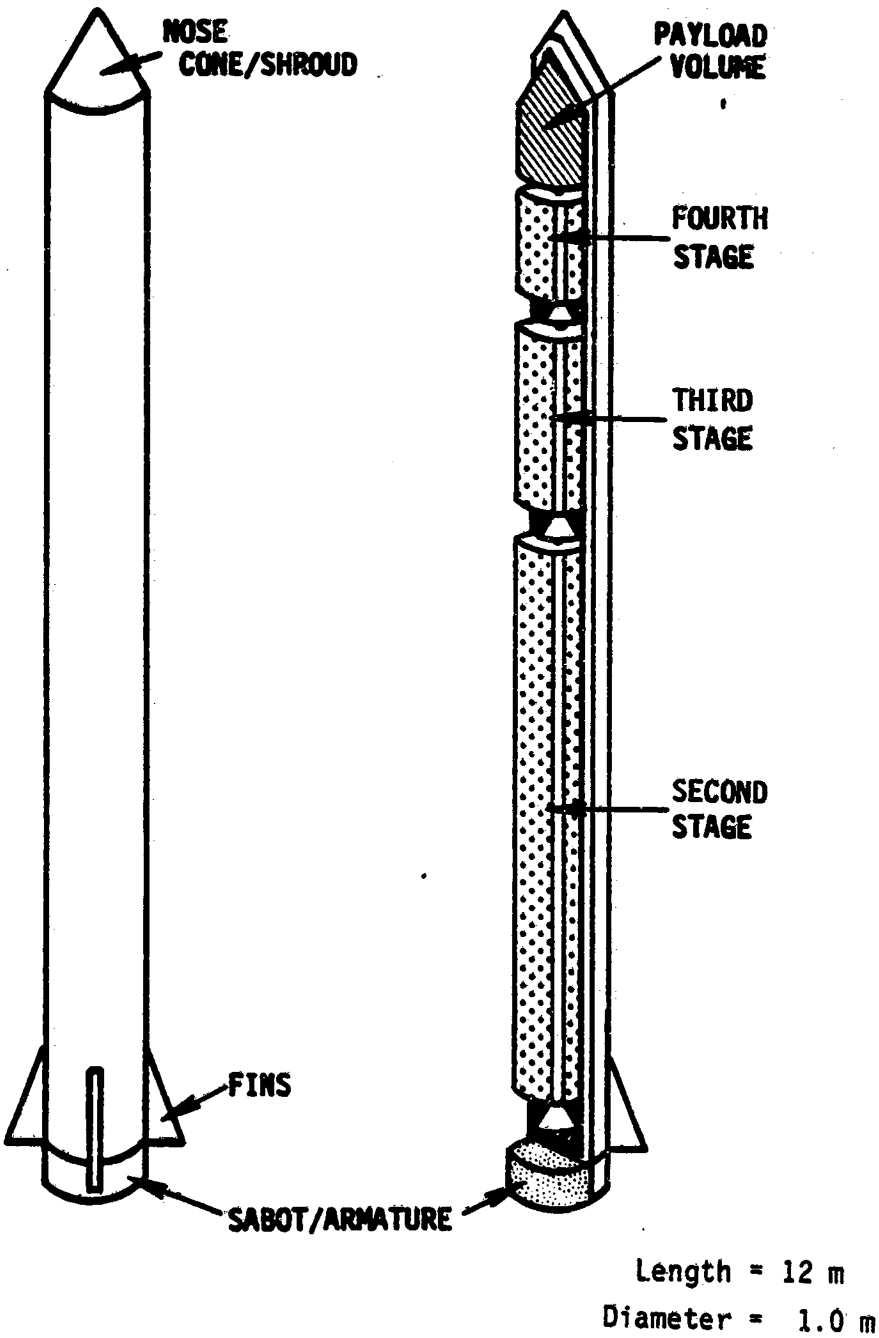


FIGURE 4-14. HYBRID RAILGUN/ROCKET PROJECTILE

Peterson, 1970). The cases would be loaded with solid propellant consisting of approximately 16 percent aluminum, 14 percent polybutadiene, and 70 percent ammonium perchlorate. Approximately 12,700 kg of solid propellant (290 sec Isp) would be required to launch an 800-kg payload to LEO after an initial 2 km/s boost via an EML.

The instruments and payload shroud/nose cone would be similar to those used on conventional expendable launch vehicles. "Pop-out" fins like those on missile systems would be used to stabilize the projectile during its atmosphere flight. The fins would be concealed inside the structural casing during launch to accommodate the round launcher tube bore. When the projectile exits the launcher tube, the fins would be opened to stabilize the vehicle. The 100-g acceleration environment should not provide any real problems, since pop-out fins are used for gun-launched missiles which also experience high accelerations.

A sabot/armature system would be attached to the back of the projectile. A solid armature was assumed because of the 2 km/s velocity limit. The sabot would insulate the projectile from the conducting armature and would be made of a non-conducting material.

4.4.2.2 Surface Transportation Systems

The payloads and solid-rocket motors would be manufactured in facilities away from the launch site. Conventional methods of surface transportation would be used to transport the payloads and motors to the launch site. A combination of truck, rail, aircraft, and ship systems would be used, depending upon the location of the launch site. Aircraft could be used to transport personnel and high-priority materials directly.

4.4.2.3 Launch Site Support Facilities

Facilities which would support launch site activities include: power plant, projectile storage and check-out facilities, general storage facilities, administration and engineering facilities, industrial area, community living area, liquid gas and water plant, and other miscellaneous facilities. Except for the power plant and projectile facilities, the areas would be similar to those described in Section 4.2.2.3 for the Earth-to-orbit rail launcher and are not discussed here.

<u>4.4.2.3.1 Power Plant</u>. Because of the lower power requirements, a dedicated nuclear power plant is not required for the hybrid EML/rocket launches. Utility power would be adequate. Transmission lines would be required and a substation located on the launch site may be required as well.

<u>4.4.2.3.2 Projectile Storage and Checkout Facilities</u>. A facility to store, process, integrate, and checkout the projectiles would be required. Storage of at least 35 projectiles (105 rocket motors) and payloads would be required, corresponding to one week's supply at

a launch rate of five per day. The solid rocket motors would be stored from time of arrival until just before launch, when the three-stage rocket would be stacked and the payload would be integrated. At that time, the assembled projectile would be moved to the temporary storage facility at the base of the launcher.

4.4.2.4 Launcher System

The hybrid railgun/rocket system would launch supply payloads to Earth orbit. The railgun would accelerate the projectile at 2 km/s, after which the three-stage rocket would supply the remainder of the required ΔV to reach the desired orbit. The launcher system would be based on a mountain side inclined at an angle of 20-degrees from the horizontal.

Figure 4-15 and 4-16 provide cross-sectional and side views of the railgun launcher system. The rail launcher would have a circular bore, 1 m in diameter. AMZIRC (a copper-zirconium alloy) would be used to construct the rails. The rails would be electrically insulated and spaced by a non-asbestos fiber-reinforced insulator material. A Kevlar force containment tube would confine the rails and insulator materials.

Distributed along the launcher tube, 750 homopolar generators and inductors would provide energy to the launcher. Switching of the current would be activated by the projectile movement through the rail segments.

Launcher systems which are discussed in the following subsections are listed below:

- Bore and rails
- Energy storage
- Launcher support structure
- Switching and control
- Temporary projectile storage facilities
- Service and access systems.

The launcher system concept options are shown in Figure 4-17.

4.4.2.4.1 Bore and Rails. The rail launcher would have a round bore which is 1 m in diameter. The minimum bore diameter was calculated for AMZIRC rails. With a force on the projectile of 14.7 MN, the resulting minimum bore area is 350 cm^2. This corresponds to a minimum bore diameter of 21 cm (for a round bore). Since the projectile was conceptualized at 1 m diameter, the bore is much larger than the minimum required for rail structural integrity.

AMZIRC was chosen for the rail material because of its good strength and conductivity properties. The rails would be separated

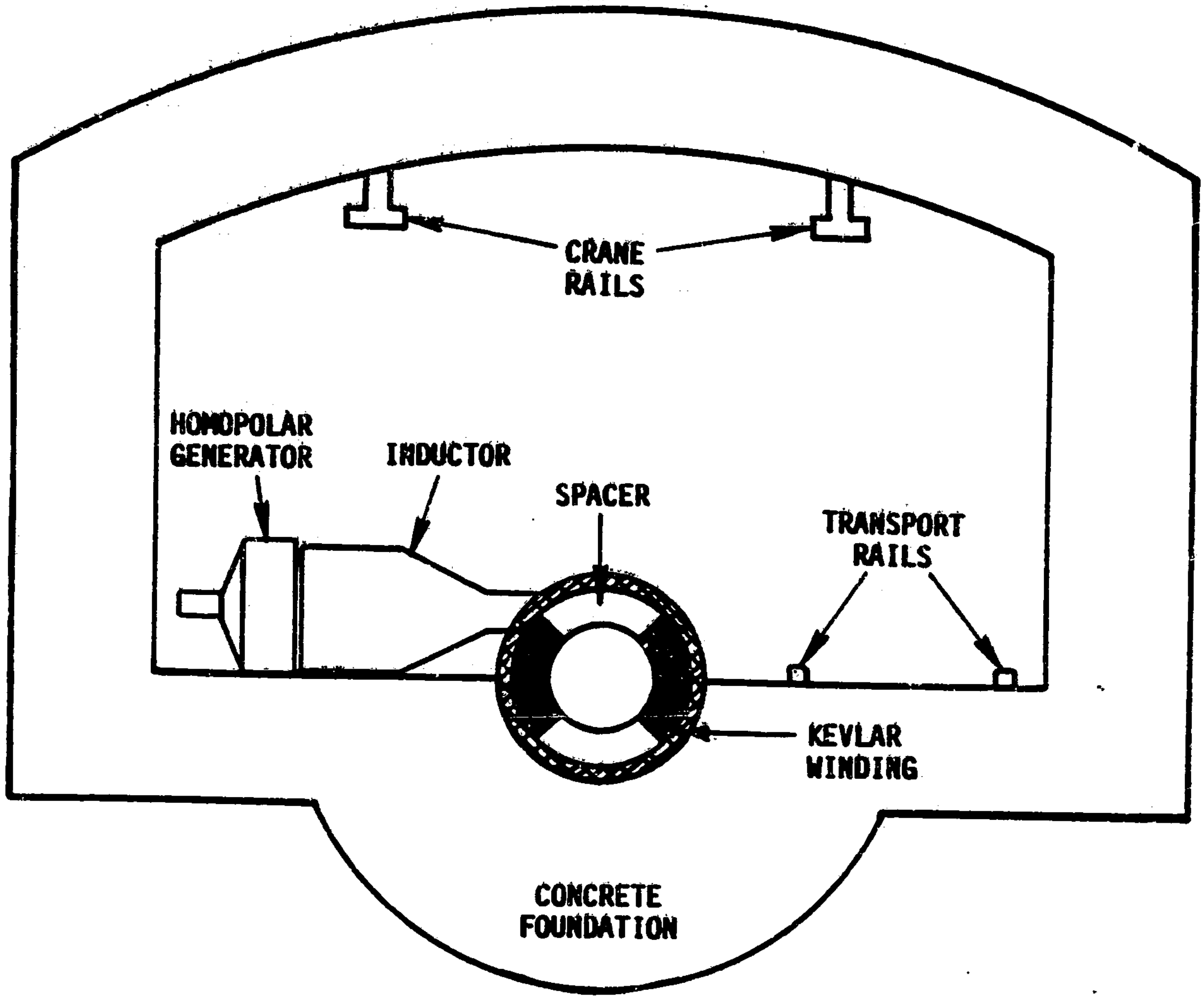


FIGURE 4-15. HYBRID RAILGUN/ROCKET SYSTEM
CROSS-SECTIONAL VIEW

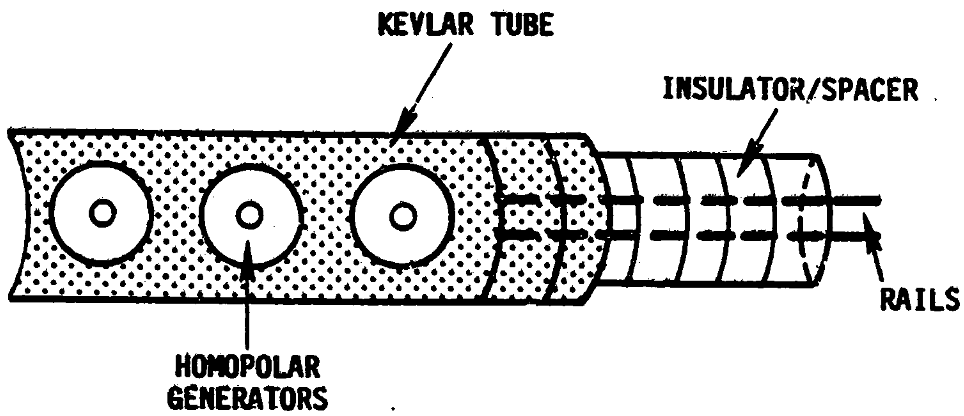


FIGURE 4-16. HYBRID RAILGUN/ROCKET SYSTEM SIDE VIEW

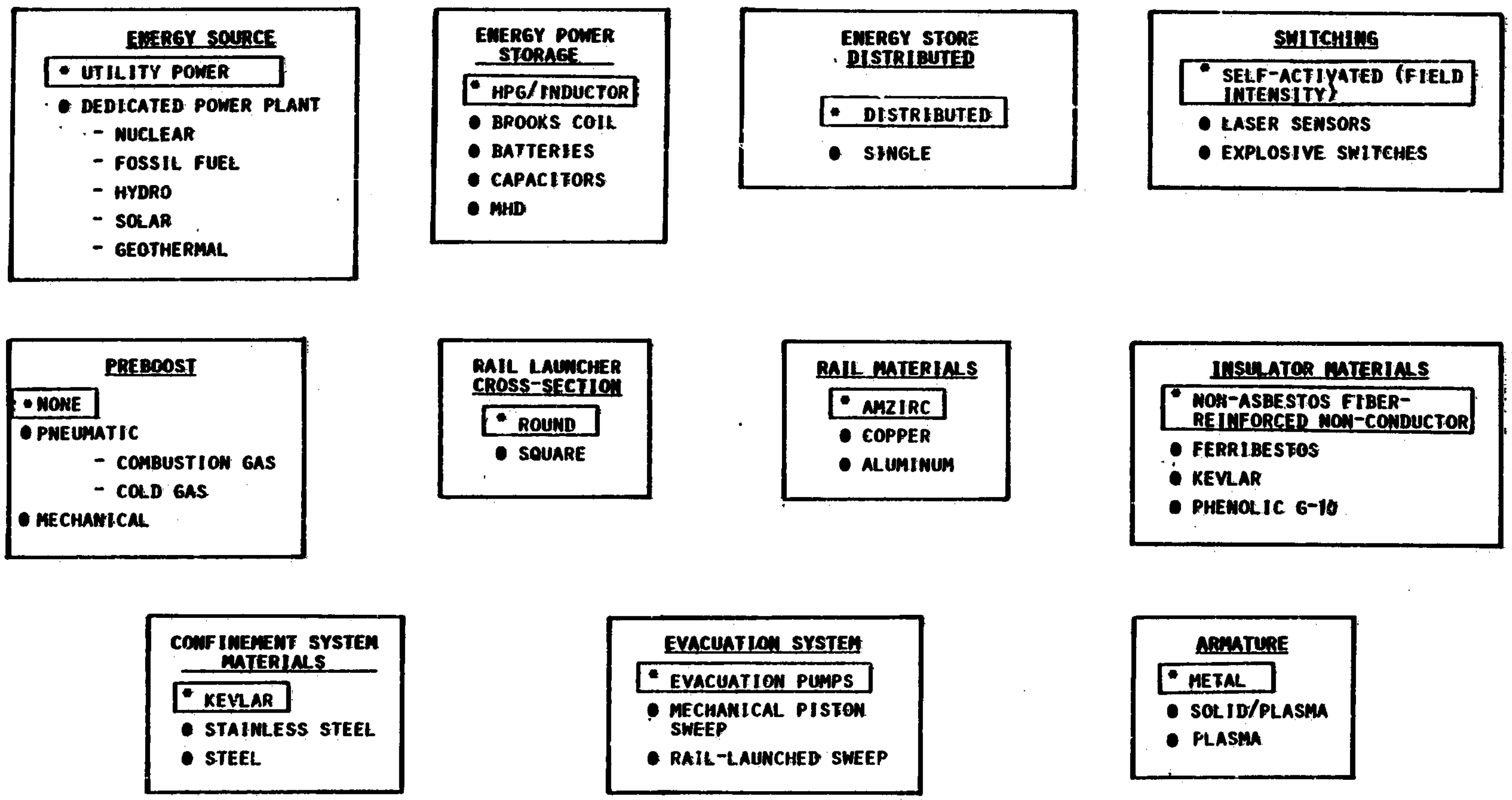


FIGURE 4-17. OPTIONS CHART - HYBRID RAILGUN/ROCKET LAUNCHER

by non-asbestos fiber-reinforced insulator materials. To contain tube stresses during launch, the launcher tube is wrapped in Kevlar. The bore section is seen in Figure 4-15.

4.4.2.4.2 Energy Storage. The homopolar generators (HPGs) and inductors discussed in Section 4.2.2.4.2 would be used as the energy storage devices for the hybrid launcher system. Assuming the same transfer efficiencies from HPG to inductor to rail, 750 HPG/inductor units (56-MJ HPG) would be required to launch the 15,000 kg projectile at 2 km/s. The storage devices would line the length of the launcher tube at 2.7 m center-to-center spacing, as shown in Figure 4-16.

4.4.2.4.3 Launcher Support Structure. The proposed launcher support structure is illustrated in Figure 4-15. A concrete foundation would support the launcher tube maintaining its alignment over its 2-km length. The concrete bed would also help to prevent rail separation and damage caused by launch forces. The homopolar generators and inductors would rest along one side of the foundation, eliminating the need for an additional HPG support structure.

4.4.2.4.4 Preaccelerator System. As indicated in Figure 4-17, a preboost system is not technically required because a solid armature is used (no erosion of the rails due to plasma dwell time). However, it was felt that a small preaccelerator may be advantageous to initiate the projectile motion up the 35-degree elevation angle. Velocities of up to 100 m/s were felt to be sufficient.

4.4.2.4.5 Switching and Control. Switching mechanisms would be similar to the Earth-to-orbit rail launcher (Section 4.2.2.4.5).

4.4.2.4.6 Temporary Projectile Storage Facilities. Each day, projectiles would be moved from the large storage and check-out facility in preparation for launch. Temporary storage of projectiles would be accomplished in a facility located at the base of the rail launcher system. The facility would be required to store one day's supply of projectiles or at least 10 projectiles (2050 high model).

4.4.2.4.7 Service and Access Systems. Access to the launcher system would be provided by crane and transport rails shown in Figure 4-15. This access would allow routine maintenance or emergency repair to occur. The rail system was chosen over others because of the 35-degree launcher elevation angle.

4.4.2.5 Monitoring Systems

Projectile trajectory monitoring would be required after launch for range safety reasons and in flight to insure the rocket motors are performing as expected. Telemetry would be included in the rocket projectile to communicate the required data. The projectile would also need to be monitored on orbit to facilitate retrieval for delivery to the space station.

4.4.2.6 Space Destination

Low-Earth orbit is the primary destination of the hybrid rail launcher. Specifically, the system was designed for launch to a 500-km circular orbit. Other orbits would be possible, at the expense (or gain) of payload mass.

4.5 Hybrid Coaxial Accelerator/Rocket Concept

4.5.1 Concept Definition

The hybrid coaxial accelerator/rocket concept for material delivery to low-Earth orbit consists of five major operational activities:

(1) Projectile fabrication
(2) Surface transport of projectile
(3) Projectile preparation at launch site
(4) Launch operations
(5) Trajectory monitoring.

These activities are the same as those described in Section 4.4.1 for the hybrid railgun/rocket concept, except for the launch operations activities which are discussed here. Definitions of the individual hybrid system elements are presented in Section 4.5.2.

The projectile is removed from the temporary storage facility and placed in the launcher tube. The Brooks coil energy storage inductor is charged. All launcher systems are checked out and the launch tube is evacuated before final launch countdown begins.

Countdown begins after weather and wind conditions are checked. The proper authorities would be notified to obtain clearance to launch. All persons would be cleared from the immediate launch vicinity.

When all systems are readied, the computerized launch sequence begins. Projectile motion is initiated and switching of current into the active segments behind the projectile coils occurs. The muzzle launch velocity is 2 km/s. Rocket procedures are the same as described in Section 4.4.1.4.

4.5.2 System Element Definition

The six major coaxial accelerator system elements described in this section are:

- Projectile/payload characteristics
- Surface transport systems
- Launch site support facilities

- Launcher system
- Monitoring systems
- Space destination.

4.5.2.1 Projectile/Payload Characteristics

4.5.2.1.1 Payload. The payload would be identical to that described in Section 4.4.2.1.1 for the hybrid railgun/rocket system.

4.5.2.1.2 Projectile. The proposed projectile for the hybrid coaxial/rocket concept is shown in Figure 4-18 and consists of the following subsystems:

- Rocket motor stages
- Solid propellant
- Payload
- Nose cone
- Projectile coils
- Fins.

The three solid-rocket motor stages are similar to those assumed for the hybrid railgun/rocket (Section 4.4.2.1.2). During fabrication, however, 80 copper rings (projectile coils) would be imbedded in the structural casing. Each copper ring would be 2-cm thick with an outer diameter of 0.8 m. The axial length of the rings would be 6 cm.

The rest of the projectile is similar to the hybrid railgun/rocket projectile. A mass summary of the proposed projectile is given in Table 4-4.

TABLE 4-4. HYBRID COAXIAL/ROCKET PROJECTILE MASS SUMMARY

Projectile Subsystem	Mass (kg)
Payload	800
Propellant	12,700
Propulsion System/Casing	900
Instruments	20
Nose Cone	180
Fins	100
Projectile Rings	650
Total	15,350

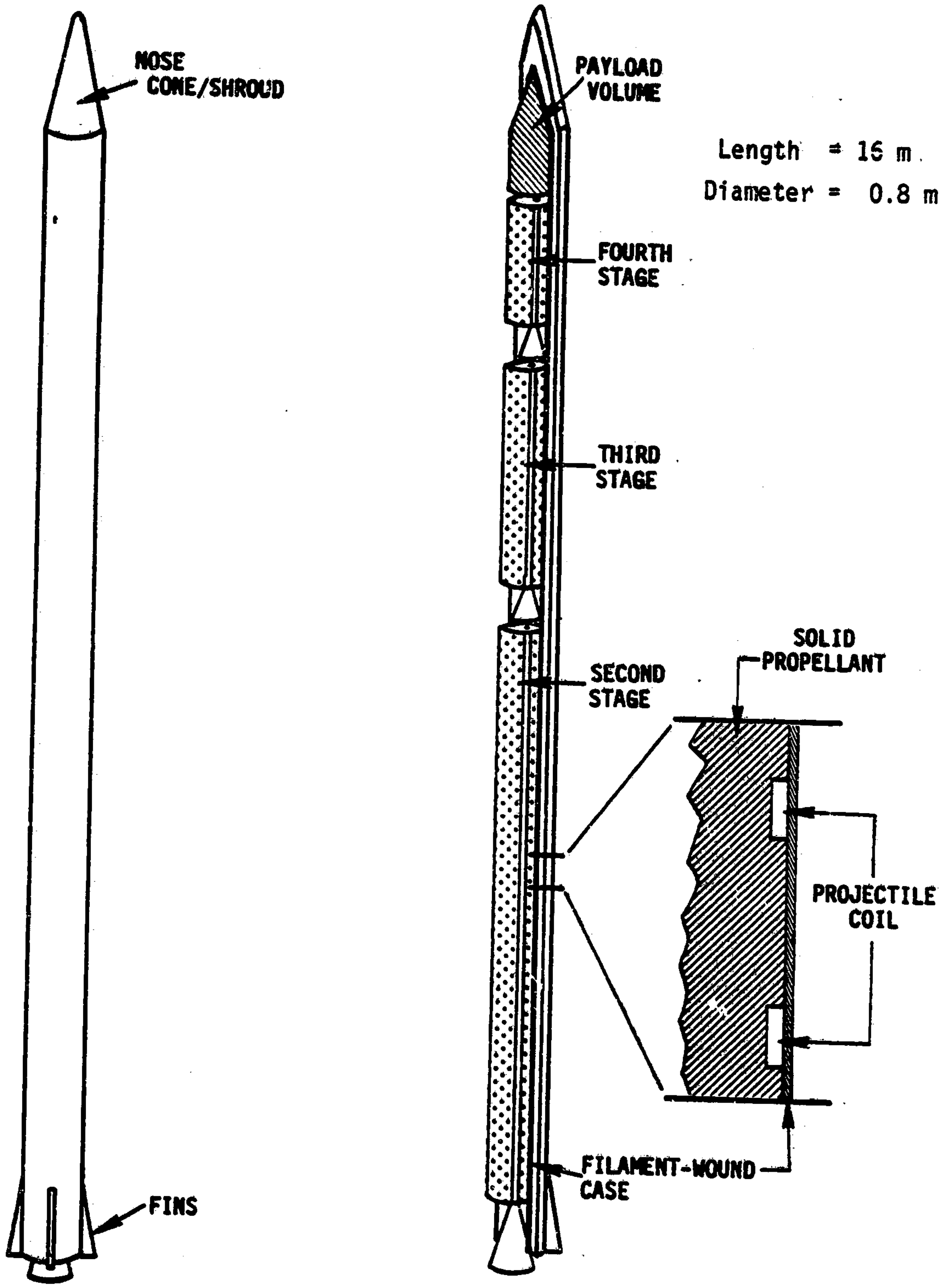


FIGURE 4-18. HYBRID COAXIAL/ROCKET PROJECTILE

4.5.2.2 Surface Transportation Systems

Surface transportation would be accomplished by systems identical to those used for the hybrid railgun/rocket concept. The methods would depend upon the location of the launch site.

4.5.2.3 Launch Site Support Facilities

Launch site support facilities would be similar to the facilities described in Section 4.4.2.3 for the hybrid railgun/rocket concept. The facilities would include: utility power plant substation and/or transmission lines, projectile storage and general storage facilities, administration and engineering offices, industrial area, community living area for a remote site, liquid gas (LN_2 required to cool Brooks coil energy storage inductor) and water plant, and other necessary facilities.

4.5.2.4 Launcher Systems

The hybrid coaxial/rocket system would launch a rocket projectile to a velocity of 2 km/s using a coaxial accelerator. The remaining velocity required to reach low-Earth orbit would be supplied by the three-stage rocket. The launcher would be located on a mountain side with an elevation angle of 35 degrees. Figure 4-19 provides a cross-sectional view of the launcher system.

The drive coils are 2040 m long and made of a copper-alloy material. The drive coils are encased in Kevlar to provide electrical insulation and tube structural containment.

A single large Brooks coil energy storage inductor would supply energy to the drive coils. Switching would be coordinated so that ten turns behind each of the 80 projectile coils would be active at all times as the projectile is accelerated through the launcher tube.

This section describes the following launcher systems:

- Drive coils
- Energy storage
- Launcher support structure
- Preaccelerator system
- Switching and control
- Storage facilities
- Service and access systems.

The hybrid coaxial accelerator/rocket concept options are provided in Figure 4-20.

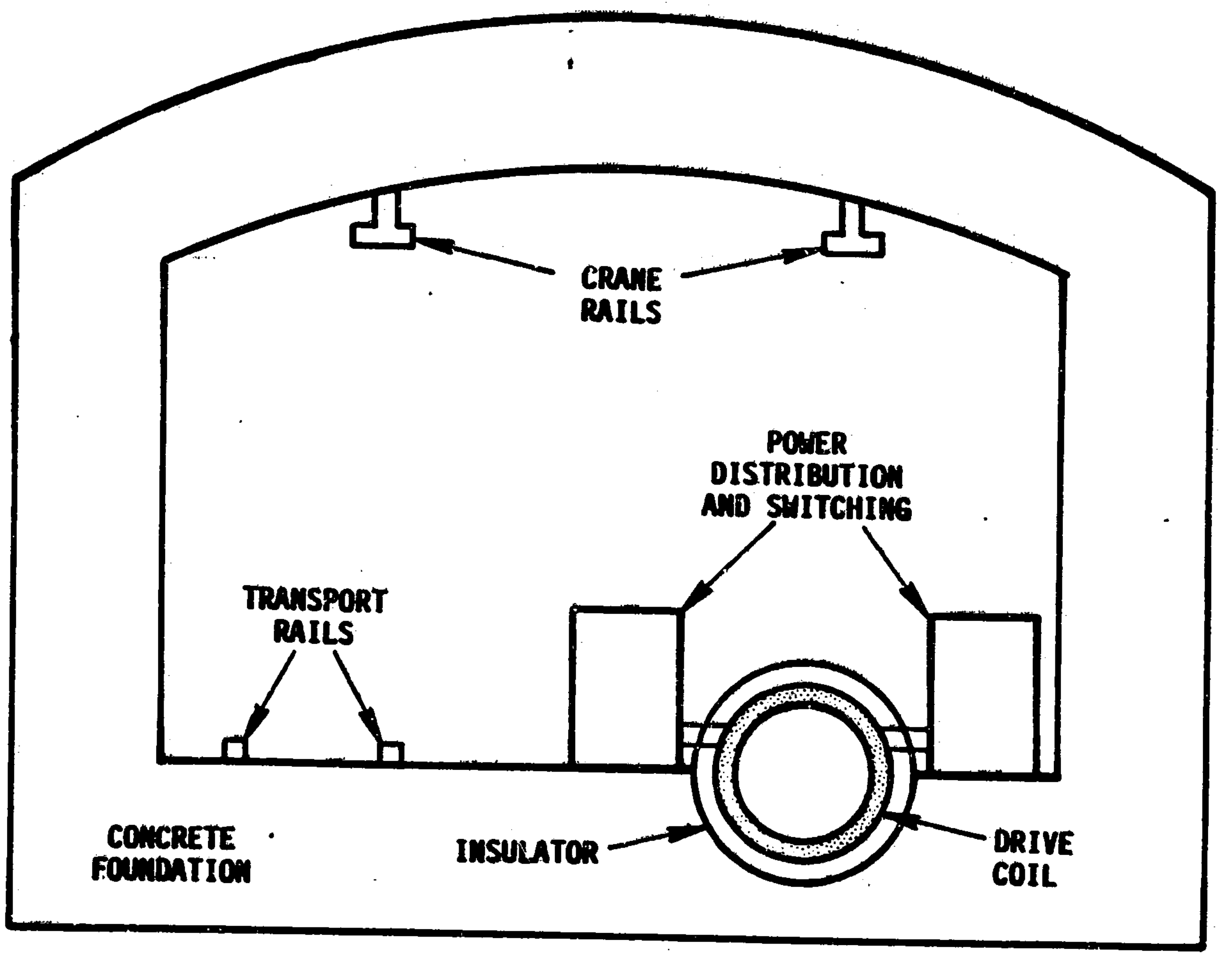


FIGURE 4-19. HYBRID COAXIAL/ROCKET SYSTEM CROSS-SECTION VIEW

ENERGY SOURCE
- * UTILITY POWER
- ● DEDICATED POWER PLANT
  - NUCLEAR
  - FOSSIL FUEL
  - HYDRO
  - SOLAR
  - GEOTHERMAL

ENERGY POWER STORAGE
- * BROOKS COIL
- ● HPG/INDUCTOR
- ● BATTERIES
- ● CAPICATORS
- ● MHD

ENERGY STORE DISTRIBUTION
- * MULTISTAGE
- ● SINGLE STAGE

DRIVER COILS
- * CONTINUOUS HELIX
- ● DISCRETE COILS

PROJECTILE COILS
- * RINGS
- ● COILS (WINDINGS)

DRIVER COIL MATERIALS
- * COPPER ALLOY

DRIVER COIL LINER MATERIAL
- * NON-METALLIC MATERIAL
- ● TEFLON

PROJECTILE COIL MATERIALS
- * ALUMINUM
- ● COPPER

SWITCHING
- * TRIGGERED VACUUM GAP
- ● SOLID STATE DEVICES
- ● THYRATRONS

INSULATOR MATERIALS
- * NON-ASBESTOS FIBER-REINFORCED NON-CONDUCTOR
- ● FERRIBESTOS
- ● KEVLAR
- ● PHENOLIC 6-10

CONFINEMENT SYSTEM MATERIALS
- * KEVLAR
- ● STAINLESS STEEL
- ● STEEL

EVACUATION SYSTEM
- * EVACUATION PUMPS
- ● MECHANICAL PISTON SWEEP

FIGURE 4-20. OPTIONS CHART - HYBRID ROCKET/COAXIAL LAUNCHER

4.5.2.4.1 Drive Coils. The drive coils would be fabricated from a single layer of rectangular copper-alloy wire into a continuous helical winding. Dimensions of the drive coils are: inner radius of 40.2 cm, outer radius of 41.2 cm, length (of helix) of 2040 m. Ten turns would be active behind each projectile coil with an active axial length of 10 cm.

A Kevlar tube structure would encase the drive coils to provide electrical insulation between turns and to contain hoop stresses during launch. The Kevlar structure does not need to be as thick as for the hybrid railgun/rocket because the hoop stresses are lower due to induction effects from the projectile coils.

4.5.2.4.2 Energy Storage. The energy storage device for this concept would be a single Brooks coil inductor. Muzzle kinetic energy of the projectile would be 30 GJ. Again assuming an 85 percent transfer efficiency (see Section 4.3.2.4.2), the energy storage requirement is 35.3 GJ. The Brooks coil would be made of aluminum wire and would be cooled to liquid nitrogen temperatures.

4.5.2.4.3 Launcher Support Structure. Figure 4-19 shows the launcher tube partially embedded in a concrete foundation. The support structure is used to maintain launcher rigidity over its 2-km length and to prevent drive coil damage due to hoop stresses during launch.

4.5.2.4.4 Preaccelerator System. A small preaccelerator system was envisioned to initiate projectile motion. Velocities of no more than 100 m/s were seen to be necessary.

4.5.2.4.5 Switching and Control. Switching and control of currents through the launcher system would be similar to that discussed in Section 4.3.2.4.5 for the Earth-to-orbit coaxial accelerator system.

4.5.2.4.6 Temporary Projectile Storage Facilities. The projectile storage facility located at the base of the launcher would be similar to that described in Section 4.4.2.4.6 for the hybrid railgun/rocket launcher.

4.5.2.4.7 Service and Access Systems. To provide access to the launcher system for maintenance and repair purposes, a rail transport system similar to the hybrid railgun/rocket concept would be used.

4.5.2.5 Monitoring Systems

Trajectory monitoring and orbital tracking systems would be identical to the corresponding rail launcher system (Section 4.4.2.5).

4.5.2.6 Space Destination

The primary payload destination would be a 500-km circular orbit in which a space station was presumed to be orbiting. Other low-Earth orbits would be possible with varying payload masses.

5.0 SAFETY AND ENVIRONMENTAL IMPACT ASSESSMENT

This section documents the preliminary safety and environmental impact assessments of the reference concepts. No significant differences were seen between the railgun and coaxial accelerator concepts; therefore, they are considered to be of similar environmental risk, except where specifically noted. Section 5.1 presents the preliminary safety assessment. The environmental impact analysis is summarized in Section 5.2. The 1982 ESRL study should be referred to for additional data and information regarding safety and environmental impact.

5.1 Preliminary Safety Assessment

A preliminary safety assessment identified the following major accident events:

- Reentry of hardware (deliberate and accidental)
- Projectile break-up
- Liquid propellant spills (Earth-to-orbit concepts)
- Propellant fires at the launch site.

These events were identified as those which could impose a hazard to the human population or the biosphere.

The largest safety risk was seen to be reentry of the nose cone and payload support structure for Earth-to-orbit launches and of the nose cone and spent stages for hybrid systems, as well as the possibility of projectile break-up in the atmosphere with its resultant release of material and the uncontrolled reentry of projectile pieces. Should these events occur over populated areas, the damage could be significant. The launch site location must be sited so as to avoid any overflight of populated areas.

Propellant-related risks would be comparable to those experienced by existing NASA space programs. Adequate safety procedures, similar to those required for other programs, must be used to prevent the possibility of toxic propellant spills and "on-pad" fires. Every effort must be made to avoid toxic exposures to workers and the local uncontrolled human population.

5.2 Environmental Impact Assessment

An in-depth environmental impact assessment of the reference concepts was not possible in this study. However, we did review the reference concepts to assess certain environmental impact areas to determine if any critical issues exist which might prevent EML development for space missions; none were found.

The environmental impact areas for the reference concepts were assessed in two major categories: (1) facility development and

construction and (2) testing and operations. These areas are discussed in the next two sections.

5.2.1 Facilities Development and Construction

The environmental impact for facilities development and construction is highly dependent upon the location at which the site is constructed. No specific location for the launch site was selected in this study. If the launch site were placed in a remote area, the environmental impacts to the area could be significant; however, the overall impact to the quality of the human environment from a remote site would likely be lower than the impact from a non-remote site. Major impacts for launch site development might involve the following: relocation of inhabitants, destruction of vegetation and wildlife habitats, extinction of local animal species, and disturbance of archeological sites. Site selection criteria should take account of these possible impacts and could be used to minimize these effects.

The types of facilities that are anticipated for the concepts are listed below:

- Launcher system
- Power plant and utility lines
- Roads
- Airfield and rail lines
- Buildings.

The types of environmental impacts caused by the construction of these facilities should be typical of any construction activity in an undeveloped area. The construction of the launcher system and airfield runways could pose significant impact to the area. The construction of power plant or utility lines, roads, buildings, and rail lines is not expected to pose significant environmental effects.

Materials usage must also be addressed. Although large amounts of aluminum, copper, and steel are necessary for construction of the launcher systems, the impact to U.S. and world annual consumption estimated for the year 2000 is expected to be minimal. The materials requirements for construction of each of the reference concepts are presented in Table 5-1.

5.2.2 Testing and Operations

This section discusses the expected environmental impacts from normal testing and operation of the reference concept systems. The impacts were assessed assuming the fifty-year average of flights per year from the traffic models shown in Tables 2-5 and 2-18. Ten flights per day were assumed for the Earth-to-orbit systems and five flights per day were assumed for the hybrid EML/rocket systems. Areas of concern relating to this impact area include:

TABLE 5-1. FACILITY CONSTRUCTION MATERIALS REQUIREMENTS

Concept	Material	Estimated Material Requirement, MT	Fraction of Annual U.S. Consumption by the Year 2000[a]	Fraction of Annual World Consumption by the Year 2000[a]
Earth-to-orbit Coaxial	Aluminum	96000	0.00500	0.00160
	Copper	300	0.00006	0.00001
Earth-to-orbit Railgun	Aluminum	3600	0.00019	0.00006
	Copper	5300	0.00098	0.00019
	Iron	61000	0.00041	0.00006
Hybrid Coaxial	Aluminum	38000	0.00200	0.00060
	Copper	500	0.00009	0.00002
Hybrid Railgun	Aluminum	750	0.00004	0.00001
	Copper	3000	0.00056	0.00011
	Iron	7500	0.00005	0.00001

(a) From Teeter and Jamieson, 1980.

- Sonic booms
- Power plant emissions
- Chemical effluents
- Solid waste disposal
- Materials usage
- Orbital debris (hybrid EML/rocket concepts).

The following subsections discuss each of these concerns.

5.2.2.1 Sonic Booms

The impact of sonic boom generation from the EML launches was investigated. The magnitude of the sonic booms produced by the reference concepts was calculated following the procedure described in Section 5.3.2.1 of Rice, el al (1982). The equations used in the assessment were derived from Sedov (1959). The derived pressure-rise equation was:

$$\Delta p = \frac{0.082 \rho V^2 C_d d^2}{x^2}$$

where Δp = pressure rise
ρ = atmospheric density
V = projectile velocity
C_d = drag coefficient
d = projectile diameter
X = radial distance from the disturbance.

Table 5-2 lists the distances calculated for various sonic boom overpressures. The critical distances were calculated assuming a drag coefficient of 1.0 which represented the sabot jettison for the Earth-orbital railgun concept and the stabilization fin extension for the other concepts. Large differences in critical distance are evident between the Earth-orbital concepts and are explained by the factor-of-two difference in projectile diameter (1.0 m for the railgun projectile and 0.5 m for the coaxial projectile). The hybrid projectile diameters are similar (1.0 m for railgun and 0.8 m for coaxial).

The overpressure limits were provided in CPIA (1972). At an overpressure of 20.7 N/cm^2, the lethal threshold is reached. Persons within the critical distance for lethality (within 15 m for the worst case--Earth-orbital railgun concept) would likely be killed. An average human being would experience ruptured eardrums at 3.45 N/cm^2 within 40 m of the launcher muzzle for the worst case. Window breakage of typical glass would occur at 0.345 N/cm^2, while "uncontrolled areas" would typically experience 0.138 N/cm^2. Sonic booms on the order of supersonic aircraft at high altitudes would occur between 165 and 700 m

TABLE 5-2. CRITICAL DISTANCES FOR EXPECTED SONIC BOOM OVERPRESSURES

Type of Limit	Limit Over Pressure(a) (N/cm^2)	Critical Distance from Launcher Muzzle (m)(b)		Critical Distance from Launcher Muzzle (m)(b)	
		Earth-to-Orbit Railgun	Earth-to-Orbit Coaxial	Hybrid Railgun	Hybrid Coaxial
Lethal	20.7	15	10	5	5
Eardrum Rupture	3.45	40	20	10	10
Window Breakage	0.345	120	60	35	30
Typical-Uncontrolled Areas	0.138	190	95	55	45
Typical-Supersonic Aircraft	0.010	700	355	205	165

(a) From CPIA, 1972.

(b) Assumptions - Earth-to-Orbit $V = 6.85$ km/s
$d = 1.0$ m (Railgun)
$= 0.5$ m (Coaxial)

- Hybrid $V = 2.0$ km/s
$d = 1.0$ m (Railgun)
$= 0.8$ m (Coaxial)

- Rounded to nearest 5 km

- Sea-level air density

depending upon which reference concept is assumed. At distances of the order of hundreds of meters away from the launcher muzzle, most animals would leave and seek other habitats. People living or working within several kilometers would likely be annoyed by the boom. Based upon this preliminary assessment, it is concluded that localized damage to the biosphere would be evident in the region near the muzzle of the launcher and that animal species in the vicinity of the muzzle would probably migrate to other locations. Effects at larger distances are not believed to be significant.

5.2.2.2 Power Plant Emissions

A nuclear power plant was assumed for the Earth-orbital reference concepts. Normal emissions from nuclear reactors are not expected to pose significant hazard to the environment. The radiation dose to plant workers and the risk of accidents are not expected to be different from other nuclear facilities currently operating under federal guidelines. The hybrid EML/rocket concepts assume the use of utility power. The environmental impact should be similar to existing power plants. Emissions from the power plant are not expected to be of significant environmental impact.

5.2.2.3 Chemical Effluents

The major chemical effluent resulting from testing and operations are those related to the use of solid propellant in the hybrid systems. HCl emitted from the solid propellant motors at high altitude could cause a reduction in the ozone concentration in the stratosphere and cause ionospheric disturbances. Hybrid solid systems, however, would be of less consequence than an all solid launch vehicle that burns propellant in the lower toposphere. Also, it is expected that various types of cleaning solvents and various propellant contaminants could be released into the biosphere (air and water). These activities are not expected to be of major significance and are expected to be comparable to those of current Space Shuttle launch activities.

5.2.2.4 Solid Wastes

It is expected that solid wastes, including the production of waste propellant in solid motor production for the hybrid system, would be similar to those from current Space Shuttle operations or from typical industrial operations. No significant environmental impact is expected from the generation of solid waste produced from the testing and operation of the reference concept systems.

5.2.2.5 Materials Usage

Table 5-3 presents the major material requirements for the Earth-to-orbit reference concept projectiles at the rate of ten launches per day and compares these requirements with the estimated annual U.S. and world consumption in the year 2000. The major materials requirement

TABLE 5-3. **MAJOR OPERATIONAL MATERIALS REQUIREMENTS FOR EARTH-TO-ORBIT EML REFERENCE CONCEPTS (HYBRID CONCEPTS NOT INCLUDED)**

Material	Estimated EML Requirement, MT	Fraction of Annual U.S. Consumption by the Year 2000[a]	Fraction of Annual World Consumption by the Year 2000[a]
Iron	18740[b]/2543[c]	0.00006/0.00001	0.00001/0.000001
ClF_3	3103	(d)	(d)
$-Cl_2$	1188	0.00004	0.00001
$-F_2$	1915	0.0011	0.00029
N_2H_4	1095	0.12[e]	(d)

(a) From Teeter and Jamieson, 1980.
(b) Railgun requirement.
(c) Coaxial requirement.
(d) Data not available in Teeter and Jamieson, 1980.
(e) From Faith, 1965.

for the hybrid EML/rocket concepts is solid propellant. At five launches per day, 23,180 MT of solid propellant is required per year. At first glance, this seems to be a large amount; however, one Space Shuttle launch uses 1010 MT of solid propellant. The annual hybrid EML/rocket requirement is then approximately equal to 23 STS launches which are reasonably expected by 2000. The solid propellant ingredients are aluminum (roughly 16 percent), polybutadiene (about 14 percent), and ammonium perchlorate (approximately 70 percent) which are common materials.

Although significant upgrade may be necessary to produce certain materials, namely ClF_3, N_2H_4, and solid propellant, the use of these items is not critical to the consumption of the Earth's resources.

5.2.2.6 Reentry of Hardware

This issue was previously addressed in the safety assessment (Section 5.1) and in the mission requirements analyses (Section 3.5.2). During testing and operational activities, the nose cone and steel payload support structure would be jettisoned and would reenter for every launch of the Earth-orbital concepts (ten per day average). Spent stages and nose cones would reenter for hybrid EML/rocket launches (average of five per day). Minimal risk to the population and the potential for recovery of materials (if economical) would occur if the launch site and resulting landing areas were properly selected (see Section 3.5.2).

5.2.2.7 Orbital Debris

Orbital debris is not expected to be a problem for the Earth-to-orbit concepts. An Orbital Maneuvering Vehicle (OMV) would rendezvous with the payload on orbit and transport it to the Space Station. The propulsion systems would be returned to Earth via the Shuttle.

Solid rocket exhaust particulate is currently receiving attention in the orbital debris area. The micron-size particulate may cause pitting of spacecraft and damage to solar arrays. The final stage burn would be an orbital insertion maneuver into the Space Station orbit. Most of the particulate would deorbit; however it is possible that some particulate would remain in orbit and impact the Space Station systems. Further analysis is required, including detailed trajectory analysis and an investigation of alternatives such as a liquid-propellant upper stage. This debris issue should not prevent hybrid EML/rocket development.

5.3 Conclusions

Based upon the preliminary safety and environmental impact assessments for the reference concepts, no significant safety or environmental impact problems have been found. Population overflight must be considered when selection of the launch site is made. The initial construction of facilities is expected to create some environmental

impact to the local area; however, this is not expected to be significant. Sonic booms would create localized problems for animals surrounding the launcher system, but few effects are expected on the human population. No major issues have been found thus far in the safety and environmental impact evaluation to prevent EML development for space missions. Economics appears to be the most important non-technical issue.

6.0 EML COST ESTIMATES

This section documents cost estimates for the coaxial and railgun electromagnetic launchers and examines the cost competitiveness of the EML concepts as applied to near-term (2000-2010) bulk supply missions. The EML concepts are compared to conventional chemical launch systems, such as the Space Shuttle and a solid-propellant rocket.

Costs for the EML reference concepts (Earth-to-orbit EML and hybrid EML/rocket concepts) using railgun and coaxial accelerators have been estimated according to the Work Breakdown Structure (WBS) in Tables 6-1 and 6-2. The resulting investment costs estimates are shown in Table 6-3. The goal of these estimates is to provide a comparison of the railgun and coaxial technologies in the area of costs, and to examine the cost competitiveness of these concepts versus fully chemically powered launch systems such as the Space Shuttle and an all-solid-rocket launch vehicle. The cost information developed here relies strongly on our previous investigation of railguns (Rice, Miller and Earhart, 1982).

Both the ESRL report and this report use 1981 dollars. To adjust total cost estimates given in 1981 dollars to 1984 dollars, multiply by 1.16 to reflect Consumer Price Index inflation of about 5 percent per year. Material costs, however, have not exhibited uniform inflation, and in some cases are lower than in 1981.

The Work Breakdown Structure does not include research, technology development, and design efforts prior to formal initiation of an EML development. These are not included because the research is applicable to many other activities and some of these costs may be paid by those activities, and because advanced research and technology costs are highly uncertain.

The research and design costs used for the Earth-to-orbit EML designs are the same as used in the ESRL report, about 10 percent of the initial investment for that high-capability system. The expected cost for research and design is \$466 M, and the low and high estimates are \$320 M and \$633 M. The low and high costs are believed to represent the 90 percent confidence interval for research and design expenditures, while the expected value approximates the mean of the cost distribution. This interpretation is given to all use of the terms low, expected and high cost estimates. For the hybrid EML/rocket, the research and design costs are expected to be somewhat lower, reflecting the lower level of capability required. The costs are assumed to be two-thirds of the estimates for the Earth-orbital EMLS. This procedure gives a range from \$200 M to \$400 M with the expected value \$300 M. Project resources do not permit the detailed investigation of the resource requirements for research and design for either concept. The comparison of the associated costs of the railgun and coaxial EML concepts is conducted at the system level with every attempt to keep the systems mission-equivalent. At the present level of understanding of these concepts, it appears that

TABLE 6-1. EML DEVELOPMENT AND INVESTMENT WORK BREAKDOWN STRUCTURE (WBS)

1.0 Facilities and Supporting Systems

- 1.1 Land
- 1.2 Power Plant or Substation
- 1.3 Personnel Support Facilities (housing, roads, sanitation, school)
- 1.4 Shipping Docks, Storage, and Transportation Facilities
- 1.5 Airfield and Hanger
- 1.6 Industrial Area (Equipment Refurbishment)
- 1.7 Administration/Engineering Buildings

2.0 Launcher Systems

- 2.1 Mountainside Structures
  - 2.1.1 Launcher Tube Housing
- 2.2 Launcher Tubes
  - 2.2.1 Copper Alloy Conductors (rails or coils)
  - 2.2.2 Spacers-Insulation
  - 2.2.3 Kevlar Containment
  - 2.2.4 Vacuum Container and Exterior Insulation
- 2.3 Launcher Energy Storage (includes hydraulic motors and hydraulic distribution) and Supporting Structures
- 2.4 Inductors and Switches (includes LN_2 distribution system)
- 2.5 Preboost System
- 2.6 Power Conversion Facilities
- 2.7 Water Distillation Plant
- 2.8 Gas Handling Facilities
  - 2.8.1 Liquid Nitrogen Plant and Storage
  - 2.8.2 Evacuation System for Launcher Tube
  - 2.8.3 Water Electrolysis Plant
- 2.9 Elevator Systems and Projectile Handling Devices
- 2.10 Control Center, Controls, and Monitoring Systems
- 2.11 Tracking Systems

TABLE 6-2. EML DEVELOPMENT TEST PROGRAM AND OPERATIONS WORK BREAKDOWN STRUCTURE (WBS)

3.0 Projectiles and Mission Peculiar Equipment

4.0 Operations

 4.1 Management and Support

  4.1.1 Management
  4.1.2 Engineering
  4.1.3 Facility Support

 4.2 Power Plant Operations (Supplies and Crew)

 4.3 Technical Personnel and Supplies

  4.3.1 Control Center Crew
  4.3.2 Launcher Equipment Support Crew
  4.3.3 Equipment Refurbishment Crew
  4.3.4 Power Conversion Facility Crew
  4.3.5 LN_2 Plant/Vacuum System Crew
  4.3.6 Projectile/Payload Operations Support Crew
  4.3.7 Facility Utilities Crew

5.0 Development Test Program

 5.1 Test of Launcher Segment(s)
 5.2 Development of Projectiles
 5.3 Transient Housing at Launch Site
 5.4 Launcher Operations Costs During Tests

TABLE 6-3. EML INVESTMENT COST ESTIMATES ($, M, 1981)

	Earth Orbital Launcher						Hybrid EML/Rocket Launcher					
	Coaxial			Railgun			Coaxial			Railgun		
	Low	Expected	High	Low	Expected	High	Low	Expected	High	Low	Expected	High
Facilities & Supporting Systems												
Land	9.6	12.0	16.0	9.6	12.0	16.0	3.2	5.0	6.4	3.2	5.0	6.4
Power Plant or Substation	220.0	250.0	280.0	220.0	250.0	280.0	66.0	75.0	84.0	66.0	75.0	84.0
Personnel Support Facilities	30.0	53.0	90.0	30.0	53.0	90.0	30.0	53.0	90.0	30.0	53.0	90.0
Transport and Storage Facilities	60.0	130.0	200.0	60.0	130.0	200.0	60.0	130.0	200.0	60.0	130.0	200.0
Industrial Area	40.0	60.0	80.0	40.0	60.0	80.0	40.0	60.0	80.0	40.0	60.0	80.0
Administrative/ Engineering Buildings	2.0	5.0	8.0	2.0	5.0	8.0	2.0	5.0	8.0	2.0	5.0	8.0
Subtotal	361.6	510.0	674.0	361.6	510.0	674.0	201.2	328.0	468.4	201.2	328.0	468.4
Launcher Systems												
Mountainside Structures	38.0	57.0	76.0	38.0	57.0	76.0	38.0	57.0	76.0	38.0	57.0	76.0
Launcher Tubes	15.3	19.1	23.1	152.0	198.8	418.4	24.7	31.0	37.4	90.7	115.8	239.9
Launcher Energy Storage	248.5	331.4	497.2	425.6	532.0	638.4	93.7	124.8	187.3	94.0	168.0	201.6
Switching	16.0	20.0	24.0	76.0	95.0	114.0	16.0	20.0	24.0	76.0	95.0	114.0
Preboost System	1.0	2.0	3.0	80.0	100.0	120.0	1.0	2.0	3.0	1.0	2.0	3.0
Power Conversion	--	--	--	27.5	31.3	35.0	--	--	--	8.3	9.4	10.5
Water Distilling Plant	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5
Fluid and Cryogenic Systems												
- Liquid Nitrogen/ Cryogenics Plant	12.6	14.3	20.8	18.5	22.2	39.0	7.0	7.2	7.8	7.0	7.2	7.8
- Vacuum System	1.0	1.5	1.5	1.0	1.5	1.5	1.0	1.5	1.5	1.0	1.5	1.5
- Water Electrolysis Plant	--	--	--	0.2	0.3	0.4	--	-	--	--	--	--
Handing Devices/Systems	2.2	6.6	11.0	2.2	6.6	11.0	2.2	6.6	11.0	2.2	6.6	11.0
Control Center, Controls, and Monitoring	8.0	21.0	38.0	8.0	21.0	38.0	6.0	10.0	15.0	6.0	10.0	15.0
Tracking System	5.0	10.0	15.0	5.0	10.0	15.0	5.0	10.0	15.0	5.0	10.0	15.0
Subtotal	350.3	485.4	712.1	836.5	1078.2	1509.2	197.1	272.6	380.5	331.7	485.0	697.8

the coaxial accelerators provide lower levels of stress (for normal operations) on both the launcher and the projectile. This lower level of stress leads to lower design requirements and less mass in the launcher and projectile, and therefore a lower level of costs. It also appears that the non-kinetic electrical energy storage devices (Brooks coils) for the coaxial concepts cost less than the homopolar generators (HPGs) used for the railgun concepts. Railgun technology, however, has been demonstrated at higher accelerations and velocities than has coaxial technology and the railgun technology may yet prove to have other advantages which may not be apparent at the present time, especially for high energy missions. The system cost estimates here include: (1) system development and construction, (2) an initial flight test program, and (3) thirty years of operations. A cost summary section presents an overview of costs developed and the cost per unit mass of payload.

6.1 Development and Investment Cost Estimates

This section discusses the derivation of development and investment costs for facilities and supporting systems and for the launcher systems. Low, expected and high estimates are presented in Table 6-3. The low and high estimates can be considered an estimate of the 90 percent confidence interval for expected costs. To maintain visibility of the cost differences between railgun and coaxial systems, facilities costs are the same for both technologies unless there is an obvious reason to change them.

6.1.1 Facilities and Supporting Systems

Six basic categories are considered and costed (1) land, (2) electrical power, (3) personnel support, (4) transport facilities, (5) industrial area, and (6) administrative/engineering buildings and access roads.

6.1.1.1 Land

For all concepts a mountainous island or a mainland site near the equator is assumed. Slopes of 20 degrees (Earth-to-orbit EML) and 35 degrees (hybrid EML/rocket) would be required for distances of at least two kilometers. For the hybrid EML, a site at 28.5 degrees latitude may be acceptable to facilitate launches to orbits with this inclination, but the number of launch opportunities for a single space station would be limited to one per day. For an equatorial site, there would be 16 launch windows per day to a single equatorial space station. An equatorial site would also be desirable for launches to higher orbits such as geosynchronous equatorial orbit, but these missions are outside the primary scope of this investigation.

While free use of land is a possibility, a cost of $2470 per hecture ($1000 per acre) is assumed. Because of its power plant requirement, an Earth-to-orbit launcher facility is expected to occupy from 24 to 40 km^2 (15 to 25 sq mi) while the hybrid EML/rocket launcher

facility is expected to occupy from 12 to 25 km^2 (5 to 10 sq mi). The estimated costs for land for the Earth-orbital facility would then range from \$9.6 M to \$16.0 M with an expected cost of \$12.0 M. For the hybrid EML facility the land cost estimates range from \$3.2 M to \$6.4 M with an expected cost of \$5 M.

6.1.1.2 Power Plant

For the hybrid EML/rocket launcher, the power requirements are such that an independent base-load plant would not be efficiently used. Use of commercial power or sale of excess power from a joint venture power plan would be desired for this concept. For the Earth-to-orbit EML, 100 MWe capacity is considered necessary for all purposes and this should be divided into two 50 MWe units to provide a maintenance reserve. (The energy requirement for a launch to LEO is about 42.4 MWh.) Civil reactor availability currently runs about 66 percent due to both scheduled (about 20 percent) and unscheduled (about 14 percent) maintenance. For economic reasons, civil power reactors are larger than 200 MWe, but naval and other small nuclear reactors are believed to be available in the appropriate size range. Given the use of two independent reactors, availability of power from at least one reactor should be about 90 percent $[1-(1-0.66)^2 = 0.89]$ and should improve in the future. For further discussion of the factors influencing this choice, see Rice, et al, 1982, pages 6-7 and 6-8.

For the hybrid EML/rocket concept, 15 to 30 MWe of capacity would be required for both the EML itself and for personnel support. Clearly, use of commercial power would be desired because baseload plants below 100 MWe are not considered to be economical in this country. Based on available documentation of plant costs (Friedlander, Electrical World, October, 1981), a 200MWe baseload nuclear plan is projected to cost from \$2200 to \$2800 per kWh capacity with \$2500 per kWh expected.

A 100-MWe baseload coal plant is expected to cost from \$1400 to \$1800 per kWh, with an expected cost of \$1600 per kWh of capacity (Friedlander, 1981). Extrapolating the curves given by Friedlander for coal-powered units to 30 MWe, however, leads to estimates of \$2000 to \$2600 per kWh of capacity, with an expected value of \$2300 per kWh.

Accordingly, the uniform cost per kWh of capacity used ranges from \$2200 to \$2800 with an expected value of \$2500 per kWh. This leads to estimates of \$220 M to \$280 M with \$250 M expected for the Earth-to-orbit EML facility with a 100-MWe nuclear power plant. For the hybrid EML/rocket facility, a 30-MWe non-nuclear plant would be expected to cost from \$66 M to \$84 M, with an expected cost of \$75 M. If, as expected, commercial power is used, these estimates would be high, and an estimate of \$10 M to \$25 M, with an expected value of \$15 M might be expected to cover transmission lines, a substation, and a small amount of emergency outage power-generation capability. This estimate is based upon twenty percent of the estimates for providing power at the site and is in turn based upon Friedlander, 1981, and other articles in

Electrical World. The estimates are strongly dependent upon the launch site being located within a reasonable service distance from the commercial plant or major transmission line. To preserve comparability between concepts, the higher estimates are used.

6.1.1.3 Personnel Support Facilities

Personnel support facilities include housing, roads, sanitation, and school buildings. It is assumed that there would be a permanent workers' community with reasonable amenities, and that most workers would work in some sort of building. The cost of these facilities is estimated at $100,000 per permanently stationed employee. If the facility could be located in the U.S., some of this cost would be borne by the workers and local government, but most missions under consideration appear to require a location outside the continental U.S.

Railgun concepts, because they use homopolar generators, would require a maintenance crew larger than foreseen for coaxial concepts, but this impact on facilities costs is expected to be small. Accordingly personnel facilities are costed on the basis of personnel estimates for launchers as given in Section 6.2. From 300 to 900 personnel are estimated, with an expected number of 530. This leads to personnel facility estimates of from $30 M to $90 M, with an expected estimate of $53 M.

6.1.1.4 Transport Facilities

Transport facilities capable of handling large objects in relatively high volume over short periods of time would be required. These include air cargo, land and shipping facilities appropriate to the location of the launcher. To the extent that existing facilities can be used, additional costs could be avoided. Sea, air, and land facilities are discussed below are estimated to cost from $60 M to $200 M with an expected cost of $130 M.

6.1.1.4.1 Shipping Docks, Storage and Transport Facilities. The cost of surface transport and storage facilities required would depend on the specific features of the site, such as, terrain, how much development already exists, and whether there is a natural harbor. While the initial site development could probably be supported by small ships, it is expected that a protected pier would be required for the construction phase. Accordingly, these facilities are estimated in the range from nothing to $40 M with a $20 M expected cost. For example, if the site were located in northern Mexico, a port would probably not be needed. For equatorial sites, it is very likely that, at a minimum, existing port facilities would need to be improved.

6.1.1.4.2 Land Transport Facilities. Two-lane heavy-traffic roads cost from $1 M to $9 M per kilometer depending on the nature of the terrain (Ohio Department of Highways). Heavy-duty rail lines cost from $4 M to $7 M on terrain suitable to rail use. Because a specific

site is not proposed here, detailed land transport cost estimates cannot be made. An allowance of 20 miles of roads and/or railroads is made with cost estimated in the range of $40 M to $80 M with an expected cost of $60 M.

6.1.1.5 Airfield and Hangar

Since high-value or sensitive cargos would not be used, an airport capable of accommodating frequent flights of the largest aircraft would not be necessary. The local geography would be the major determinant of the airfield cost, and this cost could vary by as much as a factor of ten depending upon the site chosen.

The ESRL report provided for an airport which could handle the largest standard cargo aircraft and had two 6000 m runways, taxiways, 100 m x 100 m hangar, and a fuel depot; the airport was estimated to cost from $26.3 M to $100 M with an expected estimate of $56 M. Because the facility is not envisioned for transporting hazardous and/or sensitive payloads, it is probable that one runway would suffice and that the cost could be constrained within estimates of $20 M to $80 M with an expected estimate of $50 M. Cost considerations, given in the ESRL report, include $3000 to $5000 per lineal meter of runway with minimal soil preparation, factors of 1 to 3 applied to these estimates for grading, $450/m^2 for hangar construction, $1 M for the fuel depot, and ten percent of runway costs for taxiways.

6.1.1.6 Industrial Area

Since the EML concept would employ a considerable amount of moving machinery (such as homopolars, gas liquefaction compressors, etc.), numerous maintenance and repair activities are anticipated. Thus, a facility which could repair and refurbish the equipment would be needed. There would also be a need to store replacement hardware components in a warehouse. Because of the uncertainty of the requirements for this facility, it is arbitrarily estimated at $40 M to $80 M with an expected cost of $60 M, including both buildings and industrial equipment.

6.1.1.7 Administration/Engineering Buildings

The administration and engineering functions are expected to reach a peak during development and initial operations, and then drop to a lower level as initial operational problems are resolved. Activities would rise to higher levels only if additional demand, justifying new or replacement launchers, is achieved. The engineering development staff would most likely be accommodated in inexpensive buildings which could be used later to accommodate transient personnel during the operational phase. Since the initial motivation for construction of buildings would likely be the development test program, an estimate of $3 M to $5 M is charged to the development test program.

For the permanent staff, however, there are expected to be from 100 to 400 people who will need permanent office or other working

space. This is expected to cost about $20,000 per worker, resulting in an administration/engineering buildings cost of $2 M, $5 M, and $8 M for low, expected, and high estimates of 100, 250, and 400 workers needing these facilities.

6.1.2 Launcher Systems

All launcher systems are considered to be constructed along mountain sides. This would avoid the considerable cost of constructing tunnels over 2 km long as well as extensive underground access and working areas. This cost-reduction potential, however, restricts the number of potential sites, because slopes of approximately 20 degrees will be required for the Earth Orbital launcher concepts and slopes of approximately 35 degrees will be required for the hybrid EML/rocket concepts. Despite their appearance, mountain slopes of 20 to 35 degrees over a length of 2 km or more are relatively rare, and other geographical factors such as slope orientation and down-range safety zones further restrict the number of acceptable sites. An artificial mountain with a 2 km-long slope at 35 degrees would be approximately 1 km high. The earth-moving and stabilization problems associated with constructing an artificial mountain 1 km high would result in higher costs than tunneling.

6.1.2.1 Launcher Tube Housing

For all concepts, the launcher tube and associated equipment would need to be firmly anchored to the side of the mountain and have a substantial cover from which overhead cranes can hang to move launcher equipment. Such a structure is assumed to be at least 2.1 km long to accommodate equipment at the breech and muzzle. This housing is expected to have the complexity of a four-lane superhighway structure carrying elevated traffic. Four-lane superhighways currently cost from $2.5 M to $18 M per km ($4 M to $30 M per mile) (Ohio Department of Highways) with the lower figure representing construction on level farmland and the higher figure representing elevated structures. The launcher housing would be built in a remote, mountainous area and would require substantial amount of site preparation. For a 2.1 km length, this leads to estimates of two to four times the maximum superhighway cost ($38 M to $76 M, with $57 M the expected cost). (If a subterranian complex were to be selected the facility construction cost is estimated to cost from $250 M to $540 M with an expected cost of $300 M--see discussion on pages 6-11 to 6-13 of Rice, et al, 1982.)

6.1.2.2 Launcher Tubes

All concepts would use launcher tubes with copper alloy conductors, insulation to hold the conductors in place, a Kevlar wrapping to contain normal and some accidental launch forces, and a vacuum container. The coaxial designs would operate at a lower current and higher voltage than railgun launchers. Since launch stresses are

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proportional to the square of the current, and the coaxial launcher's lower current stresses are reduced by induction effects of the projectile coils, this results in a much lower level of radial forces on the launcher tube. The coaxial launcher tube, however, must still be protected against the effects of accidental currents and accidental ignition of propellants. The additional wrapping to contain accidental overloads reduces the potential savings in tube construction costs that could be possible if overloads were not considered. Launch stresses for railguns are much higher than for coaxial launchers. Accordingly, the stress confinement wrappings allocated for normal stress confinement for the railgun designs are considered to be adequate for accident protection.

Calculations of the launcher tube costs are presented in Table 6-4. The tubes would have common components--conductors, insulation force confinement wrapping and an exterior vacuum container which would also provide a mechanical connection to adjacent tube segments. It is expected that the tubes would be manufactured in segments of convenient length to permit replacement of worn or damaged segments. The launchers would have circular bore cross sections with the exception of the railgun for Earth-orbital missions, which would have a square-bore cross-section. The square-bore design is carried over from the ESRL report. Launcher tube configurations and their cost implications are discussed in the following four sections.

<u>6.1.2.2.1 Copper Alloy Launcher Conductors</u>. The launcher conductors would be subjected to brief current pulses which should not provide so much heat that active cooling is required. The material selected is AMZIRC which is approximately 99.85 percent copper and 0.15 percent zirconium (Engineering Alloys Digest, Inc., 1961). Based on a typical price for copper forms of $1.76 per kg and a price for zirconium of $16.50 per kg, the conductors would cost $1.78 per kg. The density for AMZIRC is taken as 8.96 g/cc.

It is expected that the conductors would be formed, heat treated, surface machined, and later assembled into a complete tube segment amenable to handling and transport. Because these are traditional manufacturing practices and no advanced technology appears to be involved, the appropriate labor factor for fabrication and installation in quantities this large is in the range of 2 to 3; a labor factor of 2.5 is the midpoint and is used to form the expected cost.

<u>6.1.2.2.2 Electrical Insulation</u>. The rails or drive coils would require mechanical support and electrical insulation. For the coaxial launchers, where the voltages are high and the forces relatively low, it is expected that a synthetic rubber or plastic should suffice. Railgun launchers will need insulative materials with high compressive strength. Many potential candidates currently contain asbestos, which is considered to have unacceptable human health concerns in most applications. It is expected that a substitute can be found and would cost less than $1.00 per kg and would have a density in the range of

TABLE 6-4. LAUNCHER TUBE COST CALCULATIONS (2000 m TUBE)

	Radius (Thickness) (m)	Volume (m^3)	Density (MT/m^3)	Mass (MT)	Cost per kg ($/kg)	Materials Cost ($, M)	Labor Factors	Cost Range, $, M Low	Expected	High
Coaxial Launchers - Earth Orbital										
Winding (CuZr)	0.218(0.01)	28.0	8.96	251.0	1.78	$0.447 M	2,2.5,3	0.900	1.100	1.300
Insulation	0.228(0.02)	59.8	1.2	71.8	2.20	$0.157 M	1,1.5,2	0.157	0.236	0.315
Force Confinement (Kevlar)	0.248(0.05)	179.0	1.38	247	28.20	$7.000 M	2,2.5,3	14.000	17.500	21.000
Vacuum Conf. (Al)	0.300(0.01)	38.3	2.70	103.5	1.68	$0.173 M	1.5,2,3	0.260	0.350	0.520
TOTAL								15.317	19.186	23.135
- Hybrid EML/Rocket Launcher										
Winding (CuZr	0.402(0.01)	51.1	8.96	458	1.78	$0.815 M	2,2.5,3	1.630	2.037	2.445
Insulation	0.412(0.02)	106.0	1.2	127.2	2.20	$0.280 M	1,1.5,2	0.280	0.420	0.560
Force Confinement (Kevlar)	0.432(0.05)	287.1	1.38	396.3	28.20	$11.175 M	2,2.5,3	22.350	27.938	33.525
Vacuum Conf. (Al)	0.482(0.01)	61.2	2.70	165.2	1.68	$0.277 M	1.5,2.3	0.417	0.555	0.832
TOTAL								24.677	30.950	37.362
Railgun Launchers - Earth Orbital										
Rails (CuZr)	(2)·(1.2)·(0.25)	1200	8.96	10752	1.78	$19.138 M	2,2.5,3	38.276	47.845	57.414
Insulation	1.5 outside	11.540	3-5	34,620-57,700	1.00	$34.620-57.700 M	1,1.5,2	34.620	51.930	115.400
Force Confinement (Kevlar)	1.5(0.05-0.1)	958.1-1974.7	1.38	1379-2804	28.20	$38.910-79.096 M	2,2.5,3	77.82	97.275	237.288
Vacuum Conf. (Al)	1.55(0.01-0.03)	195.4-608.8	2.70	527.6-1643	1.68	$0.836-2.761 M	1.5,2,3	1.329	1.772	8.283
TOTAL								152.045	198.822	418.412
- Hybrid EML/Rocket Launcher										
Rails	(1/2)0.6(0.2)	879.6	8.96	7881.6	1.78	$14.029 M	2,2.5,3	28.058	35,073	42.087
Insulation	0.8(0.2)	3141.6	3-5	9424.7-15,703.0	1.00	$9.424-15.708 M	1.1,5,2	9.424	14.136	31.416
Force Confinement (Kevlar)	1.0(0.05-0.1)	644.0-1319.5	1.38	927.4-1900.0	28.20	$26.152-53.580 M	2.2,5,3	52.304	65.380	160.74
Vacuum Conf. (Al)	1.05-1.1(0.1-0.3)	132.6-420.3	2.7	358.0-1134.9	1.68	$0.602-1.906 M	1.5,2,3	0.9[illegible]	1.204	5.718
TOTAL								90.689	115.793	239.961

3 to 5 g/cc. For the coaxial accelerators, the synthetic rubber or plastic is expected to cost less than $2.20 per kg and have a density of 1.2 g/cc. Labor factors of 1 (materials cost includes labor), 1.5, and 2 are used to estimate the low, expected, and high estimates of tube insulator costs.

6.1.2.2.3 Radial Force Containment. To hold the launcher tube in place against the pressures developed during launching, the tube would have to be wrapped for support. Kevlar fiber wrapping is believed to be the best material available having the required strength at a reasonable cost. The Kevlar thickness required is currently estimated to be between 5 and 10 cm (6.5 cm for the coaxial hybrid EML/rocket launcher). Kevlar is made from two components, yarn and epoxy resin. The yarn is currently being sold in quantity at $26.40 to $44.00 per kg depending upon quality control. The epoxy resin is currently selling at $4.40 per kg. The degree of epoxy impregnation is a design variable, and a typical mix is 60 percent fibers/40 percent epoxy. This combination has a density of 1.38 g/cc as contrasted to Kevlar fibers with a density of 1.44 g/cc (Kevlar-49 Data Manual, 1976, 1982). The calculated cost per kg for a combination using aerospace-grade yarn (at $44/kg) is then $29.20/kg of composite. Direct costs of labor to fabricate are given by DuPont personnel as being approximately equal to material costs. Since the winding will have to be penetrated by conductors, it is most likely that a complex buildup pattern would be selected and a machine would be used to make the winding. For this type of operation, a direct labor factor of 2 times the material cost is used with the low material cost estimate from the low force containment estimates. For the expected and high cost estimates, labor factors of 2.5 and 3 are used with the high materials cost estimates.

6.1.2.2.4 Vacuum Containment. An exterior container would be required to provide a vacuum seal and a mechanical connection for the segments. This is assumed to be 1 cm thick on the coaxial launchers and one or three cm thick on the railgun launchers. An inexpensive plastic coating would provide electrical protection for normal operations. Aluminum is assumed to be used, but steel could be used if additional strength is desired. Aluminum in simple forms costs $1.68 per kg and has a density of 2.7 g/cc. The cost of the insulative coating is contained in labor factors of 1.5, 2, and 3, used to form the low, expected and high cost estimates for the segment containers.

6.1.2.3 Electrical Power Storage

For the coaxial launchers, a single liquid-nitrogen-cooled Brooks coil is envisioned to store electrical energy needed during launch. For the railgun launchers, many homopolar generators (HPGs) with liquid-nitrogencooled inductors are selected. The coaxial launchers use relatively high voltages and relatively low current to deliver power to the projectile while the railguns use high current at relatively low voltages to deliver their power to the projectiles. Additional equipment including large transformers would be required to use the Brooks coil

with a railgun; other design changes would be required to use HPGs with a coaxial accelerator. Accordingly, potential applications of mixtures of these technologies are not considered. Technologically advanced concepts such as superconducting storage ring under investigation by the University of Wisconsin are not considered because the costs are highly uncertain. The costs for the Brooks coils and homopolar generators are derived in Table 6-5 using information provided by P. Mongeau (Brooks coil) and R. Marshall (HPGs). The electrical energy storage for all designs is increased over the kinetic energy requirement of the projectiles to cover energy losses and impedance mismatches.

<u>6.1.2.3.1 Brooks Coils</u>. A Brooks coil has a "life-saver" shape (see Figure 4-13) and uses a single strand of conductor; aluminum was selected over copper because its lower density (2.7 vs 8.9 g/cc) would result in a lower cost for coils of the same size. Cooling the coil with liquid nitrogen would result in a decrease in resistance (by factor of about 10) and greatly increase electrical energy storage efficiency. The major design problems for the cooled Brooks coil are considered to be stress confinement and insulating the coil loops from each other. The stress confinement requirement is assumed to be managed by a reinforced concrete structure emplaced at the site and backed by the rock of the site. The loop insulation requirement is assumed to be satisfied by insulating blocks which are not fastened to the conductor and have a thermal expansion coefficient which is very close to that for aluminum, such as Bakelite. Such large amounts of conductor are required to make the coils needed for the launchers that the labor to build the coil is relatively small in comparison to the cost of the metal. For this reason, labor factors of 1.5, 2, and 3 are used to provide the low, expected, and high cost estimates. These estimates include the confinement structure, internal LN_2 plumbing, and insulation, but do not include switching and control costs. For the Earth-orbital launcher, the cost of the Brooks coils is estimated to cost from $237.5 M to $475.1 M, with an expected cost of $316.7 M. For the hybrid EML, the cost estimates range from $93.7 M to $187.3 M, with $124.8 M expected. Only one Brooks coil is costed, and this could severely affect system availability if the coil were severely damaged.

<u>6.1.2.3.2 Homopolar Generators (HPGs)</u>. The homopolar generators (HPGs) are considered to be the largest source of uncertainty in the mechanical design of the railguns. While very capable machines have been built in laboratories, the HPG experience still represents a relatively immature technology. There is also little experience in their manufacture or their use in operational systems, and there is no experience to indicate a reasonable number of spares.

To avoid heat build-up in the launcher facility from conversion of electrical into mechanical energy in the HPGs, it is likely that some form of conversion external to the launcher area would be required. Hydraulic conversion is selected for the HPGs because the hydraulic fluid could carry away excess heat. Reconversion from mechanical to electrical energy would also generate heat; provision must be made to provide circulating fluids or air to preclude heat build-up.

TABLE 6-5. ENERGY STORAGE REQUIREMENTS

Projectile Energy	Mass, kg		Velocity (m/sec)		Energy (Joules)
Coaxial--Hybrid EML (1/2)	15,400	x	$(2000)^2$	=	3.0×10^{10}
--Earth-Orbital	3,250	x	$(7000)^2$	=	8.0×10^{10}
Railgun--Hybrid EML	15,200	x	$(2000)^2$	=	3.0×10^{10}
--Earth-Orbital	5,900	x	$(7000)^2$	=	1.5×10^{11}

Brooks Coil Size, Mass, Cost (Scaled from Kolm and Mongeau, 1983).

$$\text{Mass (Al, MT)} = \frac{E(J) \times 2230}{2.07 \times 10^9} \qquad \text{Diameter (m)} = \sqrt[3]{\frac{E(J) \times 10^3}{2.07 \times 10^9}}$$

Energy increased by 15 percent for impedance losses:

Coaxial Hybrid EML Coil Energy = 3.0×10^{10} J x 1.15 = 3.45×10^{10}
Coaxial Earth-Orbital Coil Energy = 8.0×10^{10} J x 1.15 = 9.15×10^{10}

Brooks Coil	Energy	Mass (MT)	Diameter (m)
Hybrid EML	3.45×10^{10}	37,166	25.5
Earth Orbital	9.15×10^{10}	98,646	35.3

Aluminum Cost, $1680/MT; Labor Factors: 1.5, 2, 3

Single Brooks Coil	Cost Estimates, $, M, 1981		
	Low	Expected	High
Hybrid EML	93.7	124.8	189.3
Earth-Orbital	248.5	331.4	497.2

56 Megajoule Homopolar Generators; Number and Cost

Energy increased by 38 percent for impedance losses in transfer from HPG to inductor (85 percent efficiency) and inductor to rails (85 percent efficiency):

Railgun--Hybrid EML HPG Energy 3.0×10^{10} x 1.38 = 4.15×10^{10} Joules
Railgun--Earth-Orbital HPG Energy = 1.5×10^{11} x 1.38 = 2.0×10^{11} Joules

--56 MJ per HPG--

Number of Hybrid EML HPGs = $4.15 \times 10^{10}/56 \times 10^6$ = 742
Number of Earth-Orbital HPGs = $2.0 \times 10^{11}/56 \times 10^6$ = 3573

TABLE 6-5. (Continued)

Adjust for spares:

Number of Hybrid EML HPGs = 800
Number of Earth-Orbital HPGs = 3800

Cost of HPGs:

For production runs of 10,000, $1,000 to $1,500 per MJ (R. Marshall, UT) or $56,000 to $84,000 per HPG with an expected cost of $70,000 per HPG.

For production runs of about 1000, multiply by 3.
For production runs of about 4000, multiply by 2

Homopolar Generators (Number)	Cost Estimates, $, M, 1981 Low	Expected	High
Hybrid EML (800)	94.0	168.0	201.6
Earth-Orbital EML (3800)	425.6	532.0	638.4

The preliminary estimates of the size of a 56-MJ homopolar generator believed appropriate for these designs are 1.8 m in diameter and 1.5 m long, with a mass of about 10 MT. The size of these devices is such that they could be placed on one side of the hybrid EML/rocket launcher tube, but would have to be arranged in a crescent about the 7 km/sec Earth-to-orbit launcher tube.

Information on existing HPGs and possible production runs on the order of 10,000 indicates that HPGs should cost between $56,000 and $84,000 each, with an expected cost of $70,000. For a device weighing 10 MT, the expected cost is $7.00 per kg, about that for an automobile. Since the production runs for the Earth-orbital launcher and the Hybrid EML/rocket launcher concepts will be about 3600 and 800, respectively, the effects of learning on production costs are not expected to be as great as for a production run of about 10,000. Accordingly, the costs for the hybrid EML railgun rocket launcher concept which would require 742 installed HPGs (750 for redundancy) are increased by a factor of 3. The costs for the Earth-orbital launcher concept, which would require 3585 installed HPGs (3600 for redundancy), are increased by a factor of 2. Since an appropriate level of spare HPGs cannot be forecast at the present time, a level of about 5 percent is used. Thus 800 HPGs for the hybrid EML/rocket launcher concept would cost between $94.0 M and $201.6 M with an expected cost of $168.0 M. The 3800 HPGs for the Earth-orbital launcher would cost between $425.6 M and $638.4 M, with an expected cost of $532.0 M.

6.1.2.4 Switching and EML Inductors

Reusable switches capable of handling the power and switching rates contemplated for both coaxial and railgun launchers represent a major area of technological uncertainty. The costs for the switches (and their development) accordingly are very uncertain.

Storage inductors would be required for the railgun systems, because the HPGs would not be able to convert their mechanical energy into electrical energy quickly enough to feed the launcher efficiently. Since the coaxial launchers use a Brooks coil to store electrical energy, it is possible that secondary storage inductors may not be needed along the launcher; however, the self-inductance of the long conductors needed to transmit the current from the Brooks coil to the launcher tube coils and the short action times of these coils do not permit this assumption. Accordingly, secondary inductors in equal numbers and ratings, are costed for both coaxial and railgun launches.

For the railgun launchers, preliminary calculations by R. Marshall (Rice, et al, 1982) indicate the secondary inductors must store approximately 48 MJ of energy at a current of 4 MA to achieve the assumed efficiency of 85 percent. To prevent resistive energy losses, the inductor must also have a resistance of less than 2.7×10^{-6} ohms. For inductors of coaxial or toroidal configurations, mass is sensitive to the number

of turns and the conductivity of the material. Since normal conducting metals drop in resistance by approximately a factor of 10 when their temperature drops from room temperature to that of liquid nitrogen (LN_2), it is presently considered desirable to use LN_2-cooled inductors. This results in a calculated significant reduction in inductor mass. Marshall's preliminary calculations indicate that a four-turn inductor of this size would have a diameter of 1.5 m and a length of 1.8 m. The inductor can also reasonably be expected to contain the LN_2 used to cool it. Foamed insulation currently has problems with cracking and separation upon repeated cryogenic cycles; research is being conducted in this area for application to reusable space vehicles. Thus, it is reasonable to expect that foamed insulation would be appropriate at the time of implementation. Contained foam (preformed) insulation would always be available as a back-up technology. Accordingly, the inductors are costed with a labor factor of 10 times the raw material price to reflect the uncertainty of the switching technology. The current price for aluminum ingots is \$1.68/kg (AMM, 1984). The requirement for low conductivity translates into a requirement for controlled purity and thus may bring the price up to \$2.00/kg. Thus, the materials cost for an inductor of 1 to 1.5 tons is \$2000 to \$3000. Other materials and labor, at a factor of 10 times the primary materials price, raise the cost per inductor to \$20,000 to \$30,000 for each unit. The low, expected, and high estimates for the inductor subsystem are then formed by unit prices of \$20,000, \$25,000, and \$30,000, and the same level of spares, as for the HPGs.

Switching for the coaxial accelerators is expected to have a slightly different set of requirements in that individual loops or groups of loops in the launcher coil must be controlled to produce several current pulses in very rapid succession. Because a detailed design is not available, a cost analysis is not possible at this time. It is expected that the costs of distributing and switching coaxial launcher tube coils would be about the same as for the railgun accelerators. The cost estimates, accordingly, are the same as for the railgun accelerators.

For the hybrid railgun EML, 800 inductors/switches (including spares) would be required with unit prices in the range of \$20,000 to \$30,000; this leads to an estimate of \$16.0 M to \$24.0 M, with an expected value of \$20.0 M. For the Earth-orbital railgun, 3800 inductors/switches would be required; these are estimated to cost in the range of 76.0 M to \$114.0 M with an expected cost of \$95.0 M. These costs are also used for the equivalent coaxial accelerators.

6.1.2.5 Projectile Injection Systems

A preboost system is believed to be desirable for optimum launcher operation and would be required for plasma-armature railguns to reduce rail erosion at low velocities. A design for the preboost systems has not been developed, only preliminary concepts are suggested. These all use gas to overcome the standing inertia of the projectile. In all cases, marginal operating costs are expected to be low (\$50-\$200 per launch). Except for the Earth-orbital EML, the capital costs are also relatively small.

For both hybrid EML (coaxial and railgun) concepts, stored compressed air is believed to be sufficient to drive the projectile along a 100 to 150 m tube with sufficiently high acceleration that initial launcher acceleration would be smooth. If the acceleration achieved by this system is as low as 1 g, the velocity upon entering the launcher would still be about 50 m/s. The cost of a compressed air system to include compressors, storage tanks, piping and valves, is believed to cost in the range of \$1 M to \$3 M with an expected cost of \$2 M. Because the coaxial Earth-orbital EML does not have arc-erosion problems, the compressed gas preboost system is also considered appropriate, and is charged to this launcher. The Earth-to-orbit railgun launcher, however, is required to have higher preboost velocities to operate properly. For this reason a hydrogen/oxygen driven piston system is proposed. This pre-accelerator, also about 100 to 150 m long, would be used to accelerate the projectiles. A piston would drive a mixture of nitrogen and hydrogen gas which would in turn accelerate the projectile. For the railgun the sabot necessary to distribute electromagnetic launch stresses would also suffice to prevent leakage. The cost of the propellants, given the availability of electrical power and a gas liquification plant, will be in the range of \$100-\$200 per launch. The cost to design, manufacture, and install the launching barrel segments together with the breech section are expected to far outweigh the cost of the steel used. It is estimated that the design, manufacture and installation of the barrel could be accomplished for \$80 to \$120 M with an expected cost of \$100 M.

The high cost of the conceptually simple system is due to the need to inject a large quantity of gas at high pressure in a short time (0.4 sec). High-pressure rocket engine pumps would be required to inject the liquids. These would have an operating time of only a few seconds per shot, so that service life should be very long (10 to 20 years), based on modest extrapolation from current Space Shuttle experience. While rocket engine technology would be used, there are incentives to permit large increases in mass of components and housings to provide safety. This is one area where growth in allowable mass could reduce costs. While an SSME currently costs on the order of \$20 to \$30 M, many components, such as nozzles, engine mounts, etc., would not be needed. Thus, it is reasonable to expect that hardware components adapted to this task, including spares, could be purchased at about half the cost for one SSME at the present time. The design effort, however, would be significant and accounts for most of the costs estimated.

6.1.2.6 Power Conversion Plant (Railgun Launchers)

Because motoring the homopolar generators would significantly reduce brush life, and to reduce heat build up in the launcher facility, railgun HPGs are envisioned to use hydraulic power from a conversion facility near the launcher. It is expected that hydraulic motors would both save space in the launcher facility and be somewhat lower in cost than electric motors. While it would be possible to transmit power in the form of steam from the power plant, the transmission flexibility

of electrical power suggests that it would be better to accept the conversion inefficiencies of reconverting the electrical power back into mechanical power at a station near the launcher. Such conversion would also permit an ambient-temperature hydraulic fluid to carry away the heat from driving the HPG rotor. This form of thermal control is considered necessary for the Earth-orbital railgun and desirable for the hybrid railgun/rocket system, even though the presence of the hydraulic fluid (assumed to be water-based) may increase the risk of an electrical accident.

Since electric to high-pressure hydraulic conversions of this size do not appear to have been undertaken previously, no good analogy is available to draw upon. The conversion power cost was established as costing one-fourth that of the power plant on a per kilowatt of capacity or $550 to $700 per kWh, with $625 expected. Conversion capacity of 15,000 kWh would be required for the hybrid EML, with costs in the range of $8.3 M to $10.5 M ($9.4 M expected). For the Earth-orbital launcher with a launch velocity of 7 km/s, 50,000 kWh of capacity would be required with a cost range of $27.5 M to $35.0 M and an expected cost of $31.3 M. The conversion power level selected would permit recharging the homopolar generators in one hour, rather than the one and one-half hours expected between launches.

6.1.2.7 Water Distilling Plant

The launch site cannot be assumed to have sufficient fresh water either to support the launcher operations (power plant, hydraulic conversion, LN_2 plant cooling, etc.) or the operating personnel and their families. While much of the water used in launcher operations would be recycled, the same cannot be said for water for human consumption and household use. Accordingly, a water distilling plant may be needed. The plant is sized at 1,000,000 liters per day, representing 400 liters per person per day for 2500 people. This is expected to have reserve capacity for the crew as well as families and transients. The distillation plant would use the heat rejected from the nuclear plants in their bottom cycle and would therefore represent a predominantly capital cost. This type of facility is expected to cost about $2.5 per liter-day of capacity, or $2.5 M. Because of the effective integration of this system into the power plant, the uncertainty in cost is very high--it may cost much less or somewhat more depending upon the specific designs selected. Solar evaporation and condensation is also available in this price range, but would have higher operating costs. Solar evaporation would be used if the site does not require a nuclear plant.

6.1.2.8 Gas Handling Facilities

Three types of gas handling facilities are expected: (1) a liquid nitrogen plant and storage area; (2) an evacuation system for the launcher tubes; and (3) hydrogen and oxygen storage for the preboost system needed for the Earth-orbital railgun. These are discussed in the following three sections.

6.1.2.8.1 Liquid Nitrogen Plant and Storage. To provide acceptable inductor masses, the resistance of their conductive material must be dropped by approximately an order of magnitude from that available at room temperatures. Liquid nitrogen cooling of the inductors was selected over superconduction because the state of superconducting technology does not permit reasonable cost estimates in the foreseeable future. The major uncertainities in selecting LN_2 cooling are the requirements of LN_2 due to insulative losses in the inductors and their plumbing and to the efficiency of transmitting electrical power through the inductors to the launcher. These would involve both thermal and electrical losses, placing a heat-sink requirement on the LN_2 and requiring insulated plumbing/ductworks. The major uncertainty, however, is believed to be the insulation requirements and the costs needed to meet them. Based on the heat of vaporization for LN_2, 47.6 kcal/kg (0.05534 kWh/kg), and an assumed 15 percent of input energy as a combined thermal and electrical inefficiency causing LN_2 boil-off, the requirements for LN_2 are calculated for the maximum launch rate of 16 launches per day. From information provided by J. Cost, Air Products Company, a plant providing 325 MT of LN_2 per day would cost \$4 M and would scale upward by a 0.6 power law on cost. Mr. Cost believes the 0.6 power law is slightly optimistic, so a 0.7 power law is used to calculate the expected costs. The costs of multiple units of 325 MT per day of LN_2 capacity are used to calculate the high cost estimate. A summary of the calculations is presented in Table 6-6, where the dissipated energy is 15 percent of 115 percent of the $1/2mv^2$ energy, or 17.5 percent of the required projectile energy for each launcher. The cost of the LN_2 plant is small in relation to other costs, as shown in Table 6-6.

6.1.2.8.2 Evacuation System for Launcher Tube. The evacuation of the small-diameter launcher tubes to approximately 1/100 (7.6 mm Hg) atmosphere would require the removal of less than 1300 kg of air. For the larger tubes, removal of up to 3000 kg would be required. This could be accomplished with rotating impeller pumps, able to achieve high volume throughput. The removal of 99 percent of the air would leave 13 to 30 kg of air in the bore. At least three pumps are estimated to be required. Each pump would be able to handle the evacuation unassisted. The installation is estimated at \$1 to \$1.5 M for ductwork, shelters, pumps, and motors.

6.1.2.8.3 Water Electrolysis Plant. To provide hydrogen and oxygen for the Earth-orbital railgun's preboost system, a water electrolysis plant would be needed. Since hydrogen has much lower viscosity than air, it has been used in large electrical generators to reduce the atmospheric friction between rotors and stators. This hydrogen has usually been produced by electrolysis of water with the electricity produced by the generators. Accordingly, the cost of the electrolysis plant is contained within the estimate for the power plant. The facilities to liquify the gases are provided within the estimate for the liquid nitrogen plant. In addition to these elements, there would also be additional costs for storage and handling facilities. These are estimated at \$0.2 M, \$0.3 M, and \$0.4 M for the low, expected, and high costs of these facilities.

TABLE 6-6. LIQUID NITROGEN REQUIREMENTS

Determining Assumption: Energy losses resulting in LN_2 boil off are 17.5 percent of projectile energy.

Launcher	Projectile Energy, Joules	LN_2 Energy 16 Shots/Day @ 17.5 Percent, Joules
Coaxial--Hybrid EML	3.0×10^{10}	8.4×10^{10}
--Earth Orbital	8.0×10^{10}	2.2×10^{11}
Railgun--Hybrid EML	3.0×10^{10}	8.4×10^{10}
--Earth Orbital	1.5×10^{11}	4.2×10^{11}

LN_2 heat of vaporization = 47.6 kcal/kg = 199,254 Joules/kg

Plant Capacity	
Launcher	LN_2 Plant Output/Day, MT
Hybrid EML	421.5
Coaxial--Earth Orbital	1104.1
Railgun--Earth Orbital	2107.9

$$\text{Plant Cost Scaling: } C\ (\$,\ M,\ 1981) = (\$4\ M)\left(\frac{\text{Capacity}}{325\ \text{MT}}\right)^{x}$$

$x = 0.6, 0.7, 1.0$

Multiply by 1.5 for storage, plumbing, etc.

Launcher	Plant Cost, $, M, 1981 Low	Expected	High
Hybrid EML	7.0	7.2	7.8
Coaxial--EO	12.6	14.3	20.8
Railgun--EO	18.5	22.2	39.0

6.1.2.9 Handling Devices/Systems

Handling devices, principally cranes and small railcars, would be needed to manipulate the projectiles and tube segments, as well as homopolar generators and/or inductors. The handling devices are envisioned to travel along suspended beams attached to the floor, roof, or walls of the launcher tube housing. Light elevators can cost as low as $500/m and heavy freight elevators can cost $1250/m (personal conversation, M. Minelt, Otis Elevator Co., March, 1982). It is expected that there would be multiple cars or cranes per set of tracks and that they would be of very heavy duty construction. Accordingly, unit costs for the handling systems are believed to fall in the range from $1000/m to $5000/m with an expected value of $3000/m. For a launcher with a nominal length of 2000 m and a loading-preboost length of 100 to 200 m, the estimated costs for the handling devices and associated equipment are estimated to cost in range of $2.2 M to $11 M with an expected value of $6.6 M.

6.1.2.10 Control Center, Controls and Monitoring Systems

A preliminary system design, as well as specification of the control requirements, is needed before accurate estimates of the control costs can be made. It is assumed that inductors and coils or homopolar generators and inductors could be monitored and their switches controlled from a master control center for a relatively low cost per unit. A tentative estimate of $1000 to $10,000 per HPG/inductor set is used for the railgun launchers, and $5000 is used as the expected value. Because the switching and control systems are not well described for the coaxial launchers, the equivalent costs are assumed for the coaxial EMLs as for the railguns. An additional $5.0 M is added for the central control system and other unidentified costs. For the hybrid EML/rocket launcher concepts, the costs are estimated to range from $6.0 M to $15 M for the 800 HPG/inductor sets (or equivalent inductors/switches for the coaxial launcher), with an expected value of $10 M. For the Earth-orbital EML, concepts the costs range from $8.0 M to $39 M ($21 M expected) for 3800 HPG/inductor sets or their equivalents for the coaxial launcher.

6.1.2.11 Tracking System

A tracking system would be required to monitor the trajectory of launched projectiles. Since these launchers are not intended to handle high value or hazardous payloads, the necessity to have accurate knowledge of the trajectory under abnormal conditions can be relaxed. For this reason, adoption of a military tactical radar station is selected to cover the near-launch-site trajectory, with long-range monitoring and control in space conducted elsewhere. Accordingly, low, expected, and high values for the radar station are established as $5 M, $10 M, and $15 M.

6.1.2.12 Accident Recovery Systems

Because hazardous materials are not anticipated to be transported by the launchers, no formal accident recovery system is planned, and none is costed. It is expected that misfired projectiles which could be recovered easily would be, but this would be a very minor part of operations activity.

6.2 EML Operations Cost Estimates

The costs to operate a launcher facility have two components: the recurring costs associated with each projectile, and the annual costs for personnel and supplies; projectile costs are summarized in Table 6-7; personnel and supply costs are summarized in Table 6-8.

While the projectile is part of the launch system, payloads are usually considered to be part of some other mission or activity and are not costed here. The cost of the bulk payloads considered for the supply missions would be very low, in any case. The costs for their development are addressed in the cost estimates for an Operational Test Program.

The annual cost estimates cover operation of the facility for use as a launch site only. While provisions are made for people and consumables to load the bulk payloads, this would be a simple procedure. Only tasks such as loading of liquid propellants and other fluids, initiation of guidance systems, and verification of status would normally be undertaken at the facility. Use of the facility for research or other programs would involve additional costs.

6.2.1 Projectiles and Mission Peculiar Equipment

The costs for Earth-orbital projectiles are highly uncertain due to technological advances needed to achieve and demonstrate the capabilities required. In addition, the annual quantities required (hundreds or thousands) may not be large enough to assure that major savings through mass production (in the manner of automobiles) can be achieved. The ability to achieve the low estimates depends upon keeping labor costs low.

For the hybrid EML/rocket launcher concept, the projectiles would be derivatives of existing solid-propellant stages with relatively low technology risks. Those stages must, however, be produced at rates much higher than achieved to date. The next three major sections discuss the costs for the four projectiles and their mission peculiar equipment. Costs are summarized in Table 6-7.

6.2.1.1 Hybrid EML/Rocket Projectile (Coaxial and Railgun)

6.2.1.1.1 Payload. Payload costs are not considered in our analyses. Most payloads are expected to be bulk materials and should not be affected by the accelerations of the electromagnetic booster.

TABLE 6-7. COST ESTIMATES FOR EML PROJECTILES ($, K, 1981)

WBS Element	Coaxial Hybrid EML Projectile			Railgun Hybrid EML Projectile			Coaxial Earth Orbital Projectile			Railgun Earth Orbital Projectile		
	Low	Expected	High	Low	Expected	High	Low	Expected	High	Low	Expected	High
1. Payload	--	--	--	--	--	--	--	--	--	--	--	--
2. Shroud or Nose Cone	0.5	1.0	3.0	0.5	1.0	3.0	0.2	1.0	5.5	0.8	1.3	2.1
3. Structure	(a)	(a)	(a)	(a)	(a)	(a)	3.0	6.6	14.0	3.0	7.0	11.0
4. Thermal Protection	(b)	(b)	(b)	(b)	(b)	(b)	1.0	2.0	3.0	1.5	3.3	7.0
5. Fins	0.1	0.3	0.5	0.1	0.3	0.5	1.3	2.0	3.3	1.3	2.0	3.3
6. Sabots or Coils	1.0	1.4	1.7	1.0	1.4	1.7	1.2	1.8	2.4	0.9	1.0	1.2
7. Handling, Equipment, and Supplies (c)	1.0	2.0	5.0	1.0	2.0	5.0	1.0	2.0	5.0	1.0	2.0	5.0
8. Instrument Package	1.0	2.0	3.0	1.0	2.0	3.0	1.0	2.0	3.0	1.0	2.0	3.0
9. Propulsion System and Propellants	78.0	117.0	156.0	90.0	135.0	180.0	25.0	100.0	200.0	25.0	100.0	200.0
10. Transportation	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5
TOTALS	82.1	124.2	169.7	94.1	142.2	193.7	34.2	117.9	236.7	35.0	119.1	233.1

(a) Costed as the case for propulsion system.
(b) Not needed.
(c) Includes costs of preboost system.

TABLE 6-8. PERSONNEL AND SUPPLY COST ESTIMATES (PEOPLE AND $, M, 1981)

WBS Category	Earth Orbital Launcher						Hybrid EML/Rocket Launcher					
	Coaxial			Railgun			Coaxial			Railgun		
	Low	Expected	High	Low	Expected	High	Low	Expected	High	Low	Expected	High
4.1 Management and Support												
4.1.1 Management	15	20	40	15	20	40	15	20	40	15	20	40
4.1.2 Engineering	30	50	100	30	50	100	30	50	100	30	50	100
4.1.3 Facility Support	30	50	75	30	50	75	30	50	75	30	50	75
4.2 Power Plant Operations												
4.2.1 Crew	25	50	75	25	50	75	10	20	30	10	20	30
4.2.2 Supplies ($, M)	2.0	3.0	5.0	2.0	3.0	5.0	1.0	2.0	3.0	1.0	2.0	3.0
4.3 Technical Personnel and Supplies												
4.3.1 Control Center Crew	20	40	60	20	40	60	20	40	60	20	40	60
4.3.2 Launcher Equipment Crew	50	75	100	50	75	100	50	75	100	50	75	100
4.3.3 Equipment Refurbishment Crew	50	75	100	75	100	125	50	75	100	75	100	125
4.3.4 Power Conversion Facility Crew	--	--	--	10	15	20	--	--	--	10	15	20
4.3.5 LN_2 Plant/Evacuation System Crew	20	30	40	20	30	40	20	30	40	20	30	40
4.3.6 Projectile Operations Crew	30	40	50	30	40	50	30	40	50	30	40	50
4.3.7 Facility Utilities Crew	50	100	200	50	100	200	50	100	200	50	100	200
4.3.8 Technical Supplies	3.0	5.0	8.0	3.0	5.0	8.0	3.0	5.0	8.0	3.0	5.0	8.0
Personnel Total	320	530	840	380	570	885	305	500	795	340	540	840
Annual Personnel Cost at $50 K per year ($, M)	16.0	26.5	42.0	19.0	28.5	44.3	15.3	25.0	39.8	17.0	27.0	42.0
Total Annual Cost	21.0	34.5	55.0	24.0	36.5	57.3	19.3	32.0	50.8	21.0	34.0	64.0

6.2.1.1.2 Nose Cone or Payload Shroud. For the hybrid EML concepts, the nose cone or payload shroud would be an adaption of existing designs. This shroud is intended for a 2-3 km/sec aerodynamic regime which is similar to those currently experienced by expendable launch vehicles. Production rates of the adapted design are expected to reduce the costs significantly to yield a range of $500 to $3000, with an expected value of $1000.

6.2.1.1.3 Structure. The primary structure for the hybrid EML projectile would consist of the solid rocket motor cases and their interstages or connectors. The provision of the inductive rings on or near the surface of the coaxial projectile is expected to have only a small effect on the design costs for the coaxial projectile, and result in no appreciable increase in the manufacturing cost over a case without provision for rings. The cost of the rings is expected to be approximately equal to the cost of the sabot needed for the railgun projectile and is discussed in Section 6.2.1.1.6. The railgun projectile would require a stronger case, especially in the rear part of the projectile where provision must be made for thrust transfer from the sabot. However, because both cases must be designed to withstand 100-g accelerations, the cost differential between the railgun coaxial projectiles due to structure strengthening is expected to be about 15 percent. Production rate effects associated with the high volume of projectiles are expected to keep the costs of the structures low in relation to other costs. Because the projectile case is an integral part of the solid rocket motors, specific cost estimates are made in the propulsion section (6.2.1.1.8).

6.2.1.1.4 Thermal Protection System (Side Body). For the hybrid EML concepts, the projectiles would not require thermal protection.

6.2.1.1.5 Fins. These projectiles would require stabilization (not steering) fins. Their operating regime would not be severe and their cost is expected to be in the range of $100 to $500 with $300 being the expected cost.

6.2.1.1.6 Sabot or Rings. Because of relatively low launch stresses, the sabots for the railgun hybrid EML projectile would not require exotic materials, but rather an inexpensive non-conductor to protect the projectile together with a conductive plate, probably made with copper or aluminum alloy. The sabot cost should be little more than the price of simple constructs from a manufacturer in the range from $1.30 to $2.00 per kg; labor factors of 1.1 to 1.3 are appropriate. The coils or rings used on the coaxial accelerator projectile are expected to be made of a copper alloy and have no unusual requirements. For both the coaxial and railgun launchers, cost estimates are in the range of $1000 to $1700, with the expected value of $1400 based on a mass of 650 kg for the coaxial projectile rings.

6.2.1.1.7 Auxiliary Handling Equipment and Supplies. The projectile's auxiliary propulsion system and guidance, navigation and

control system would require final checkout and preparations at the launch site. This checkout procedure is expected to require a few man days. Low, expected, and high cost estimates are $1000, $2000 and $5000 for time and materials.

<u>6.2.1.1.8 Instrument Package</u>. The instrument packages for all projectiles would be designed for high production and low unit costs. It is assumed that assemblies with moving parts (gyros, etc.) would be avoided or designed to be cost effective, and complex functions would be handled by software rather than hardware. To keep costs down, multi-year procurements of components/assemblies would be used, even if final assembly is regulated by demand. Based on these conditions and annual demand for thousands of instrument packages, it is likely that the cost of the instrument package (20 kg for hybrid EML projectiles) would lie in the range of $1000 to $3000. The expected value is selected as the mid-range or $2000.

<u>6.2.1.1.9 Propulsion System</u>. The projectiles for the hybrid EML concepts would consist of three stages with total masses (including payload) of 15,400 kg for the coaxial projectile and 15,200 kg for the railgun projectile. In contrast, the current Scout expendable launch vehicles has a mass of about 21,500 kg (excluding payload) and the three Scout upper stages have a total mass of 6687 kg, or about 44 percent of the proposed projectiles.

For the latest production run of 15 Scout vehicles, production costs were about $4 M per vehicle and the total launch cost is in the range of $8 M to $10 M at a launch rate of about three per year (EDD, 1976). In the late 1960s, Scout Launch vehicles were being manufactured for $1 M each, with launch services on a basis equivalent to current charges being about another $1 M. At that time, the launch rate was about 10 per year. Thus, in addition to inflation, the launch rate decrease has resulted in a significant cost increase because overhead costs cannot be spread across many launches.

The projectiles must be mass-produced at much lower costs. Because the stages would essentially be cannisters filled with relatively simple and inexpensive chemicals, it is expected that they could be produced in quantity at about the same cost per kilogram as an automobile or $6.00 per kilogram. Material costs for propellants (or $3.00 per kg), however, are expected to keep the cost from going much below this level. The structure of the projectile is expected to be formed from fiber matrix composites (e.g., Kevlar) which currently costs about $40 per kg, but contribute only 5 to 10 percent of the projectile's mass. It is also reasonable to expect the cost of these materials to drop in the future. Accordingly, the cost per kilogram f· the propulsion system (which includes the structure) is estimated to be in the range from $6 to $12 with $9 expected. This leads to estimates of the coaxial projectile from $78,000 to $156,000, with an expected value of $117,000 for propellant and structural casing mass of about 13,000 kg. This estimate includes a small auxiliary or attitude control propulsion system. For the railgun

projectile, the estimates are increased by 15 percent to consider the stronger motor cases and other structural components. This yields an estimated range of $90,000 to $180,000 and an expected estimate of $135,000.

6.2.1.1.10 Transportation. Transportation charges for all projectiles are established at $500, reflecting surface transportation in economical lots. This is consistant with the projectile's major use as a method of bulk transport.

6.2.1.2 Earth Orbital Projectile (Coaxial Launcher)

6.2.1.2.1 Payload. Payload costs are not considered in our analyses. Most payloads are expected to be relatively inexpensive bulk materials which would not be affected by the high accelerations of the launcher.

6.2.1.2.2 Nose Cone. The nose cones for the Earth-orbital launchers would undergo a much more severe thermal environment than that for the hybrid EML projectile, and an ablative nose cone was assumed. Considerable technical verification is needed to develop experience and confidence in metallic ablators, but inexpensive metals such as steel probably can be used and their costs can be quite low. Thus, materials cost per kg can range from less than $1.00 for modestly priced alloy steels to $10 for some specialty stainless steels. Labor factors for high production rates are expected to be in the range from 2 to 5. Thus for a nose cone mass of 110 kg, the costs can range from $220 to $5500. Accordingly, a low estimate of $220 reflects use of modestly priced steel and a labor factor of 2, an expected estimate of $1000 reflects use of higher priced steel and a labor factor of 5, while the high estimate of $5500 assumes use of expensive steel ($10 per kg) and a labor factor of 5.

6.2.1.2.3 Structure. An inexpensive steel structure with a mass of 300 kg would be needed to provide strength. The materials cost for this steel would be in the range of $0.55/kg to $1.00/kg. Low labor factors (in the range of 2 to 5) would yield estimates in the range from $165 to $1500. The low end of this range is not considered feasible, so the estimates are increased to $1000 to $3000 with $2000 being the expected value.

6.2.1.2.4 Thermal Protection System (Sidebody). A thermal protection system would be required and is envisioned to be made of carbon-carbon (C-C) material (or other advanced composites) and have a mass of 200 kg. At the present time C-C materials are expensive to fabricate because they are labor intensive in all manufacturing stages and in quality control. Given historical progress in materials development, it is expected that C-C materials can be produced in the time frame of the launchers' at an installed price equivalent to that of tungsten metal or $33 per kg. A likely range in this estimate is from $15 to $70 per kg. Thus for 200 kg of sidebody structure and thermal

protection, the costs would range from $3000 to $14,000 with an expected value of $6,600.

6.2.1.2.5 Fins. The use of fins for aerodynamic stability would be required, and their mass is estimated at 20 kg. At this low mass, exotic materials such as carbon-carbon composites or tungsten do not produce high costs and are associated with materials costs in the $30 to $40 per kg range and with labor factors of 2 to 5. The expected labor factor is 3. This yields cost estimates in the range from $1300 to $3300 per projectile, with an expected estimate of $2000.

6.2.1.2.6 Coils. The inductive coils for the projectile would have a mass of 300 kg and would be made from copper alloy. Materials costs are less than $2.00 per kg and labor factors of 2, 3, and 4 yields estimates in the range of $1200 to $2400 with $1800 the expected value.

6.2.1.2.7 Checkout Equipment and Supplies. The launcher site checkout is expected to require a few man days, and is estimated to cost from $1000 to $5000, with an expected cost of $2000 (see Section 6.2.1.1.7).

6.2.1.2.8 Instrument Package. The instrument packages for all projectiles are expected to cost about the same. Through design for high production rates, this cost is expected to be in the range from $1000 to $3000, with an expected cost of $2000 for a 20-kg package.

6.2.1.2.9 Propulsion System. From the definition of the projectile, the dry mass of the propulsion system would be 425 kg, and the mass of the propellants would be to achieve a 500-km circular orbit would be 1150 kg. The available cost information on propellants exhibits wide range--from $0.18/kg for chlorine to $30/kg for propellant-grade hydrazines. For the currently expensive propellants, new production facilities would be required. With economical production facilities, the propellant costs are expected to be about $6/kg; for the 1150 kg propellants in this projectile, the propellants would cost $6900.

Because the projectile for the coaxial accelerator would have a light support structure/thermal protection system it is expected to require tanks to contain the fuel (N_2H_4) and oxidizer (ClF_3). It is conceptually possible that the support structure could serve as tankage, but safety considerations will probably not permit such a single-point failure mode. The tankage requirements, however, are not expected to be a major cost problem on the basis that tanks will be produced at rates well into the thousands per year.

To achieve a reasonably attractive cost for propulsion units, it is necessary to hypothesize advances in several production technologies (e.g., computerized machining, welding, and testing), in addition to high production rates. At the present time, the major cost problems are believed to be in the area of engine mechanical control rather than in the engine or tankage. It is in these areas that improvements in

production and testing technologies are believed to have the biggest payoff in reduced costs. High production rates for a 425-kg (dry weight) propulsion system and 75-kg ACS/astrionics package are believed to be able to reduce unit costs to the level of an automobile, or about $10,000, but this is considered optimistic, especially for the early part of the program. The low cost is therefore established at $25,000 with an expected cost of $100,000, and a high cost estimate of $200,000, all of which include the cost of propellants. The estimate of $100,000 is about one-tenth that of a comparably sized low-g unit produced today at a production rate of 5 to 10 units per year.

6.2.1.2.10 Transportation. Transportation charges for all projectioes are established at $500. See Section 6.2.1.1.10 for additional information.

6.2.1.3 Earth-Orbital Projectile (Railgun Launcher)

6.2.1.3.1 Payload. Payload costs are not considered in this analysis. Most payloads are expected to be bulk materials with relatively low unit costs, and resistant to damage from launch forces.

6.2.1.3.2 Nose Cone. The nose cone for the Earth-orbital launcher is expected to undergo a more severe thermal environment than that of the hybrid EML. An ablative nose cone is considered to be required. The discussion of Section 6.2.1.2.2 indicates that the cost of materials can have a wide range.

Inexpensive or moderately-priced steel is expected to be appropriate for this application. The least expensive steel costs about $0.55/kg and some heat resistant alloys cost about $1.00/kg. Moderately-priced stainless steels cost about $3.00/kg. Some very high temperature steel alloys can cost in the range of $10/kg. Appropriate labor factors for high production rates are in the range from 2 to 5. Thus, for a nose cone mass of 420 kg, the cost per projectile can potentially range from $420 to $42,000. The expected cost estimate is $1260 reflecting steel at $1.00/kg and a labor factor of 3. The range selected for this assessment is $840 to $2100 reflecting low and moderately-priced steels and labor factors of 2 and 5.

6.2.1.3.3 Structure. The structure of the railgun projectile would be required to sustain substantial launch stresses and must be thermally protected against atmospheric heating. The support structure is envisioned to be made of inexpensive steel with a mass of 2730 kg, and the thermal protection system (Section 6.2.1.3.4) would be made of relatively expensive composites with a mass of 100 kg. The structural steel would have a cost of $0.55 to $1.00 per kg, and labor factors are expected to be in the range of 2.0 to 4.0. Accordingly, the estimated cost for the steel structure is from $3000 to $11,000 with an expected cost at the midrange of $7000.

6.2.1.3.4 Thermal Protection System. The TPS material would be made of an advanced composite which is expected to have an installed

cost in the future about that of tungsten metal or \$33/kg. The range on this cost is expected to be plus or minus a factor of two or about \$15 to \$70. Accordingly, the TPS for the railgun projectile is estimated to have a cost range of \$1500 to \$7000 with an expected cost of \$3300.

6.2.1.3.5 Fins. Use of small fins for aerodynamic stability would be required, and their mass is estimated at 20 kg. As in the case of the coaxial Earth-orbital projectile, the cost range is \$1300 to \$3300 per projectile, with an expected estimate of \$2000.

6.2.1.3.6 Sabots. Both a forward and aft sabot would be needed to support the projectile correctly in the tube during acceleration. The aft sabot would also serve to conduct the railgun current across the back of the projectile. The sabot masses are estimated at 200 kg for the forward sabot and 100 kg for the aft sabot. The aft sabot is expected to be made of an inexpensive non-conductor to protect the projectile and a conductive plate to provide a current path. The forward sabot would be made of a non-conductive material, possibly a plastic. Under the assumption that the strength associated with exotic materials is not required for the railgun accelerations, it is likely that the copper alloy will cost approximately \$1.78 per kg while the plastic will cost \$2.00 per kg. The labor factors for the sabots should be low (1.5 to 2.0), so the cost range is estimated at \$900 to \$1200 with an expected value of \$1000.

6.2.1.3.7 Checkout, Equipment and Supplies. The launch-site checkout is expected to require only a few man days. As for all other projectiles, this is estimated to cost from \$1000 to \$5000 with an expected cost of \$2000.

6.2.1.3.8 Instrument Package. The instrument packages for all projectiles are expected to cost about the same. Through design for high production rates, this cost is expected to be in the range from \$1000 to \$3000, with an expected value of \$2000.

6.2.1.3.9 Propulsion System. The propulsion system and ACS/astrionics for the railgun projectile is very similar in mass to that of the coaxial projectile, and the same costs are used. These are low and high costs of \$25,000 and \$200,000, with an expected cost of \$100,000.

6.2.1.3.10 Transportation. Transportation charges for all projectiles are established at \$500. See Section 6.2.1.1.9 for additional information.

6.2.2 Operations

Operating personnel and operations support would be located both in the continental U.S., as well as at a remote mountainous site. Since most missions have been identified as going into either a 28.5-degree

orbit or a 0-degree (equatorial) orbit, the mountainous site would not be located in the continental U.S. For multiple daily launches to a single space station, the launch site would be located within a few degrees of the equator and would launch to an equatorial space station.

In addition to the technicians required for the launcher and projectile support, an ongoing engineering effort would be necessary to maintain and improve the launchers and their operational effectiveness during their lifetime. This level of engineering effort cannot be forecast precisely at this time. It is expected, however, that most of the design, assembly, and repair/rebuilding efforts would be accomplished by the operations staff envisioned in the cost estimates. Manufacturing efforts, however, would be contracted. Because of the difficulty in transporting large, very heavy equipment, it is expected that much of the final assembly, repair, and rebuild effort would be conducted on site. Accordingly, for the personnel estimates, it is expected that about half of the managers and engineers would be located in the continental U.S. and the others would be located on site. It is expected that projectiles would be built and checked out before being transported to the site. Any final efforts would consist of: loading propellants and fluids, setting initial conditions for guidance hardware, and making a final test of satisfactory payload conditions using a preprogrammed computer.

The personnel requirements are estimated to range from about 305 to 885 people, with expected estimates in the range of 500 to 540 people, depending on the specific system being considered (see Table 6-8). The cost estimates are dependent upon the assumption that the equipment is inherently reliable. The cost estimates also provide for sufficient spares that rebuilding of equipment can proceed on a schedule with little disruption for emergency repairs. For example, it is assumed that the brushes on the homopolar generators would have a normal wear life sufficient for thousands of launches, which would permit at least two years between brush replacement. The annual purchase of supplies to support the launcher facility is estimated to be about twice that needed to support the power plant. There is insufficient information to justify a specific level for supplies for the installation; these levels were selected as being a reasonable assumption. For the hybrid EML/rocket launchers, the power plant costs represent a small, non-nuclear plant; commercial power would be preferred if available. If commercial power were available, these supply costs would be considerably reduced (one-half to two-thirds), but the personnel would be needed to maintain the local distribution system.

The personnel estimates, as well as supply estimates, are given in Table 6-8. A value of $50,000 per man-year is used in estimating the cost of the staffing. This includes an allowance for launch-site personnel overhead, e.g., transportation for vacation purposes, and is a direct cost estimate. No estimate of support for programs which use the launcher for scientific or technical purposes is included in the estimates of annual expenses which range from $16.0 M to $64.0 M with expected estimates in the range from $32.0 M to $36.5 M.

6.3 Development Test Program

Details of a development test program are difficult to project at this time because of uncertainty in the technology. At this time, the development tests are envisioned as having two major aspects. The first aspect would be a test of one or two launcher tube segments to verify the performance of launcher segments and other subsystems. An inert test projectile would be accelerated at full thrust to give confidence in the design before any major construction at the launch site were undertaken.

The second aspect of the development test would be part of the transition from construction to operation. At this time, it is expected that most of the investment in the launcher would be complete. The development test would concentrate on assuring that the controls operate correctly and that the terminal velocities can be achieved reliably. In addition, there would be a need to verify adequacy of projectile designs. While some of this verification can be done by subscale electromagnetic launchers, there would still be a need to verify fullscale designs. If the fullscale tests are successful, the development test would be expected to last about one year. If they would be unsuccessful, rework of either the launcher or the projectiles would be required and non-productive costs would mount. The first component of the development test, that of full-sized segments(s) of the launcher, is expected to preclude the need for any major investment period rework of the launcher design.

In addition to the construction crew costs, which are included in the development/investment cost estimate, there are operating crew training costs, which would start during the construction phase and continue through the development test phase. These costs are approximated by using two years of operations crew costs.

These considerations are taken into account in the development test program which includes:

- Testing of an all-up launcher segment (in the continental U.S.)
- Development of projectiles, estimated at 400, 600, and 800 man-years of effort at $100,000 per man-year
- Transient housing for 500 to 1000 people at $10,000 per person and convertible to permanent transient housing of 250 units
- Two years of launcher operations cost.

The development test program estimates are given in Table 6-9. A summary of investment and operations cost estimates is given in Table 6-10.

TABLE 6-10. EML COST ESTIMATE SUMMARY ($, 1981)

Cost Category	Earth Orbital Launcher						Hybrid EML/Rocket Launcher					
	Coaxial			Railgun			Coaxial			Railgun		
	Low	Expected	High	Low	Expected	High	Low	Expected	High	Low	Expected	High
Research and Design	320.0	466.0	663.0	320.0	466.0	633.0	200.0	300.0	400.0	200.0	300.0	400.0
Facilities and Supporting Systems	361.6	510.0	674.0	361.6	510.0	674.0	201.0	328.0	468.0	201.2	328.0	468.4
Launcher Systems	350.3	485.4	712.1	836.5	1,078.2	1,509.2	197.1	272.6	380.5	331.7	485.0	697.8
Development Test Program	158.0	234.0	365.0	158.0	234.0	365.0	158.0	234.0	365.0	158.0	234.0	365.0
Total Investment ($, M)	1,189.3	1,695.4	2,414.1	1,676.1	2,054.2	3,181.2	756.1	1,134.6	1,613.9	890.9	1,347.0	1,931.2
Annual Operating Expenses ($, M)	21.0	34.5	55.0	24.0	36.5	57.3	19.3	32.0	58.8	21.0	34.0	64.0
Projectile Unit Costs ($, K)												
10 per day	34.2	117.9	236.7	35.0	119.1	233.1	82.1	124.2	169.7	94.1	142.2	193.7
5 per day	44.0	140.0	295.0	44.0	140.0	295.0	105.0	150.0	225.0	120.0	170.0	245.0
4 per day	46.0	150.0	310.0	46.0	150.0	310.0	110.0	160.0	235.0	125.0	180.0	255.0
3 per day	48.0	160.0	325.0	48.0	160.0	325.0	115.0	170.0	245.0	130.0	190.0	265.0
2 per day	50.0	170.0	340.0	50.0	170.0	340.0	120.0	180.0	245.0	135.0	200.0	275.0
1 per day	52.0	180.0	355.0	52.0	180.0	355.0	125.0	190.0	255.0	140.0	215.0	290.0
Total Program Estimates--Investment and 30 Years of Operations												
- 10 Projectile/Day	5,564	15,586	29,927	6,229	16,191	30,424	10,325	15,694	21,960	11,825	17,938	25,061
- 5 Projectiles/Day	4,228	10,395	20,225	4,805	10,814	21,051	7,083	10,307	15,697	8,091	11,675	17,265
- 1 Projectile/Day	2,389	4,701	7,961	2,966	5,120	8,787	2,703	4,175	6,170	3,054	4,721	7,027

TABLE 6-9. DEVELOPMENT TEST PROGRAM COST ESTIMATES ($, M, 1981)

	Low	Expected	High
Launcher Segment Tests	75	100	150
Development of Projectiles	40	60	80
Miscellaneous Facilities	3	4	5
Two Years of Operations	40	70	130
Totals	158	234	365

6.4 Economic Evaluation of EML Concepts

The EMLs considered in this report are one alternative for the Earth-to-space transport of bulk materials. This section provides cost information for some near-term alternatives to the EMLs and then provides a comparison of transport costs for two mission applications using EMLs or their alternatives. Section 2.0 describes and develops transport projections for both LEO space station and manned lunar base supply missions. This section evaluates the economic usefulness of EMLs to these missions.

6.4.1 Launch Systems Alternatives

In addition to the Space Shuttle and its potential derivatives, other launch systems are also considered which might be available in the year 2000 and might be competitive with EMLs. The hybrid EML/rocket concepts with EML velocity capabilities of 2 km/s are viewed as being in this category in that they are within reach of a moderately ambitious program while the Earth-orbital EMLs have slightly higher costs, as well as higher technological uncertainty. Accordingly, within the context of supply missions starting in the year 2000, the hybrid EML/rocket concepts are used for cost comparison. For the more demanding lunar base supply mission, where launch velocities of 12 km/s or more would be required, an uprated Earth-orbital EML would be needed. Such a facility would require several launch tubes and its costs would be at least those of the two-launcher system discussed in the ESRL report.

The alternative launch systems considered and described below are:

- Hybrid EML/rocket
- LEO and lunar supply EML
- Four-stage solid propellant rocket
- Single-stage gas gun
- Rocket-powered sled
- Space Shuttle and Shuttle-derived Unmanned Launch Vehicles (ULVs).

These alternatives are normalized to provide an equivalent level of transport: 800-kg payload to the LEO Space Station at 500-km altitude or 500-kg payload to a lunar space station. For both the current and derived STS vehicles, payloads might be flown on a space-available basis, but in any case would be flown in larger quantities; they are considered to bear the same cost per kilogram as general cargo.

Hybrid EML/Rocket. The expected total capital investment of the 2-km/s hybrid coaxial launcher is estimated at $1,134 M with annual operations costs of $32 M. Projectiles for this concept have an expected unit cost of $124 K at a production rate of ten per day. At lower production rates, the unit costs are expected to be about 50 percent higher because production overhead would not be amortized as thoroughly. The expected costs for the railgun hybrid EML are slightly higher and are shown in later tables.

LEO and Lunar Supply EML. This concept has not been investigated in detail in this report. It would require several launcher tubes to accomodate the expected launch rate. The facility requirements would approximate those of the ESRL system (Rice, et al, 1982). The investment accordingly is estimated to be $5 B, and annual support costs are estimated at $58 M. The projectiles, however, would cost about the same as the Earth-orbital projectiles discussed in Sections 6.2.1.2 and 6.2.1.3. The high investment cost is due to the limited launch window which will require multiple launches within a few hours on a few days each month. Launches to the LEO space station, however, would have daily launch windows.

Four-Stage Rocket. The four-stage rocket envisioned for use in the LEO supply mission would be a modified version of the three-stage projectile used for the railgun hybrid EML together with a first stage which would replace the boost provided by the EML. The entire vehicle would be reoptimized to reflect the lower levels of acceleration over longer periods of time which would be required while flying in the lower atmosphere. These lower levels of acceleration would permit the entire vehicle to be designed to slightly lower mechanical strength conditions than the EML projectiles, but no major cost savings is expected. After the initial design and test phase, the costs for the four-stage rocket would be strongly dependent upon materials usage rather than manpower as is the case with current vehicles. The production

philosophy, moreover, must be the same as for the EML projectiles: capital (machines) should be used in preference to labor.

The four-stage rocket would have a total mass of 48,300 kg with the first stage having a mass of 33,300 kg and the three upper stages a total mass of 15,000 kg. The three-stage railgun projectile is estimated to cost \$142 K with a range from \$94 K to \$194 K at a production rate of ten per day (see Table 6-7). By scaling the first stage cost from the 0.6 power law on the ratio of the mass of the three-stage projectile to the first stage, the first stage is expected to cost \$229 K, with a range from \$150 K to \$313 K. The entire four-stage vehicle, at a production and launch rate of ten per day would then cost from \$244 K to \$507 K with an expected cost of \$371 K.

To check the validity of these estimates, J. L. Van Cleave, Deputy Scout Project Manager, NASA/LaRC, was contacted. Based on a brief look at his information, Van Cleave estimated that the Scout could be produced for about \$500 K at a rate of ten per day, but that the production philosophy would have to be greatly modified. For example, the Scout vehicle is assembled and checked out at the factory and again at the launch site. To meet the quoted price, labor intensive processes such as the double checkout would have to be eliminated. Accordingly, the expected cost of the larger four-stage supply vehicle is considered to be reasonable given that it would be a new design and that production facilities and equipment would be scaled to produce the desired number of vehicles without labor intensive procedures.

The cost increase per unit due to lower production rates of one or two per day is not expected to be much higher (on the order of 50 percent higher). This increase would be principally due to poor amortization of overhead costs which would not be spread as well as at higher production rates. The expected cost per unit at a rate of 365 per year is then \$557 K, with a range from \$336 K to \$760 K. Launch services are included in these estimates, but ongoing engineering and support would cost about \$10 M per year.

The original Scout development cost was about \$16 million in 1960 dollars or about \$56 million in current dollars (Scout Program, Langley Research Center, 1976). The development of the four-stage rocket is expected to cost about \$100 M because of its larger size and higher performance, especially in terminal guidance accuracy for retrieval by space station personnel. Production facilities to achieve the rates considered (365 to 3650 per year) are expected to cost on the order of \$50 M. Final assembly and checkout at the launch site, payload mating, and launch and post-launch control facilities are included in the development and production estimates which total about \$150 M.

Gas Gun. A single-stage gas gun was briefly examined and found to be technically feasible for boosting a 15,000 kg projectile to 2 km/s within a 2 km-long tube with a cross-sectional area of 1 meter. Using the ideal gas equation ($pV = nRT = \frac{1}{2}mv^2$) with temperatures of 1273 K

and 1773 K, 51,003 kg and 36,619 kg of steam (from H_2/O_2 propellants) would be required to drive the projectile at 100 g's with 148 atmospheres of pressure. The temperatures were selected as the range which could be tolerated by steel alloys after the steam expands adiabatically within a refractory combustion chamber. To achieve higher launch velocities, however, higher temperatures would be required and these could lead to materials problems. The major technical challenge for the 2-km/s gas-gun, however, lies in the introduction of 36 to 50 MT of cryogenic propellants into the combustion chamber in two seconds.

The gas gun envisioned for the LEO Space Station supply mission is expected to cost less than the hybrid EML/rocket launcher because of its inherent simplicity and relatively low technological uncertainty. Since many of the same facilities would be required, and the propellant manufacture and storage requirements would be significant, the development/investment costs of a gas-gun facility would be greater than those for the four-stage rocket. Also, the potential for growth of a single-stage gas gun beyond 2 km/s is limited by the achievable propellant temperatures and by the ability to provide materials which can withstand those temperatures. Accordingly, higher velocities would require additional stages and thus would increase the complexity and costs of the launcher system.

The brief investigation indicates that the gas gun performance potential should be equivalent to the hybrid EML concepts and if a fixed-azimuth launcher in this velocity region is needed, the concept should be kept in consideration.

Rocket Sled. A liquid-propellant rocket sled was considered as an alternative to the hybrid EML concepts. The rocket sled would be placed on the side of a mountain slope with rails of over 2-km length. The rocket sled would accelerate a 15,000 kg projectile for 2 km, release the projectile, decelerate, and return for reuse. Initial calculations assumed the use of H_2/O_2 propellants and modified Space Shuttle Main Engines (SSMEs) with a mass of 6000 kg per engine. More than 10 SSMEs would be needed to produce the required thrust. A solid-propellant rocket sled was also considered. Preliminary calculations indicated that a very rapidly burning solid rocket motor (two-second burn time) with a diameter at least 100 times greater than the length might also be capable of providing the desired performance. Previous solid motor design experience has been with length-to-diameter ratios from about 1:1 extending to long, thin motors. This solid rocket motor would have problems maintaining adequate chamber pressure while permitting a very high mass flow. The four-stage solid rocket discussed above was felt to be superior to the rocket sled because the motors would be more conventional, the acceleration could be lower than 100 g, and a vertical launch would be preferrable.

Space Shuttle. The current Space Shuttle has a nominal payload capability of 29.5 MT (65,000 pounds) to a 296-km (160 n.mi.) orbit at 28.5 degrees inclination. Projections of the recurring Shuttle cost

per launch after its initial learning problems are resolved and higher launch rates are achieved, fall in the range from $41.7 M to $58 M in 1981 dollars. The low estimate includes the effects of probable modifications to the Shuttle to reduce operating costs and/or increase performance. The low estimate was originally provided by Frank Williams of Martin Marietta Corporation and Mike Van Hook of NASA/MSFC as part of the work for, but not reported in, the Phase II Final Review, "Technology Requirements Study, Shuttle Derived Vehicles", April 26, 1982. The high estimate is made by Battelle; it reflects inflation since the Shuttle pricing base date is 1975 and does not include additional investment in modifications to the Shuttle.

The Space Shuttle will have a performance penalty to achieve the higher Space Station orbit (500 km versus 300 km nominal STS orbit). It would also have another performance penalty, or a new launch site may be required, to support a space station at an orbital inclination other than 28.5 degrees. To account for both performance penalties associated with flights to a 500-km orbit and ongoing operational improvement costs, the $58 M recurring cost is used for the Space Shuttle, and the lift capability of 29.5 MT is reduced to 25 MT. However, to reflect the likelihood that much of the bulk cargo may be flown on a space-available basis, no additional penalty is assumed for container weight. This leads to a calculated cost of $2320 per kilogram.

Advanced Manned and Unmanned Vehicles. There are several concepts for unmanned Shuttle-Derived Vehicles (SDVs) or Unmanned Launch Vehicles (ULVs), in addition to several approaches to uprating the manned Shuttle for transport to LEO. There will also be a need for Orbit Transfer Vehicles (OTVs) with the capability of transporting a manned module beyond LEO. The OTV (and very likely the ULV) would be required to transport large equipment and/or personnel for the lunar base program. If EMLs were developed to transport bulk materials (and especially propellants) a substantial reduction in the number of ULV and OTV flights would be possible, but the development of advanced chemical propulsion vehicles cannot be avoided. Development estimates for these vehicles lie in the $1.0 B to $2.4 B range and are dependant upon modifying current technology and/or hardware (e.g. Shuttle or IUS). Recurring cost estimates for Earth-to-LEO vehicles range from $36 M (Martin Marietta--Uprated Shuttle) to $48 M (Martin Marietta--Shuttle Derived Vehicle). Since the ULV would be largely expendable, this estimate is considered low in comparison with other large expendable vehicles. A reusable OTV is expected to have somewhat lower recurring costs ($5 M to $10 M per round trip) but only when the cost of transporting its propellants to the space station is not included. For transport of bulk materials between LEO and a lunar base or a lunar space station, both an Earth-to-LEO vehicle and an OTV will be required. The ULV is believed to have a relatively low development cost ($1.5 B), but a relatively high recurring cost ($70 M) reflecting the relatively low reusability of this vehicle. The 68 MT payload yields a transport cost of $1029 per kilogram.

The OTV is estimated to have a development cost of $1.0 B, and a recurring cost of $5 M per round-trip is considered reasonable

if the cost of transporting propellants or water (and processing cost) is not included. The cost for these propellants largely consists of their transport to orbit by the ULVs, which are costed separately. When the cost of propellant transport by ULV (and by OTV to the lunar space station) is included, the cost per kilogram for OTV cargo would be about $13,200 ($6000/lb).

6.4.2 Comparison of Alternatives

Comparison of the EMLs with the alternatives indicates that the EMLs could perform these missions with considerable cost advantage. This advantage also holds under the more stringent criterion of discounting cost streams at ten percent. This comparison technique takes into consideration the value of monetary investments over time; its major effect is to penalize large early investments which, as for the EMLs, must be made before benefits of using the investment can be realized. The discounted and undiscounted cost comparisons are shown in Tables 6-11 and 6-12 for the LEO space station supply mission and the lunar base supply mission, respectively.

The results of this cost assessment of two missions with their multiple launches per day tend, however, to conceal the fact that high launch rates are essential to the economic use of the EMLs. If the launch rate were to fall to one launch per day (365 per year), the four-stage rocket would produce lower discounted total program costs than the EML. This is illustrated in Figure 6-1 where discounted program costs are plotted as a function of equivalent 800-kg payloads per day. This figure illustrates the total program costs for the LEO Space Station supply mission alternatives over the period 2000 to 2029 and presents the four-stage rocket and EML programs with development costs included. Space Shuttle, ULV, and OTV are shown in terms of total discounted program cost per kilogram of cargo without development costs. A $500/kg line is included to show its close approximation to the four-stage rocket program. A series of possible EML programs is also presented. These are hypothetical programs with development costs ranging from $1 B to $5 B. The same projectile unit costs are used for all five programs. The hybrid EML reference concept falls between the two lower EML program curves and has lower discounted costs than the four-stage rocket as long as multiple launches per day are achieved.

The lunar base bulk supply mission has a similar analysis and results. The major changes for this mission are the use of a $5 B launcher complex, use of projectiles with 500-kg payloads (versus 800 kg for the LEO mission), and the fact that the four-stage rocket cannot be used to supply the lunar space station. If the four-stage rocket were used to supply the LEO portion of this application, the lunar space station portion of the EML launches would be less economical because of lower utilization of the investment in EMLs. The cost of transporting propellants or other bulk cargo to the lunar space station by chemical propulsion techniques (e.g. OTV) is nevertheless sufficiently high (about $13,200/kg), that the EML would still be justified for this mission alone. Other uses would enhance this advantage.

TABLE 6-11. COST COMPARISON OF ALTERNATIVES FOR SPACE STATION SUPPLY MISSION (2000-2029) ($, M, 1981, EXPECTED)

Cost Category	Coaxial Hybrid EML	Railgun Hybrid EML	4-Stage Rocket	Current STS
(Investment Period)	1985-99	1985-99	1995-99	Past
Total Investment	1,134.6	1,347.0	150.0	Paid
Annual Operations	32.0	34.0	10.0	Included
Projectile Unit Costs, 800 kg Payload				
5/Day	0.150	0.170	0.570	1.856
1/Day	0.190	0.215	0.663	1.856
30 Year Recurring Cost (2000-29)(800 kg P/L) (1-4.5 Launches/Day)	6,112.7	6,692.1	15,536.6	56,651.9
44-Year Total Cost	7,246.3	8,039.1	15,686.6	56,651.9
44-Year Total Cost Discounted at 10 Percent	940.6	1,059.3	1,106.1	3,693.9

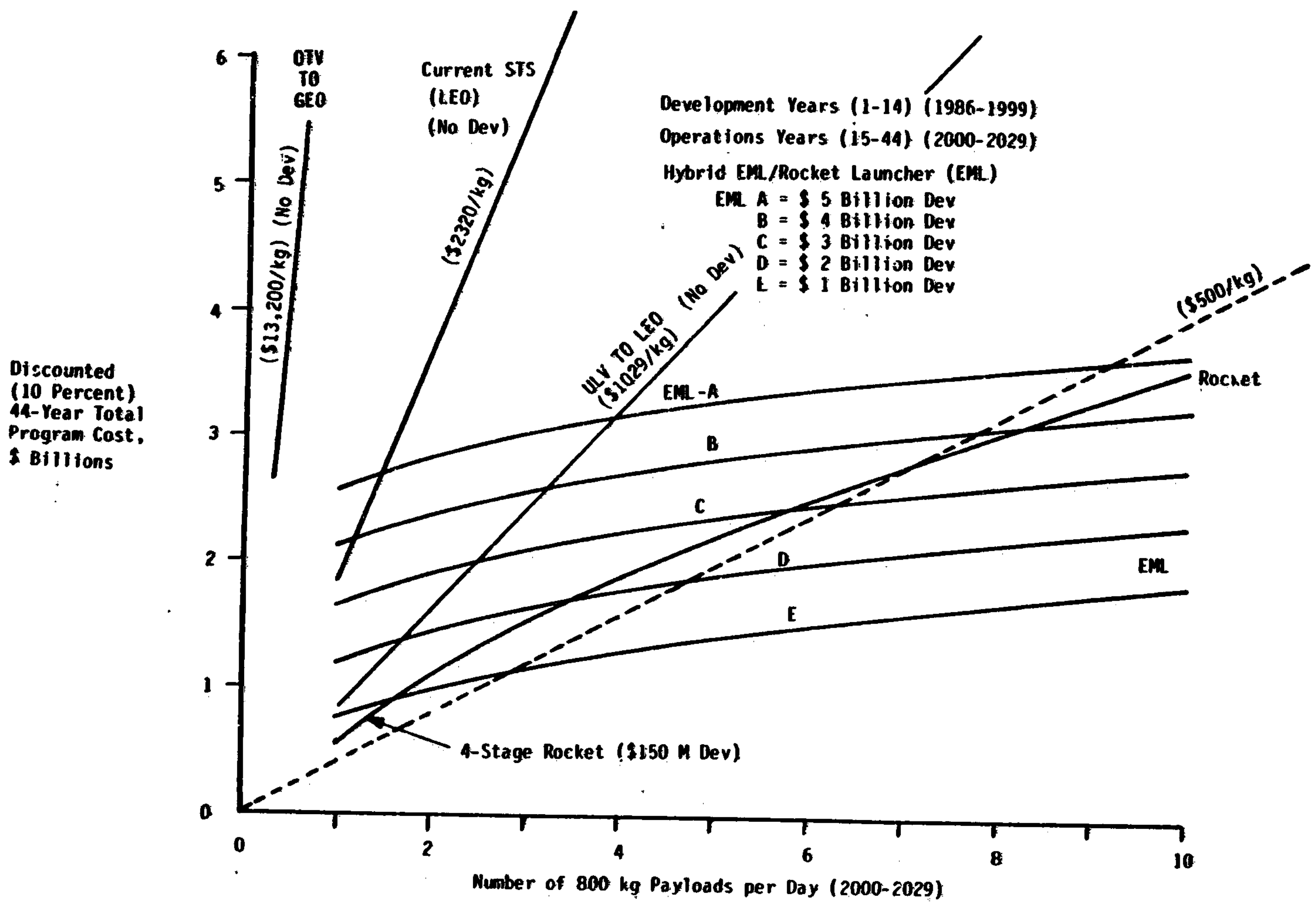


FIGURE 6-1. EFFECTS OF DISCOUNTING ON TOTAL PROGRAM COSTS

TABLE 6-12. COST COMPARISON OF ALTERNATIVES FOR 2010-2030 LUNAR BASE SUPPLY MISSION ($, M, 1981, EXPECTED)

Category	All Chemical Propulsion (STS, ULV, OTV)			Chemical Propulsion and EML (STS, ULV, OTV, EML)			
(Investment Period)	2000-2004			2000-2009			
Total Investment (STS-Paid;/ ULV-$1.5B; OTV-$1.0B; EML-$5B)	$2500M			$7500M			
(Flights: 2010-2030)	STS 126	ULV 546	OTV 651	STS 126	ULV 0	OTV 126	EML 17,584
Cost Per Flight	$58M	$70M	$5M	$58M	--	$5M	$0.170*
Cost of Flights	$48,738M			$12,145M			
30-Year Total Cost	$51,238M			$19,645M			
30-Year Total Cost Discounted at 10 Percent	$2,635M			$1,891M			

* Plus $ 58M annual operations.

It should also be noted that the two missions discussed are independant and therefore have additive launch rates. This would help in achieving economical use of an EML facility for LEO and lunar operations. It would be difficult to justify an expensive (more than $2 B) EML facility in the face of competition from the four-stage rocket, if fewer than two launches per day were expected.

7.0 TECHNOLOGY ASSESSMENT

The objective of this task was to assess the status of current EML technology as it applies to electromagnetic launcher concepts for space applications, and to provide supporting research and technology (SR&T) recommendations to NASA. The information developed over the course of this study has been used to assess the technology. This section of the final report is broken down into two major subsections. First, the technology evaluation subsection, will discuss some of the technological areas that need further work. The second subsection provides a suggested plan for NASA to follow, if it desires to develop EML technology for propulsion applications in the space program.

7.1 Technology Evaluation

Technology development areas were identified based upon review of the open literature, discussions with EML experts, and attendance at several EML conferences to assess the state-of-the-art technology. As a result of the technology evaluation activity, recommendations were made regarding certain areas which require NASA supporting research and technology-SR&T (see Section 7.2). The SR&T recommendations were made in those areas that NASA must resolve before an EML Earth-orbit supply launcher development program could be implemented. It is expected that the DoD will, over the next several years, develop many areas of technology needed for EML Earth-orbit supply systems.

The primary areas requiring technology development which were identified in this study are listed below:

- Scale-up of existing EML systems
- Energy distribution and switching
- Projectiles
- Energy storage systems
- Projectile brushes and armatures
- HPG brush material
- Launcher tube bore
- Preboost systems
- Launcher structural support.

These areas are discussed in this section.

7.1.1 Scale-up

Scale-up of existing systems is required for space missions. Existing EML systems are under 10 m in length (the Los Alamos HYVAX railgun will be 13-m long to accelerate projectiles at 15 km/s); to be effective

space launchers, 2000-m long EML systems must be operational. Projectiles must also be scaled. The reference concept projectiles range from 3000 to 15,000 kg and must be launched between 2 and 12 km/s. Railguns have reached 10 km/s velocity, but for gram-size projectiles. Coaxial accelerators have launched 5 kg projectiles, but at launch velocities on the order of 0.1 km/s. Scale-up of current systems must be demonstrated before beginning development of EML systems.

7.1.2 Energy Distribution and Switching

Demonstrations of distributed energy storage (DES) railgun systems have been progressing over the last few years. A DES demonstration program was begun at the University of Texas at Austin Center for Electromechanics (UT-CEM) in 1980. In July 1983, UT-CEM fired a ten-stage 4-m railgun accelerating the projectile to 3 km/s. The investigation goal of UT-CEM is to accelerate 1 to 3-g projectiles to 10 km/s with this railgun (Holland, 1984). Vought, in connection with UT-CEM, is building a 5-stage 3.65-m railgun with a design capability of propelling a 60-g projectile to 3-4 km/s.

By nature, coaxial accelerators are distributed systems. The dM/dz parameter in the thrust equation is not constant, so the current must be synchronized with the projectile motion (see Section 3.1.2 and Appendix D).

For all distributed energy store systems, switching of current at the right time is crucial. For railgun systems, current must be switched when the projectile reaches a new segment. The coaxial EML reference concepts continually switch drive coil turns in and out so that ten turns remain active behind each projectile coil.

Switching is considered a critical technology area, until switching issues are resolved EML system development will not take place. Critical issues are switch performance and life times for an EML system which launches an average of ten times per day over a 30-year system life. The switches must perform over many launch cycles without being replaced. Frequent replacement would be prohibitively costly in terms of time and economics.

7.1.3 Projectile Development

Critical issues to the design of an EML-launched projectile are: high acceleration constraints, aerodynamic considerations, and materials selection. All of the above are critical to EML system development and all require in-depth study.

The high accelerations of launch (100-3600 g's) produce unique requirements for NASA programs. Acceleration effects on propulsion systems (sensitive liquid-propellant lines and tankage, and solid propellant grain) need to be investigated. A finite-element analysis is required to establish confidence in conceptual designs, particularly in projectile structure.

Instrumentation and stabilization pop-out mechanisms (fins) must also withstand the acceleration loads. Existing DoD expertise in the area of high-acceleration tube-launched artillery and missiles should be able to contribute significantly to this investigation.

Aerodynamic drag considerations are important from the standpoint of the velocity required to launch a projectile through the atmosphere to obtain the desired end condition. The higher the drag, the greater is the loss of velocity and energy along the flight trajectory. The higher the drag, the greater will be the sonic boom generated by the projectile as it transverses the atmosphere. Daniel and Milton (1980) indicated that low-drag bodies are possible; experimental research is required to verify this.

Aerodynamic stability as the projectile leaves the muzzle of the launcher tube is another critical issue. Preliminary assessment conducted during this study indicates that stability of the vehicle is critical to the performance of the system. Initial concepts for establishing flight stability include the use of fins at the rear of the projectile and the center of mass being nose forward. (If the pitching rates can be made fairly low, the vehicle will not have a chance to pitch very far during the few seconds that it flies through the atmosphere. Pitching moments of the order of perhaps 10 degrees per second would still allow the vehicle to fly out of the atmosphere without any problem.) A very important consideration in the launch of the Earth-to-orbit railgun projectile is the jettison of the sabot, as the projectile leaves the muzzle of the rail launcher tube. The sabot would have to break free in a very timely way so that a pitching moment is not imparted to the vehicle. Pop-out fins are used in the round-bore systems of the other references concepts. If stability should prove to be a problem, round bores could be used for all concepts and the projectiles spun-up prior to launch. Theoretical and experimental work is required in this area.

Aerodynamic heating is also a very critical aspect of the EML concepts, especially the Earth-to-orbit concepts. Initial assessment indicates that because the projectile flies rapidly through the atmosphere (7 km/s), there is little time for aerodynamic heating to melt the nose cone to any significant degree. The heating rates are very high, and it is expected that a fairly significant area at the stagnation point would be melted away, depending upon the latent heat of fusion and the melting temperature of the nose tip material. Steel was used for the nose cone in the Earth-to-orbit reference concepts. Experimentation is needed to determine appropriate materials for use.

7.1.4 Energy Storage Systems

Energy storage technology is critical especially for near-term concepts. Homopolar generators (HPGs) are commercially available at 6.25 MJ; UT-CEM has a 10 MJ HPG. In the next five to ten years, an order-of-magnitude more storage capacity for HPGs should be expected (telephone conversation with Dale Pryor, OIME). Other railgun energy

storage devices which may merit further investigation are batteries and the inverse railgun (Marshall, 1984b).

Various storage inductors are available for the coaxial accelerators; a Brooks coil was selected. The single inductor is simple, but does represent a possible single-point failure area. The multiple HPGs on the railgun systems are more complex and massive, but offer some redundancy.

When a detailed system design is conducted, energy storage devices will need to be considered carefully before selection is made. This is not a critical area of system development, however.

7.1.5 Projectile Brushes and Armatures

Projectile brush and armature selection are dependent upon launch velocity. Several coaxial accelerator concepts which were considered used projectile brushes to pick up excitation currents from the drive coils. These concepts were not selected for reference concepts, because of uncertainties of brush survival at velocities above 1-2 km/s. Westinghouse is conducting advanced brush work. At 1 km/s, brushes survived with no rail erosion. William Snow (EML Research) expects no problems, even at 7 km/s. If experimental evidence shows that brush deterioration is not a problem during launch, these launcher concepts may be revisited for consideration in a system design.

Railgun armature selection depends upon velocity of launch. Below 2-3 km/s, a solid metal armature may be used (similar argument as for projectile brushes). Copper and aluminum are the most commonly used armature materials. Aluminum armatures have been used at velocities of up to 3 km/s. Above these velocities, plasma armatures are needed; but at the initial low velocities, erosion of the rails may occur unless preboosting of the projectile is done. Solid/plasma armatures have been suggested as the optimum solution. Extensive experimental research is needed before this armature is selected for system design.

7.1.6 HPG Brush Materials

The ability for HPGs to store energy depends directly upon the speed at which the homopolars can operate. Critical to HPG reuse economics are the brush materials that transfer the charge. Discussions with William Weldon at the University of Texas Center for Electromechanics indicate that HPG speeds are currently limited to approximately 220 m/s with long life at the brush interface. The use of advanced brush materials would allow increased speeds which would in turn allow more energy to be stored. The University of Texas Center for Electromechanics is investigating HPG brush technology and has improved the brush materials to the point where they can obtain speeds of 300 m/s using advanced materials, but still experience a great deal of erosion.

A major issue in the EML systems would be the required replacement rate of brush material. It is desirable to operate the HPGs at high speeds

and with minimal maintenance over long periods of time. The ease of brush replacement is also a major issue. Currently, the commercially-available OIME HPG must be disassembled to replace the brushes (telephone conversation with Dale Pryor, OIME). When 750 to 3600 HPGs are used, disassembly would require an excessive amount of time and would likely limit the launch rate severely. These issues are critical to the operational cost of the system. Therefore, there is a need to advance the state of technology in the area of brush materials for HPGs.

7.1.7 Launcher Tube Bore

The bore design and especially bore/projectile interfaces are key factors in reusability of EML tubes for both railguns and coaxial accelerations. The selection of bore shape, materials, and support structures is important for launcher tube longevity.

Round-bore railguns have been experimented with more since the previous ESRL study. Lawrence Livermore National Laboratory and Los Alamos Scientific Laboratory are the primary investigators. A round bore offers several advantages over a square bore. A round bore allows the capability for spin-stabilizing of the projectile. Pop-out fins can be used, reducing the bore diameter, and thus the launcher system mass. A round-bore railgun also allows for the possible remachining of the bore which would permit a long operational life for the system.

The proper selection of materials for tube structural support is critical to maintaining launcher alignment and reducing bore damage due to launch bursting forces.

Another critical technology area that needs to be investigated prior to development of an EML system, is the evaluation of: (1) projectile/bore friction, (2) sabot/projectile friction for railgun systems during the launch phase, and (3) the tolerances that are required to avoid projectile destruction during the launch phase. Aspects of friction should be evaluated for both square and round bore launchers. Analytical and experimental work should be conducted to establish the significance of friction and critical dimensions of the bore and the projectile at the time of launch. Experiments could be conducted in existing facilities. The problem of launcher tube movement as a result of continued firings is an important aspect related to the bore tolerances. This aspect determines the reusability of the launcher bore after numerous firings.

7.1.8 Preboost Systems

Reusability of the rails is a major issue for plasma-armature railguns. The long dwell time of the plasma during initial accelerations tends to erode the rails. For a large-scale railgun system to be developed, rail damage must be prevented. Pre-accelerating the projectile to 1 km/s or so greatly reduces the damage.

Preboost systems were used in all four reference concepts in this study, even though only one was a plasma-armature railgun. Preboost

systems were assumed to provide a small velocity (50 m/s or so) to initiate projectile motion to overcome the downward forces due to the incline of the 10-degree and 30-degree launcher elevation angles.

Helium gas injectors are commonly used to preaccelerate projectiles before entering the railgun section. Typically the projectile is accelerated to velocities of 0.5 to 1.0 km/s using high-pressure helium (3000 to 5000 psi). Various preboost concepts would need to be tested before selecting an appropriate method. Preboost systems are not considered to be critical to the development of an EML system in that the basic technology is available.

7.1.9 EML System Structural Support

Large support structures are required for the EML systems for space missions for several reasons. The launcher tubes are very long (the reference concepts would be 2040 m long) and correct alignment must be maintained. The bursting force on the rails or drive coils during launch must be contained. The structure for the coaxial accelerators to contain tube stresses could be somewhat lower in mass than for railguns, because the projectile coil presence decreases the forces. Launcner structural support technology is not expected to be critical to the development of EML space-mission concepts.

7.2 Supporting Research and Technology (SR&T) Recommendations

Based upon the results of this study, supporting research and technology (SR&T) efforts have been prioritized and funding estimates have been made. Table 7-1 provides our estimates of 5-year funding requirements for our recommendations in 1984 dollars.

Four major areas of activity have been categorized: (1) EML experimental research; (2) EML mission requirements studies; (3) EML systems studies; and (4) special studies. The philosophy in developing the schedule for EML SR&T was based upon the fact that DOD agencies are funding EML technology development at an ever increasing level and that NASA should only complement this effort for the space transport mission. Subsections below discuss the suggested areas of NASA-funded SR&T.

7.2.1 EML Experimental Research

To keep abreast of the latest technological developments in the EML area, Battelle recommends that NASA/LeRC continue a low-level experimental work profile with the equipment in hand and concentrate the effort on hypervelocity impact testing of Space Station power system components (space debris and meteoroids) and continue to investigate switching technology for distributed energy railgun systems and coaxial systems. Funding levels are provided in Table 7-1. Funding for Item A increases in the latter years because of testing with Space Station power system prototypes.

TABLE 7-1. ESTIMATED FUNDING REQUIREMENTS (1984, K$) FOR RECOMMENDED EML SR&T ACTIVITIES

Activity	FY-85	FY-86	FY-87	FY-88	FY-89	Total
EML Experimental Research						
A. EML Demonstration and Hypervelocity Impact Testing	150	150	200	250	300	1050
B. Energy Distribution and Switching	50	50	--	--	--	100
	(200)	(200)	(200)	(250)	(300)	(1150)
EML Mission Requirements Studies						
C. Lunar Base Supply	200	100	--	--	--	300
D. Space Station Interfaces	75	75	--	--	--	150
	(275)	(175)	(--)	(--)	(--)	(450)
EML Systems Studies						
E. Preboost Systems Analysis	--	75	--	--	--	75
F. Projectile/Sabot Design	--	--	150	150	--	300
G. Propulsion Systems and Instrumentation Design	--	--	150	150	--	300
H. Total Launcher Systems Design Study	--	--	--	100	500	600
	(--)	(75)	(300)	(400)	(500)	(1275)
Special Studies						
I. Environmental Impact Assessment	--	--	--	--	150	150
K. Unassigned Studies	50	50	50	50	50	250
	(50)	(50)	(50)	(50)	(200)	(400)
Total SR&T	525	500	550	700	1000	3275

7.2.2 EML Mission Requirements Studies

Two study areas need further development to provide better insight and benefit quantification for EML missions. Our study showed that the far-term application of EML systems for large bulk-mass transport showed promise for two NASA programs: the Space Station and the lunar base. These studies (discussed below) should be done soon to provide guidance to the rest of the recommended SR&T activity.

A consideration of EML support of Space Station has not been part of the initial Space Station pre-program and program planning (and should not be). EML support may introduce new and/or additional requirements to a growth Space Station facility. Trades such as projectile retrieval using on-board automated systems or OMV-type systems should be considered. How many OMVs are needed and how are storage and handling of both the payloads and projectiles accomplished? For example, the Shuttle could return spent projectiles to the surface. The study of these and other related issues provide a better understanding of costs, burdens, and benefits of using an EML system.

Key issues that may affect the benefits of EML supply of a lunar base include the lunar base characterization, transportation alternatives, and on-orbit support capability. Some important considerations of the lunar base characterization include the lunar-produced oxygen benefits, crew size and rotation, cryogenic storage at the lunar poles, and lunar resource/product return (possibly using a lunar-based EML). Comparison of the EML supply to alternative, more-conventional vehicles is key to deriving the benefits of EML. Transportation trade studies would also include various types, sizes, and performances of Earth-based, space-based, and lunar-based transport vehicles used to supplement the EML with manned transportation. For example, the size of an OTV-type vehicle and the types of propellants are critical considerations. The studies of on-orbit support alternatives would address effects of storage, retrieval, and refurbishment capabilities for projectiles, vehicles, propellants, and payloads. The range of effects of any issue may have a large impact on cost benefits.

7.2.3 EML Systems Studies

After the refinement of missions requirements (see Section 7.2.2) for promising EML systems applications, various EML studies need to be accomplished. Because of the need for a preboost system on railgun EMLs and the potential of this system being a replacement for the EML hybrid launcher, the first study would be to investigate and analyze preboost systems, in particular, large light-gas gun systems. Second, projectile, sabot, propulsion systems, and instrument designs need to be designed in detail based upon Items C and D (see Table 7-1). Also, trade studies need to be accomplished on solid or liquid propulsion systems. These studies would also investigate projectile aerodynamics, materials, and structural problems. The goal would be to arrive at designs for both a railgun and coaxial launcher system for Earth orbit and lunar missions.

Total funding for these is estimated at $600 K in FY-87 and -88 (see Table 7-1). Once these studies have been completed, a total launcher systems design study would be needed to determine the benefits of both railgun and coaxial systems versus conventional methods of space transport. In FY-88, a preliminary study would be conducted; a larger study $500 K would be accomplished in FY-89.

7.2.4 Special Studies

Special studies are defined to include unassigned studies and an environmental impact assessment to be accomplished in parallel effort with Item H.

8.0 REVIEW OF ALTERNATIVE EML CONCEPTS

In addition to railguns, which were investigated as an alternative to chemically-propelled launcher systems in Battelle's Earth-to-space rail launcher (ESRL) study (Rice, Miller, and Earhart, 1982), a number of other electromagnetic and electrothermal propulsion systems have been proposed. These concepts are reviewed briefly in this section and their suitability for the space missions considered in this report are assessed. Other concepts such as magnetoplasmadynamic and free-radical thrusters were not reviewed as they are considered to be primarily low-thrust techniques, at least for the foreseeable future. Those concepts that are reviewed include coaxial accelerators, electrothermal ramjets, electromagnetic rocket gun, and electromagnetic theta gun. Soviet work in electromagnetic acceleration is also reviewed.

8.1 Coaxial Accelerator Concepts

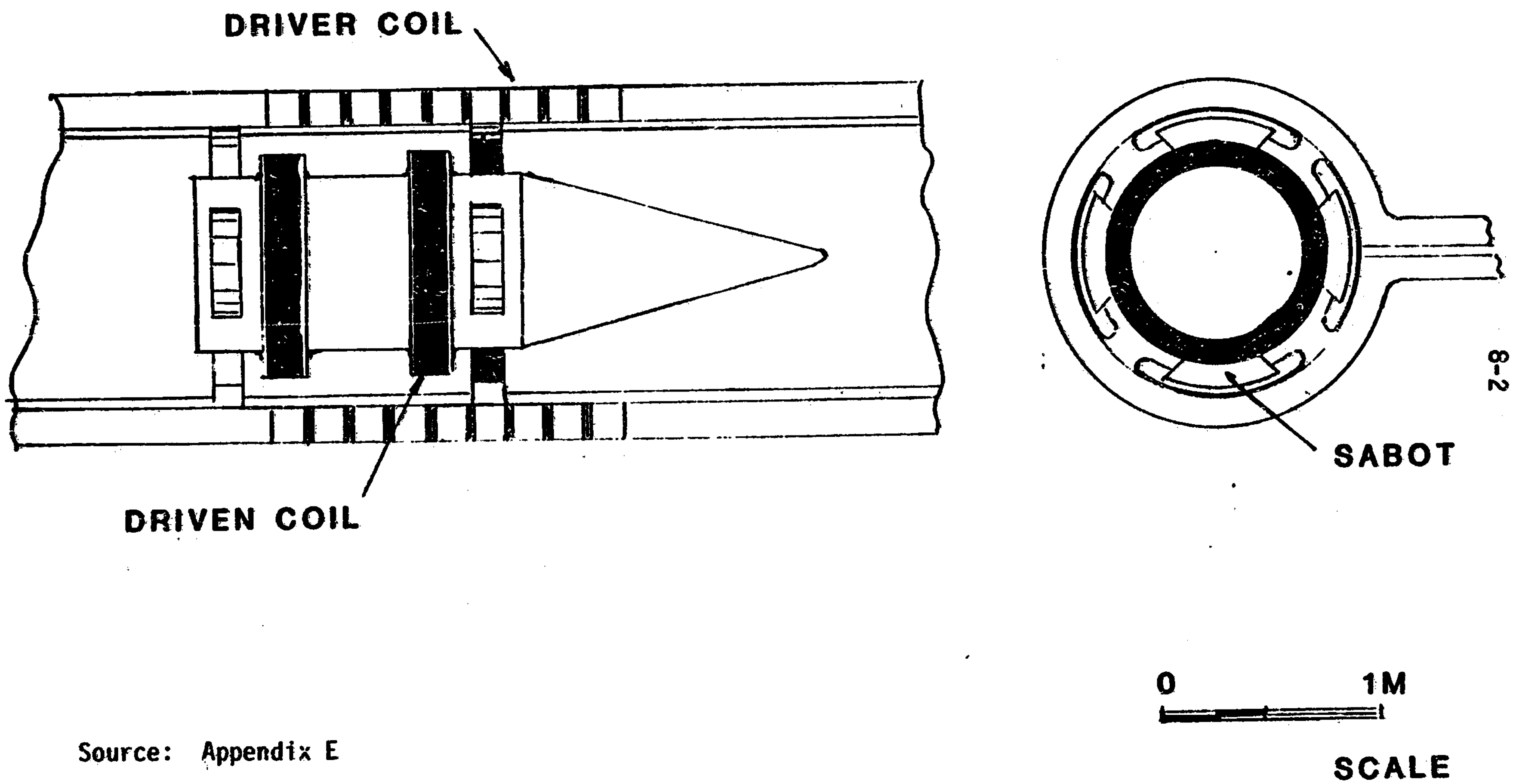
Two additional coaxial accelerator concepts were reviewed and were not selected for inclusion in the reference concepts. These were the Frequency Controlled Coil Driver, designed by O. K. Mawardi of Collaborative Planners, Inc., and the Solenoid Earth Launcher, designed by William Snow of the U.S. Army Armament Research and Development Center and currently with EML Research, Inc. Summaries of these concepts are given in this section.

8.1.1 Frequency Controlled Coil Driver

NASA/LeRC contracted with Collaborative Planners, Inc., and EML Research, Inc., to supply coaxial accelerator concepts as input to this study. Appendix E contains the report of Collaborative Planners' effort which is briefly summarized here. Two launcher concepts were designed by O. K. Mawardi, a hybrid EML/rocket concept and an Earth-to-orbit EML concept. Because of the limited payload mass and the technical uncertainty of superconducting the projectile coils, these concepts were not selected as reference concepts for further study.

Figure 8-1 presents a cut-away view of Mawardi's launcher concept. The launcher length was calculated at 2.0 km; this corresponds to 8888 drive coils for the hybrid mission. The projectile coils would be made of aluminum and would have a mass of 406 kg. From the results shown in Appendix E for the Earth-to-orbit concept, mass summaries were derived and are shown in Table 8-1. These numbers indicate that because of the large size of the projectile coils, the available mass is limited. The available mass must include the payload, orbit-circularization propulsion system, and structural mass. The velocity for the projectiles is also insufficient to reach orbital altitudes.

For the hybrid EML/rocket concept, ten projectile coils were assumed. To orbit an 800-kg payload (payload requirement, see Table 3-1), approximately 12,700 kg of solid propellant is required. After subtracting



Source: Appendix E

FIGURE 8-1. FREQUENCY CONTROLLED COIL DRIVER

the projectile coil mass (4060 kg), the remaining projectile mass would be 10,940 kg. To use this concept, either an optimization study must be done to increase the size of the three-stage rocket or the lower payload mass must be accepted.

TABLE 8-1. MASS AND VELOCITY SUMMARY FOR MAWARDI'S EARTH-TO-ORBIT PROJECTILE

Projectile Mass (kg)	Projectile Coil Mass (kg)	Available Mass (kg)	Final Velocity (km/s)
650	406	244	4.8
1000	816	184	3.3
6500	4060	2440	2.9

The concepts use brushes and guide rails to feed excitation current to the projectile coils. Because "it is essential to maintain current density in the brushes at a low safe value" to guarantee mechanical stability, the concepts are velocity limited. To launch at velocities above 5 km/s, Mawardi suggested that the projectile coils be cooled to superconducting temperatures with the current induced in the coils before launch; this eliminates the need for brushes entirely. However, superconducting projectile coils present new problems. For example, if the superconducting coils should become normal before leaving the launcher tube, damage to the projectile and to the accelerator could occur. Superconduction of projectile coils requires further analysis.

8.1.2 Solenoid Earth Launcher

Another coaxial accelerator concept which was briefly reviewed is the Solenoid Earth Launcher, designed by William Snow of the U.S. Army Armament Research and Development Center, now with EML Research, Inc. The review was based upon notes and viewgraphs (Snow, 1984) supplied late in the study effort and telephone conversations with Snow in April 1984. The purpose of this launcher would be to launch projectiles to Earth-orbital altitudes at velocities of 6 to 7 km/s. The launcher would consist of a single helix drive coil built in segments and a projectile with a single projectile coil and current pick-up brushes.

The proposed accelerator would consist of 1000 2-m drive coil segments, with each segment containing 60 coils. Approximately 2600 MT of copper would be required to build the launcher tube with a tube thickness of 5 cm. Two homopolar generators connected in series would supply energy to each segment at 70 MJ per segment (35 MJ per HPG); therefore, 2000 HPGs would be required. Since these would be self-excited

HPGs, Snow claims that there would be no need for intermediate storage solenoids (telephone conversation with William Snow, April 1984). Triggered vacuum gap switches would be timed to open and close to activate the drive coils as a function of projectile position. A maximum of 60,000 switches would be required (one for each drive coil). Figure 8-2 presents the Solenoid Earth Launcher concept.

Snow investigated three projectile masses: 2000 kg, 3000 kg, and 4000 kg. Each projectile consisted of a payload, propulsion system for orbit insertion, structure, and one projectile coil. The projectile coil for each of the three projeciles studied was assumed to be made of aluminum with a mass of 273 kg. Current would be fed to the projectile by means of brushes which would pick up the current from the drive coil.

The 3000-kg and 4000-kg projectiles were rejected by Snow because of heating and stress during launch. The projectiles were assumed to be precooled to 150 K, while the solenoid would be water-cooled during launch. Snow's simulations indicated that the 2000-kg projectile (with a single 273-kg projectile coil) would experience a 180 K temperature rise (to 330 K) and a 100,000-psi stress of which 40,000 psi would be solenoid compensating so that the back of the projectile coil would see 60,000 psi. The 4000-kg projectile with the same coil mass would experience a 500 K temperature rise and a stress of 200,000 psi.

This concept does not use one inherent advantage that the reference concept coaxial accelerators have over the railgun. The use of multiple projectile coils provides lower projectile launch stresses by distributing the stress throughout the body, rather than having the stress applied at the back of the projectile as in the case of a railgun launch. While one projectile coil in the center of the projectile (like Snow's concept) will reduce stresses somewhat over a similarly-sized railgun projectile; to get a useful payload for space delivery, the structural mass needs to be at a minimum. Multiple projectile coils help to increase the payload-to-projectile mass ratio by lowering projectile launch stresses which decreases the amount of structural mass required; however, this occurs at the expense of added system complexity. Table 8-2 lists the payload versus structural mass for Snow's 2000-kg projectile assuming a similar propulsion system to that used in the reference concepts (payload mass to propulsion mass ratio of 0.28, see Section 3.8). For comparison, the Earth-to-orbit railgun projectile concept contained 2730 kg of structure and the Earth-to-orbit coaxial projectile (with 40 projectile coils) contained 300 kg of structure.

Another concern is the use of projectile brushes at the required 7 km/s launch velocity. Sliding brushes have been used up to 1 km/s with no damage (see Section 7.1.5). Snow believes that the brushes could take velocities of 10 km/s or more with out damage. Since the brushes would be attached to the projectile, if they could survive long enough to feed current through the launch phase, it would be sufficient as reuse would not be required.

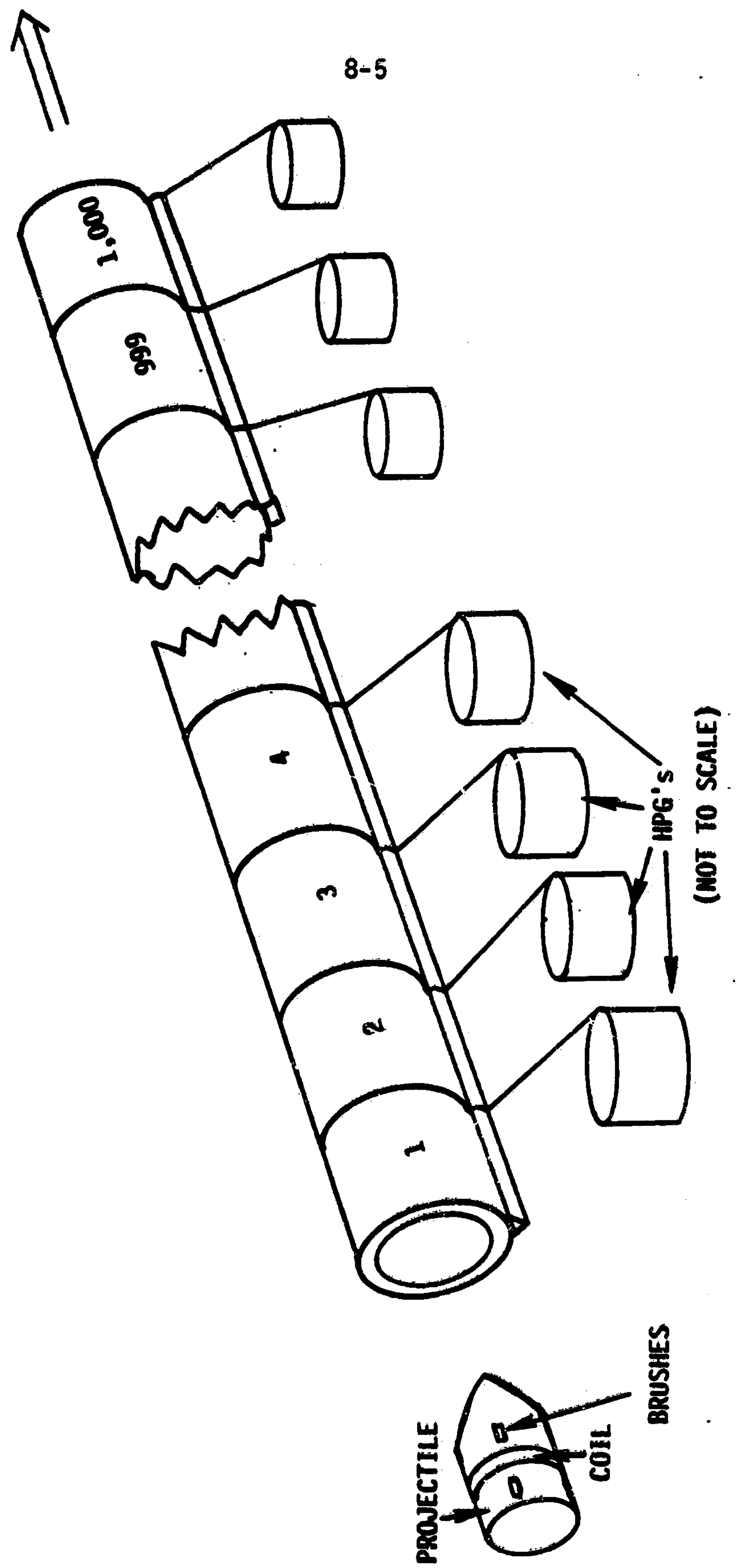


FIGURE 8-2. SOLENOID EARTH LAUNCHER

TABLE 8-2. PAYLOAD MASSES AS A FUNCTION OF STRUCTURAL MASS FOR 2000-KG PROJECTILE

Total Mass, kg	Projectile Coil Mass, kg	Propulsion System Mass, kg	Structural Mass, kg	Payload Mass, kg
2000	273	1238	0	489
2000	273	1095	200	432
2000	273	951	400	376
2000	273	808	600	319
2000	273	665	800	262
2000	273	521	1000	206

If a detailed system design for an Earth-to-orbit EML system is initiated, the Solenoid Earth Launcher should be considered along with those of the reference concepts (Section 4.0), especially if further research and experimentation yields promising results for use of brushes at high launch velocities.

8.2 Electrothermal Thrusters

In addition to the electromagnetic acceleration concepts covered later in this report, electrothermal means of producing accelerations were also investigated. The work summarized in this section was done by Wilbur, Mitchell, and Shaw of Colorado State University under NASA Grant NGR-06-002-112 (Wilbur, 1983).

8.2.1 Electrothermal Rocket

Figure 8-3 shows a simple schematic of an electrothermal rocket concept. In this concept, an on-board propellant with a high I_{sp} (such as hydrogen) is heated electrically, then exhausted through a nozzle in order to produce thrust. Heating of the propellant could be accomplished by several different processes. In one concept, the resistojet, propellant would be heated by being passed over current-carrying resistive elements. Exhaust velocities of up to 8,500 m/s have been reported using this method, which is limited by material properties of the heating elements. In a second concept, heating of the propellant is accomplished by passing electrical current directly through the propellant in the heating chamber. In this concept, labeled an arcjet thruster, exhaust velocities of up to 15 km/s have been claimed. In both the resistojet and the arcjet rockets, propellant is carried aboard the vehicle and is metered into the heating chamber.

8.2.2 Electrothermal Ramjet

A concept in which no propellant is carried on the vehicle is shown schematically in Figure 8-4. This concept, an electrothermal ramjet, requires that the propellant be distributed within a sealed launch tube. The body of the ramjet could be fabricated from the payload material reinforced, as necessary, by other structural materials. The ramjet engine, moving through the dispersed propellant, would gather the propellant in through the inlet diffuser, heat it electrically, and provide thrust by expelling it through the nozzle. Energy could be conducted into the heating chamber either electromagnetically or through pulsed electrical discharges occurring sequentially throughout the length of the launcher tube wall. A frangible cover would be needed to contain propellant within the tube and permit the ramjet to escape the tube with minimum loss of energy. Other aspects of ramjet operation are discussed following presentation of an annular flow ramjet configuration that might present fewer problems in fabrication and operation than the configuration shown in Figure 8-4.

8.2.3 Annular-Flow Electrothermal Ramjet

If the concept depicted in Figure 8-4 is modified so that the heating chamber is a space between a solid payload package and the launch tube wall, a configuration similar to that shown in Figure 8-5 results. This concept is similar to that of Figure 8-4 in that the launch tube is sealed and is filled with propellant. However, in the annular flow concept, the heat-addition region is not inside the payload; it is designed to be located between the moving payload and the stationary launch tube wall. One advantage this configuration presents is that there is no need for a "window" in the payload that would be required to permit passage of energy into the heating chamber as is required in the case of the concept shown in Figure 8-4. However, either of the concepts of Figure 8-4 and Figure 8-5 would need a first stage to accelerate a payload to the velocity needed for commencement of efficient ramjet operation. This acceleration could be produced by an electromagnetic driver (railgun or coaxial accelerator), a booster rocket, or a light-gas gun. This need for a hybrid launch mechanism using two technologies creates the potential for more problems than would use of a single technology for the launcher. Some of these possible problems are discussed later in this section.

8.2.4 Launcher Requirements

Wilbur (1983) presented the results for analysis of an Earth-escape mission for 10-kg payloads. Requirements were to provide sufficient thrust to constantly accelerate a payload of 10 kg at 30,000 g to a final velocity of 15,000 m/s. The acceleration and final velocity requirements yield a launch tube length of about 400 meters and a launch time of approximately 50 msec. The payload was assumed to have a shape similar to the payload shown in Figure 8-5 with maximum diameter being 16 cm at both diffuser and nozzle throats and minimum diameter at the

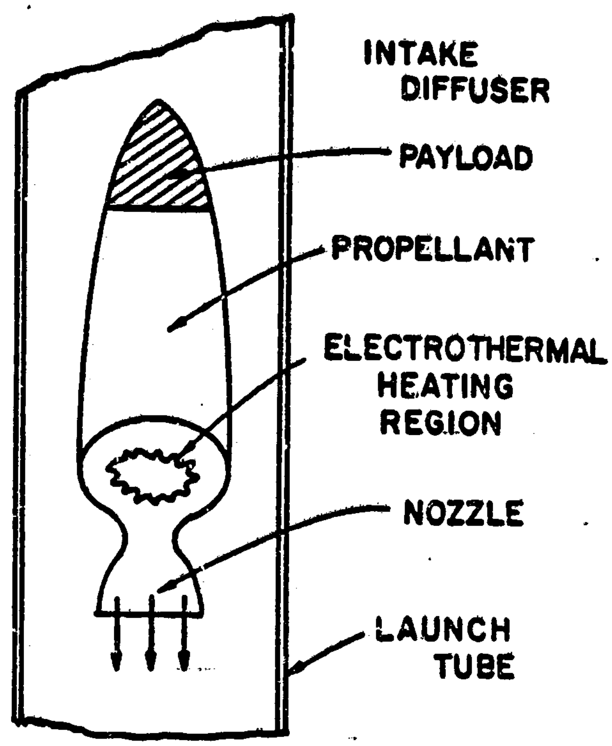


FIGURE 8-3. ELECTROTHERMAL ROCKET CONCEPT

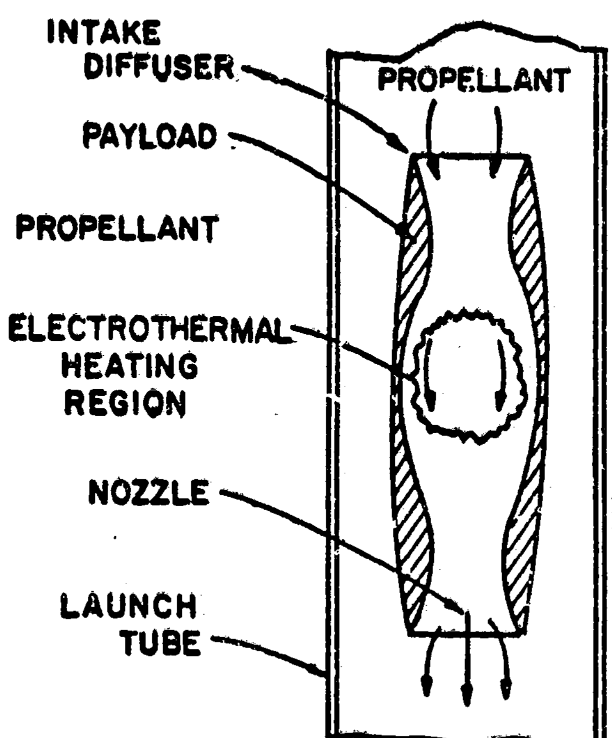


FIGURE 8-4. ELECTROTHERMAL RAMJET CONCEPT

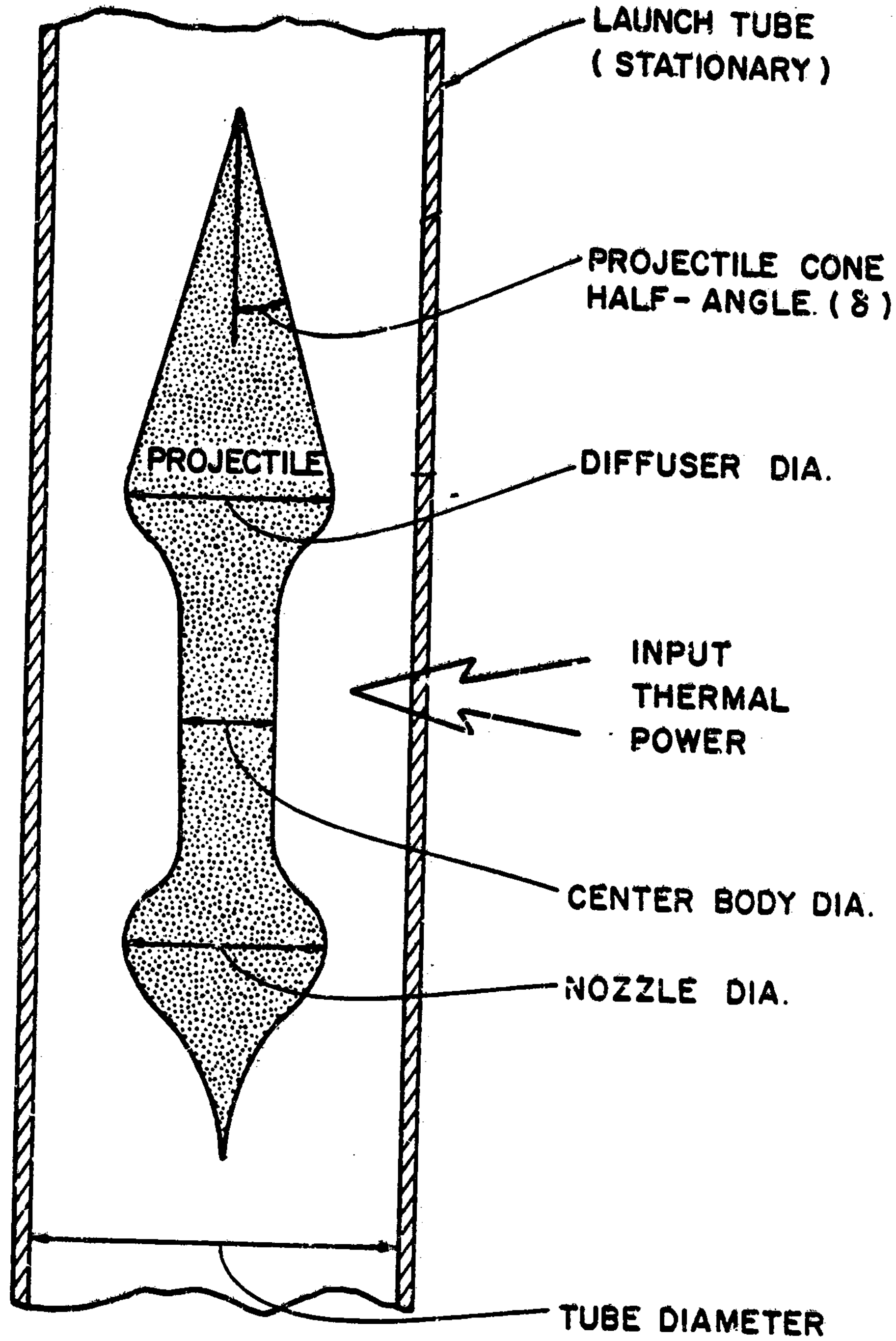


FIGURE 8-5. ANNULAR-FLOW ELECTROTHERMAL RAMJET

center of the payload being 8 cm to provide a heating chamber between the center portion of the payload and the launcher tube wall. The diffuser cone half-angle was chosen to be about 22.5 degrees to permit exhaust velocities to be as high as possible. The chosen propellant was hydrogen and was assumed to be in the launch tube initially at 300 degrees K and at a pressure of 30 atmospheres. At the end of the launch sequence, the exhaust temperature was calculated to be approximately 5,000 degrees K and the exhaust pressure, about 400 atmospheres.

The launch tube is required to be tapered as a function of the payload speed in order to permit efficient functioning of the ramjet along the length of the tube. The tapered profile of the launch tube needed for the hypothetical application is shown in Figure 8-6. This taper is needed so that a Mach number of M=1 at the nozzle throat can be maintained throughout the launch; this, in turn, leads to most efficient operation of the ramjet portion of the launcher. This requirement is based upon both thermodynamic and aerodynamic considerations.

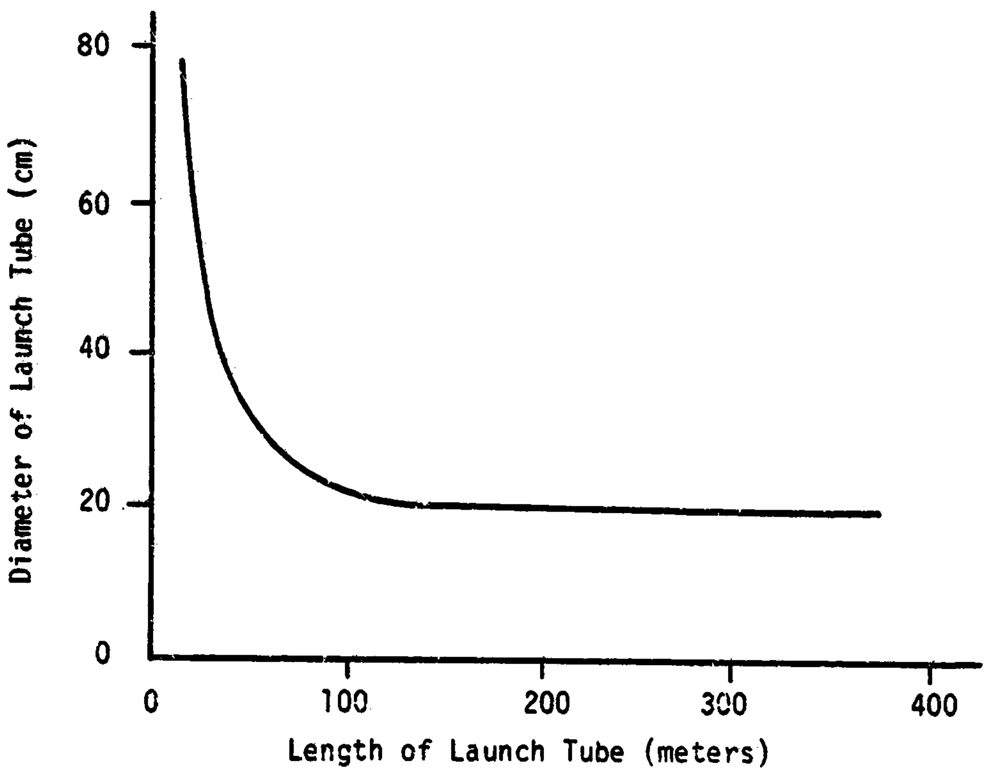


FIGURE 8-6. PROFILE OF LAUNCH TUBE

In addition to its tapered profile, the launch tube would have to be designed (1) to withstand the temperatures and pressures associated with the launch, (2) to have a high duty cycle with a correspondingly short turn-around time, and (3) to provide for transferring external energy to the internal, moving electrothermal heating region.

8.2.5 Energy Transfer

Figure 8-7 displays an instantaneous power profile of the thermal energy that must be added to the heat addition region of the electrothermal ramjet in order to sustain the design acceleration of 30,000 g until the speed of 15 km/s is reached at the end of the launcher tube. The total thermal energy input, found by integrating under the power curve, is about 2.4 GJ. Using information provided in Tables 3-3 and 3-4 of Rice, et al., 1982, and scaling to the same final velocity and payload size, gives about 1.6 GJ as the energy needed to perform an equivalent mission with a hypothetical railgun launcher.

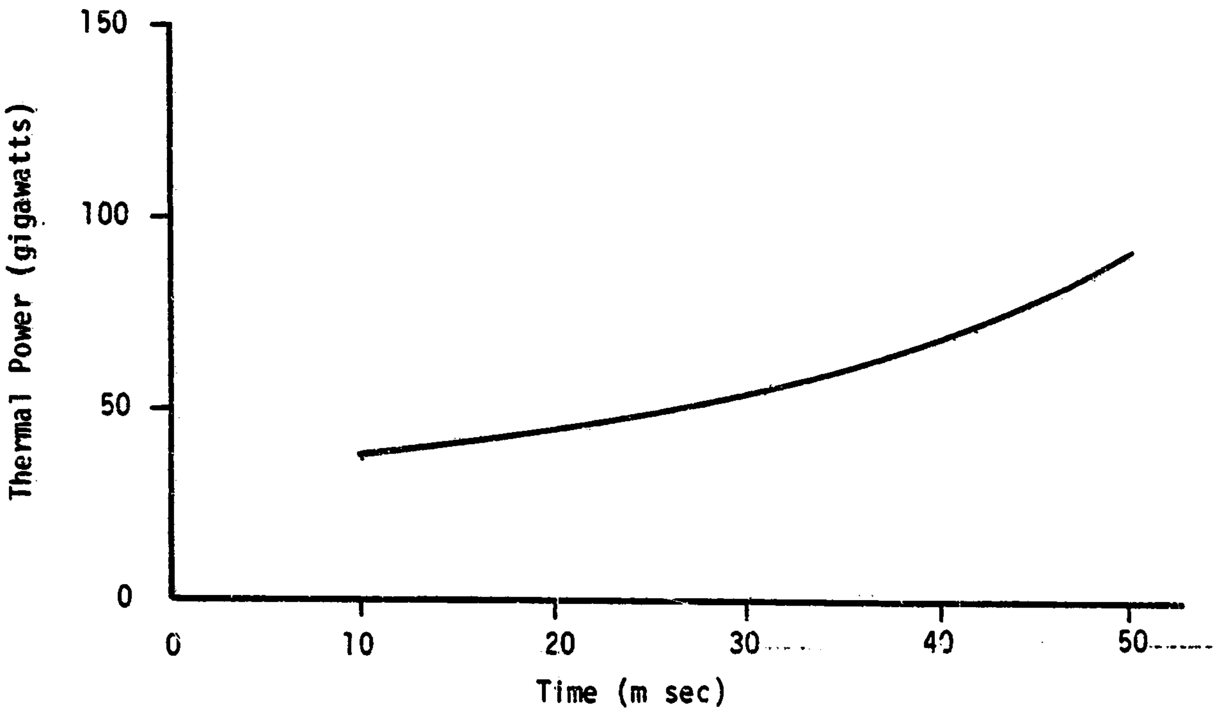


FIGURE 8-7 POWER PROFILE

A large amount of the difference between the energy requirements for the two concepts is the relative inefficiency of the electrothermal ramjet in converting thermal energy in the heat chamber to kinetic energy of the payload projectile. An instantaneous efficiency curve for the electrothermal ramjet is presented as Figure 8-8. Energy conversion

efficiencies of up to 85 percent have been postulated for railguns (Rice, et al, 1982). The electrothermal ramjet would require additional energy during the pre-boost phase of launch, making it even less attractive than a railgun alternative.

The preferred method of Wilbur (1983) for transferring energy into the thermal heating region is by means of lasers arranged outside the launcher tube wall projecting energy through windows in the wall in order to provide heating of the hydrogen propellant. When both quantum and thermodynamic efficiencies of the lasers are taken into consideration it is obvious that large penalties in terms of non-productive energy use would accrue to the laser-driven, electrothermal ramjet concept. In addition, electrical energy generation and storage capabilities similar to those described in Rice, et al, 1982, for a railgun launcher would be needed to drive the lasers.

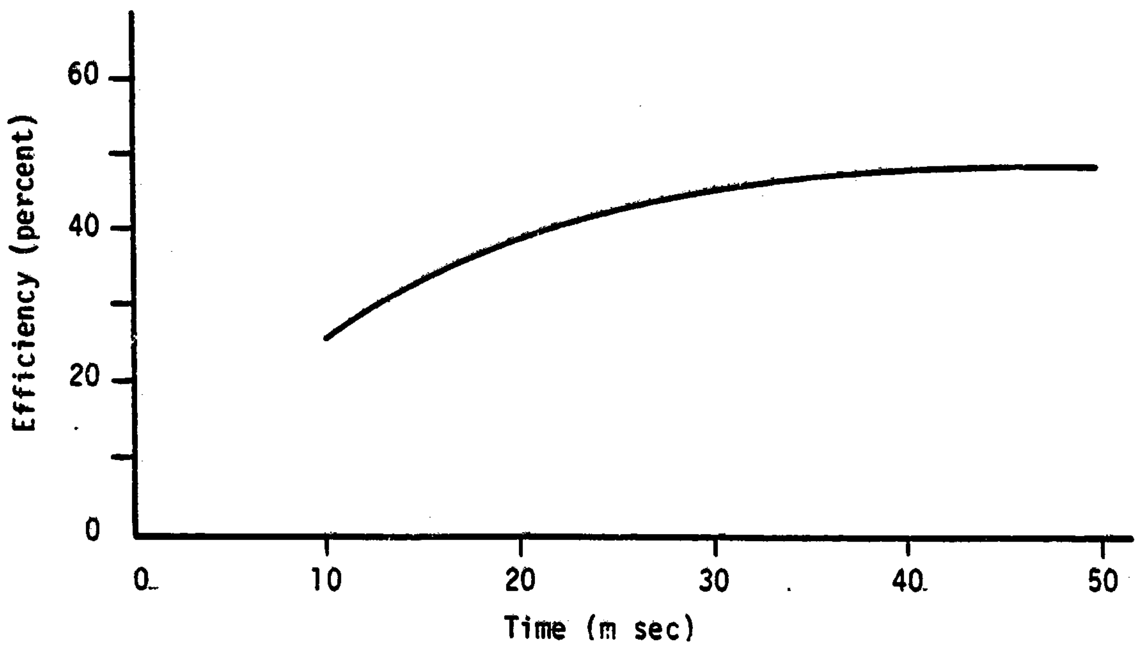


FIGURE 8-8. THERMAL-TO-MECHANICAL ENERGY EFFICIENCY PROFILE

An alternative method of supplying heating energy to the moving thermal heating region would be by means of a sequentially-fired series of electrical discharge mechanisms built into the inside of the launcher tube. This concept would have the advantage of transferring energy directly to the heating chamber without the intermediate (and inefficient) conversion to optical energy in the lasers. Wilbur (1983) did not present

any design or efficiency data on this proposed concept, but it would appear to have both initial and continuing cost advantages over the last concept unless it contains a fatal flaw that would disqualify it from consideration. This concept still has the major disadvantage that massive amounts of electrical energy would have to be stored, then transferred to the electrothermal heating region extremely rapidly and in a precision, rapid sequence. These requirements result in very high capital costs that have been discussed at some length for rail launcher concepts in Rice, et al (1982). However, there appears to be another concept, not mentioned in Wilbur (1983), that could be more attractive than either scheme discussed to this point.

Consider again the general annular ramjet configuration shown in Figure 8-5. However, instead of a pure hydrogen environment, assume that the tube is filled with a mixture of 90 to 94 percent hydrogen and 6 to 10 percent oxygen, a mixture that would sustain combustion in the presence of an igniter, but one that would not be explosive. Low-energy spark generators could be implanted in the inner walls of the launch tube and fired in sequence to maintain combustion in the moving heating region. The major advantage to this concept, which more closely resembles a pure ramjet, is that the high capital costs for electrical energy generation, storage, and conversion could be bypassed. Feasibility of this concept was not pursued because of project time and monetary constraints. Even if theoretical feasibility could be established, this concept shares a major problem with any other ramjet scheme; that is, the ramjet must be accelerated by other means to efficient operating velocities.

8.2.6 Summary

A ramjet starts to become an efficient mechanism only at velocities considered to be extremely high under most ordinary conditions. For example, in the electrothermal ramjet configuration shown in Figure 8-5 and discussed in Wilbur (1983), the authors assumed the need of a pre-boost phase to a speed of 5,000 m/s before ramjet action would be initiated. Even if this value is quite conservative, an initial stage is needed to produce some high value of velocity before efficient ramjet action can occur. This, in effect, leads to a hybrid launcher. The first stage could be a railgun or a light-gas gun. In either case, addtional complexity is introduced and stage interface problems would be encountered.

Additional areas that would have to be investigated in order to determine both feasibility and competitiveness of any ramjet concept include:

- Payload size limitations
- Repetition rate of launch
- Life of launcher tube
- Payload ablation both in the tube and in the atmosphere

- Tube reconditioning requirements between launches
- Structural and thermal compatibility between tube structural material and window material in those concepts including laser-drive heating.

8.3 Electromagnetic Rocket Gun

Dr. F. Winterberg has recently proposed and theoretically analyzed a concept for a high-velocity accelerator which he calls an electromagnetic rocket gun (Winterberg, 1983). This concept is presented schematically in Figure 8-9. The launch tube walls are made up of field coils that are sequenced to make up a traveling magnetic wave. The projectile consists of two parts; a payload section in front, and a conducting, cylindrical, metallic section in back. The hollow portion of the cylinder is filled with propellant (in Winterberg, 1983, the preferred propellant is solid hydrogen).

The magnetic field is controlled so that it has a variable magnitude with respect to the projectile as shown in Figure 8-10. The leading edge of the field includes currents in the thin walls of the back portion of the projectile. The heat generated by these currents vaporizes the propellant which then encounters a region in which the magnetic field is rapidly rising. The now gaseous propellant is heated, ionized, and compressed by the rising force of the magnetic field and transformed into a plasma. After reaching the point of maximum field strength, the hydrogen plasma is ejected from the magnetic field as a high-speed exhaust jet, imparting an accelerating force to the projectile. In effect, the magnetic field establishes a "nozzle" through which the ionized propellant moves to produce thrust.

Advantages claimed for the electromagnetic rocket gun are as follows:

- The exhaust jet can be many times as long as the projectile, permitting switching of the current necessary to establish and move the magnetic field to be done at a lower rate than is necessary in alternative electromagnetic launchers such as synchronous accelerators.
- The magnetic field can be utilized to center the projectile in the launch tube. This eliminates contact between the projectile and the side wall which has been a source of problems in other launchers such as railguns at higher velocities.
- Superconducting or ferromagnetic materials are not needed.

A large amount of effort would be required to determine whether the electromagnetic rocket could be effectively used for the applications considered in this study. As far as is known, no experimental work has been performed to validate the concept. Since there are several unique

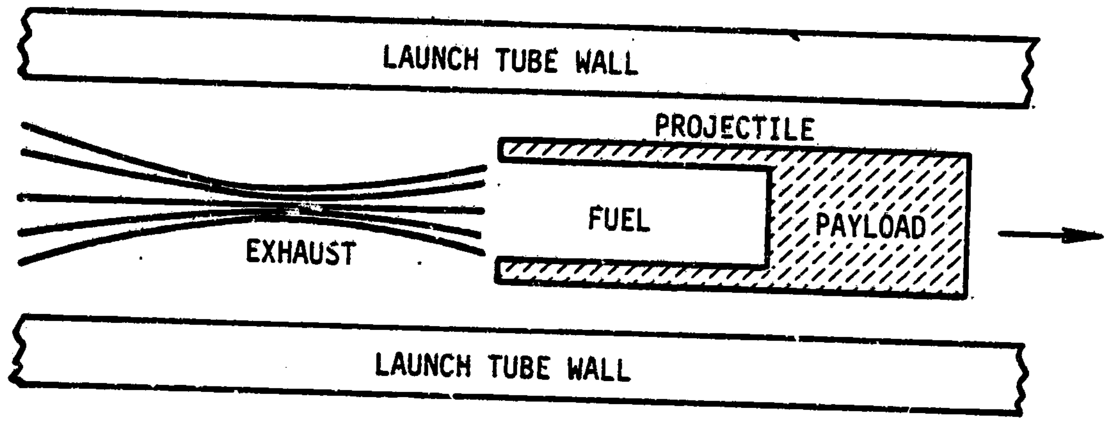


FIGURE 8-9. ELECTROMAGNETIC ROCKET GUN

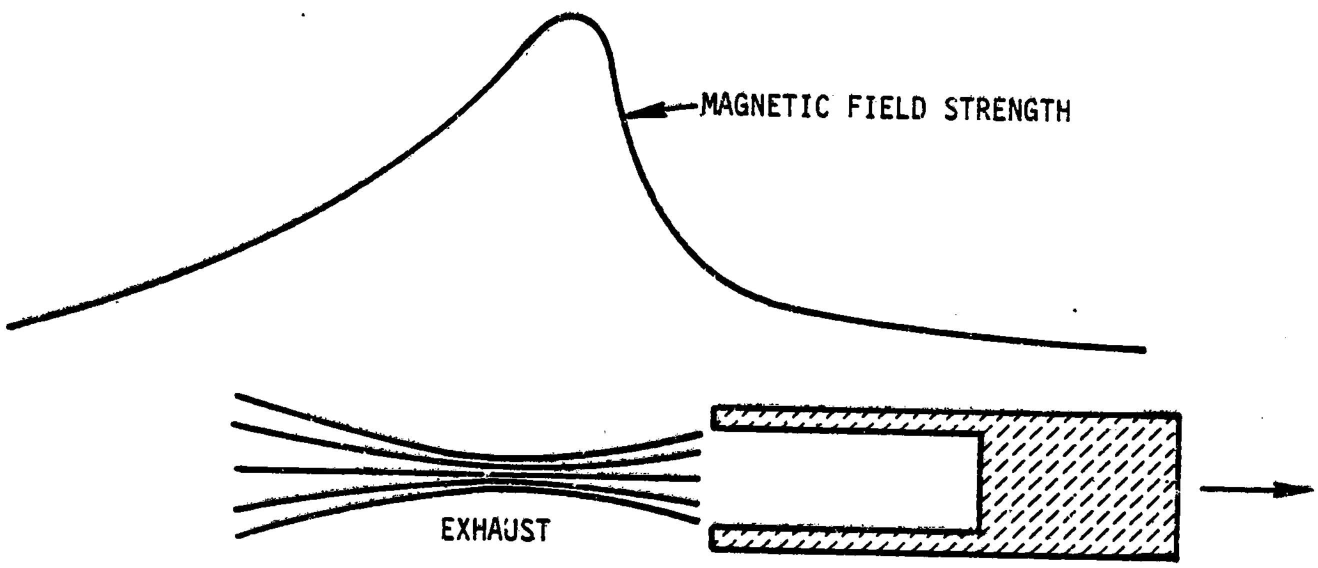


FIGURE 8-10. SHAPE OF TRAVELING MAGNETIC WAVE

features included in this concept (solid hydrogen as a propellant, electromagnetic nozzle) and since the author is concerned primarily with small size, ultra-high velocity (100 km/s) applications, consideration of the electromagnetic rocket gun was not pursued further.

8.4 Electromagnetic Theta Gun

The electromagnetic theta gun is a concept developed at Sandia National Laboratories, primarily for military applications (Burgess, 1980). The basic idea of this launcher is shown in Figure 8-11. The projectile is tubular in shape and is made of a highly conducting material. Driver coils are embedded in the cylindrical wall of the launcher and are energized sequentially as the projectile proceeds along the length of the launcher. Linear motion along the launcher tube is produced in a similar manner as rotational motion is produced in an induction motor. The magnetic field produced by the drive coils induces a current in the closed, conducting path of the projectile. This current, in turn, creates its own magnetic field which interacts mutually with the driver coil fields to produce a force in the desired direction. The projectile is guided along the launcher by a nonconducting mandrel extending the length of the launcher. If, as has been suggested (Burgess, 1980), the radial forces would tend to swage the projectile onto the mandrel, a slightly different configuration from that shown in Figure 8-11 could be designed. A second set of driver coils could be set into the mandrel, thus providing counteracting radial forces as well as adding to the axial forces producing acceleration along the launcher.

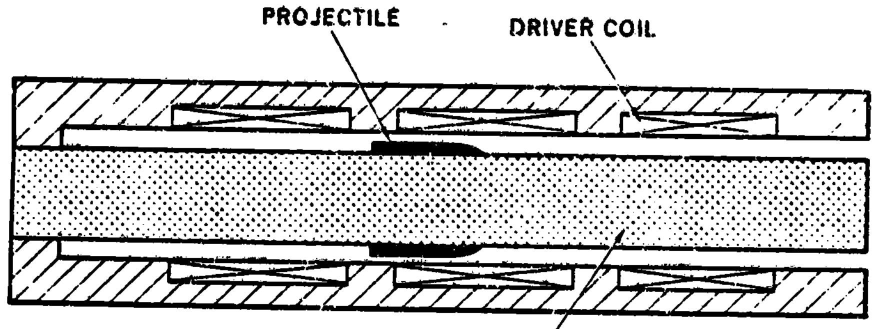


FIGURE 8-11 SCHEMATIC OF ELECTROMAGNETIC THETA GUN

The theta gun has several characteristics that render it unsuitable for the applications considered under this program:

- The cantilevered mandrel would present formidable design, manufacturing, and construction problems, since much larger launcher lengths are required than in the original theta-gun concept.

- The thin-wall-cylinder projectile configuration would be very inefficient for the types of payloads at which this study is aimed.

- The necessity for the projectile to be highly conductive represents a major problem, as well.

Thus, although the theta-gun concept may be promising for certain military applications, it does not warrant additional study for the space EML missions considered in this report.

8.5 Soviet Electromagnetic Accelerators

8.5.1 Induction Acceleration

A major thrust in electromagnetic acceleration work in the USSR is induction acceleration. Bondaletov has been the prime investigator in this area. The Soviet work consists of a one-coil inductor through which a current is pulsed. A conducting projectile (coil or ring) is accelerated by the magnetic field pulse from the adjacent inductor. Beryllium, copper, and aluminum are the preferred metals for projectile materials. The Soviets have reached velocities of 200 m/s for large projectile masses (200g). Emphasis on the acceleration of larger projectiles has been on industrial applications, including magnetic forming of metals and high-speed current switching devices (Golovin, 1982).

Work in the acceleration of smaller projectiles has emphasized performance. In 1976 velocities of 5 km/s were reached for projectile masses on the order of 1 gram. The Soviets have since reached 10.5 km/s.

8.5.2 Railguns

There is no evidence in the open literature of Soviet work in the railgun acceleration of solid projectiles. Golovin (1982) speculated that this is because the Soviets are waiting until other countries, the U.S. in particular, have published sufficient information before performing a comprehensive study of the technology and setting their own direction in the field.

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APPENDIX B

ACRONYMS AND ABBREVIATIONS

ACC	Aft Cargo Carrier
ACS	attitude control system
AMZIRC	copper-zirconium alloy
cc	cubic centimeter
CELSS	controlled ecological life support system
CFMF	Cryogenic Fluid Management Facility
cm	centimeter
COR	Contracting Officer's Representative
Cd	drag coefficient
Ci	Curie
DES	distributed energy storage
DOD	Department of Defense
DOE	Department of Energy
ECLSS	environmental control/life support system
EML	electromagnetic launcher
ESRL	Earth-to-space railgun launcher
ET	External Tank
EVA	extra-vehicular activity
g	gram
GEIS	Generic Environmental Impact Statement
GEO	geosynchronous orbit
GWe	gigawatt electric
HEPA	high-efficiency particulate filter
HLW	high-level waste
HPG	homopolar generator
IOC	initial operating capability
Isp	specific impulse
J	joule
JSC	Johnson Space Center
kg	kilogram
km	kilometer

KSC	Kennedy Space Center
LEO	low-Earth orbit
LLOX	lunar liquid oxygen
LOPER	Lunar Observation Pressurized Exploration Rig
LOX	liquid oxygen
LSS	life support system
LSTS	Lunar Space Transportation System
LeRC	Lewis Research Center
m	meter
MPS	Space Shuttle mail propulsion system
MSFC	Marshall Space Flight Center
MTHM	metric ton heavy metal
N	Newton
NASA	National Aeronautics and Space Administration
nmi	nautical mile
O/F	oxidizer-to-fuel ratio
OMV	Orbital Maneuvering Vehicle
OTV	Orbit Transfer Vehicle
PSS	payload support structure
rem	roentgen equivalent to man
RFC	regnerative fuel cell
SCAT	Surface Construction and Transport Vehicle
SCOUT	Surface Crawler Observer Unpressurized Transportation
SR&T	supporting research and technology
SS	single stage energy storage
SSEC	Solar System Exploration Committee
SSTO	single-stage-to-orbit vehicle
STS	Space Transportation System
TAV	Transatmospheric Vehicle
TPS	thermal protection system
TRU	transuranic waste
ULV	Unmanned Launch Vehicle
USAF	United States Air Force
USAF/ASD	United States Air Force/Advanced Systems Division

W	watt
WBS	work breakdown structure
ΔV	velocity increment

APPENDIX C

METRIC/ENGLISH UNIT CONVERSION FACTORS

To convert	into	multiply by
atmospheres (atm). . . .	pounds per square inch (psi). . .	14.70
atmospheres (atm). . . .	pounds per square feet (psf). . .	2116.8
calories (cal)	British thermal units (Btu). . . .	3.9685×10^{-3}
Calories per gram (cal/g).	British thermal units per pound (Btu/lb).	1.80
centimeters (cm)	inches (in)	0.3937
centimeters (cm)	feet (ft)	3.281×10^{-2}
centimeters (cm)	yards (yd).	1.094×10^{-2}
cubic centimeters (cm^3).	cubic inches (in^3).	0.0610
cubic meters (m^3). . . .	gallons (gal)	264.2
degrees Centigrade (°C).	degrees Fahrenheit (°F)	1.8 C + 32*
degrees Kelvin (°K). . .	degrees Rankine (°R).	1.8
grams (g).	pounds (lb)	2.205×10^{-3}
kilograms (kg)	pounds (lb)	2.205
kilometers (km).	statute miles (mi).	0.6214
kilometers (km).	nautical miles (n.mi.).	0.540
kilometers (km).	feet (ft)	3281
kilowatts (kW)	Btu per hour (Btu/hr)	3413
meters (m)	inches (in)	39.37
meters (m)	feet (ft)	3.281
meters (m)	yards (yd).	1.094
meters per second (m/s).	feet per second (ft/s).	3.281
metric tons (MT)	pounds (lb)	2205
metric tons (MT)	tons (T).	1.102
micrometers (μm)	meters (m).	1.0×10^{-6}
Newtons (N).	pounds force (lb_f).	0.2248
Newtons per cm^2 (N/cm^2).	pounds per square inch (psi). . .	1.4504

ASSESSMENT OF COAXIAL LAUNCHER TECHNOLOGY FOR EARTH-TO-SPACE AND ORBITAL TRANSFER MISSIONS

Report of a study performed for:

NASA Lewis Research Center

under subcontract to

Battelle Columbus Laboratories

Mr. William Kerslake
Technical Monitor

Mr. Eric Rice
Project Manager

April 1983

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APPENDIX D

TABLE OF CONTENTS

SECTION 1. EXECUTIVE SUMMARY

Electromagnetic space launch catapults have occupied the interest of science fiction writers for several decades, but have received serious attention from technologists only since 1977, when the possibility was considered at the NASA-AMES Summer Study on Space Industrialization. Ablation and energy loss in transitioning the atmosphere were found to be quite reasonable, and pulsed power technology was found to have reached the required degree of sophistication.

Shortly after the 1977 study, a DoD program in EML research was implemented by our group, then at the M.I.T. National Magnet Laboratory, on the initiative of Dr. Harry Fair, then Director of Propulsion Research at ARRADCOM-DOVER, and presently at the Defense Advanced Projects Agency (DARPA). As a result of research done since 1977, a number of important defense applications have emerged, and electromagnetic launching is generally recognized as an emerging technology with a wide range of applications.

The use of electromagnetic catapults for the launching of space vehicles is inevitable, since it is certainly capable of improving launch-mass/payload ratios from those achievable with rockets (137 for Ariane) to near unity. Electromagnetic catapults may initially replace only the first stage booster, and ultimately they may perform the entire launch. The crucial economic trade-off will always be a balance between the cost of expendable launch mass and re-useable ground-based support mass.

The electric energy required to launch payload to earth escape velocity only costs 65 cents per pound. To launch a one-ton vehicle using a one-mile long launcher would require storing the total output of a large 1,000 megawatt municipal power plant for one-and-a-half minutes, and delivering it to the launcher in one-and-a-half seconds. The launcher itself is not expensive, but the energy storage facility is very expensive.

Electromagnetic launch cost is dominated neither by energy cost nor by launcher cost, but primarily by the cost of energy storage equipment. Recognition of this circumstance is very important, and has several immediate implications.

Technologically, it implies that large storage inductors represent the most serious contender as the primary energy store, because an inductor large enough to store the energy for an entire launch is not only relatively inexpensive, but will automatically have an L/R time constant long enough to permit several minutes of charging-time, thereby reducing the prime power requirement to the level of existing powerlines or available generators. It follows of course that economies will strongly favor those launchers which are capable of deriving their energy from a single inductor.

ORIGINAL PAGE IS OF POOR QUALITY

Economically, domination by energy storage equipment cost implies the importance of ensuring a high use-rate of the energy supply facility. It suggests highly flexible multi-mission installations in which a single energy supply can feed several launchers to share its available time among missions with different launch window limitations. It also suggests minimum size vehicles launched at the highest repetition rate possible.

Several species of launcher are available. Evolution of a new technology always involves a proliferation of species, and it may be impossible to identify the fittest in advance. In aviation, the dirigible was profitable in transatlantic passenger service long before the aeroplane was proven practical, let alone superior, and it took another half- century for the winning technology mix to emerge victorious in the marketplace or the battlefield. The past thirty years of rocketry involved a comparable evolution. We have learned enough to recognize the folly of monolithic planning. Electromagnetic launch technology will ultimately involve a complex combination of techniques, as is the case in aviation and rocketry. Two basic approaches can be categorized at present:

Railguns offer

- simplicity of the launcher structure, at the cost of a more complex projectile with a contacting sabot,
- higher launch velocity capability, but only for smaller launch vehicles,
- limited flexibility in design and power supply,
- low efficiency and service life

Coaxial launchers offer

- simple non-contacting projectiles without sabot or significant structural mass, at the cost of higher launcher complexity and cost,
- flexibility in design and power supply,
- relatively lower launch velocity, but no limit to projectile size,
- high energy-efficiency and virtually unlimited service life.

A critical evaluation of the available options in coaxial launcher design, based on sophisticated trade-off analysis, indicates the winning approach to be a compromise between two well developed concepts: the discrete-coil mass-driver, and the brush-commutated helical railgun. We might call it the "continuous-commutation pulsed-induction launcher".

D-2

The coaxial launcher selected as reference design uses the basic mechanism of original mass drivers, in which discrete drive coils are tuned to resonate at the local projectile velocity by the use of sector capacitors in order that commutation can be accomplished at or near a zero-crossing of the coil current by the use of solid state switches triggered by projectile positon sensors.

Instead of discrete coils, the selected launcher uses a continuous helical winding in which commutation is performed turn by turn, so that most of the commutation energy is stored in the winding itself, and only about one percent of it is stored in commutation capacitors. This approach has three advantages: it substantially reduces the number of capacitors, it reduces the switching load for each commutation, and it maintains the inductance gradient near its peak value.

The projectile coils are a series of solid copper or aluminum rings spaced approximately one radius apart so as to distribute the thrust. This permits the use of slender projectiles without the need for substantial structural reinforcement or for a sabot.

Detailed mission analysis suggests that the most logical first step is the construction of a hybrid electro-chemical launcher in which an electric first stage of the above design supplies a launch velocity of about 3 km/s, the velocity at which rockets operate at peak efficiency. This is the only way in which EML technology can overlap existing technology in the near future.

This hybrid approach would reduce launch mass to about 20 percent of present launch mass for a given payload. Furthermore, the proposed electromagnetic launch velocity of 3 km/s is achievable with present technology.

A hybrid launcher of this type can be seriously proposed for immediate construction. It is demonstrably feasible and cost-effective.

It is concluded that the future of our space enterprise depends on electromagnetic launch technology. EML Research Inc. is dedicated to the implementation of this technology.

SECTION 2. BACKGROUND

It has become possible recently to store and release electrical energy in megajoule quantities. The impact of this art may be as revolutionary as earlier innovations which represent the milestones of pulsed power technology: the mechanical catapult, gunpowder, and chemical rockets.

Electromagnetic launch technology is destined to have a particularly significant effect on the launching of space cargo. Launch velocity requirements exceed the capability of chemical guns, and it has therefore been necessary to resort to chemical rockets, which impose a very severe dead-weight penalty. Typical launch mass to payload ratios range from 95 (Scout) to 137 (Ariane). The dead-weight, moreover, is mostly expensive and usually unrecoverable hardware.

Electromagnetic launchers offer the possibility of doing what chemical guns are unable to do: launching almost pure payload, at an ultimate cost approaching the cost of the energy. The cost of the energy to reach earth escape is only 65 cents per pound (at 1981 electric power rates). The added overhead of operation and amortization of the launcher is what the present study has attempted to assess.

D-3

Electromagnetic launchers offer these unique advantages over chemical propellants:

* Launch velocity is limited in principle only by the velocity of light, and in practice by high voltage switching technology to the order of 20 km/s.
* Available forces are an order of magnitude higher than from chemical explosives.
* Launch forces can be applied to the body, rather than the rear surface of the vehicle.
* The launch force can be monitored and controlled during the launch.
* Redundancy of critical launcher components can be provided, making the launch cycle as fail-safe as requirements dictate and economics permit.
* Launches can be repeated within minutes, without the need for repair of destruction.

Electromagnetic launch technology really does seem destined to dominate the space enterprise of the coming era.

2.1 BRIEF HISTORY OF E M LAUNCH TECHNOLOGY

Electromagnetic space vehicle launchers have been a dream of science fiction writers for many decades (Arthur C. Clarke, Robert Heinlein, Willy Ley), and they have even been predicted by an early worker in the field (Northrup, 1930) but the concept has only recently been taken seriously by technologists. At a NASA-AMES summer study on space manufacturing in 1979 it was found possible to launch projectiles as small as one ton through the atmosphere at earth escape velocity with only minor losses in mass and energy. A 1982 report by Battelle, sponsored by NASA-LEWIS, finds that it is economically feasible to dispose of nuclear waste by launching it electromagnetically ~~into solar orbit~~ or out of the solar system.

Terrestrial electromagnetic launchers have been explored sporadically for various purposes, but until recently the results have never justified a serious effort. Notable early efforts include the following:

In the thirties, Professor Northrup of Princeton, founder of Leeds Northrup Co., built a discrete-coil, mechanically synchronized launcher, which he considered the ultimate means to implement space travel. He predicted a race to the moon between USA and USSR in the seventies.

During WW-2 the Germans built a linear induction launcher which melted its copper-jacketed projectiles before they reached launch velocity, and was abandoned in favor of the V-2 rockets.

In the late forties Westinghouse, with Navy support, built an electric aircraft launcher intended for use on small islands, called the "Electropult". The device was an iron-based linear dc motor, impractical because of its enormous mass of iron, both moving and stationary.

Thus, the ordinary induction motor as well as the ordinary dc motor are not applicable to high-thrust generation.

In the seventies, Professor Winterberg of the University of Nevada proposed a travelling wave accelerator using a superconducting coil, but the process proved longitudinally unstable, i.e., not able to maintain synchronization.

In 1972, Thom and Norwood of NASA-LANGLEY proposed a brush-commutated helical accelerator, which was subsequently re-invented by the MIT group in 1979, and reduced to practice on a larger scale, in the form of a five inch caliber launcher for ten pound models of a re-supply and reconaissance glider.

In the early seventies, a group at the Australian National University in Canberra headed by Richard Marshall used the world's largest homopolar generator to drive a railgun which accelerated a six gram projectile to 5 km/s in a five meter length.

In 1978, Bondaletov of the USSR reported accelerating 2 gram aluminum rings to 5 km/s in a length of 1 cm, with a peak acceleration of 200 million gee, by a pulsed induction process, using a 60 kV capacitor bank.

In 1976, Kolm and O'Neill at MIT developed the "mass driver", a discrete coil, solid-state synchronized launcher, originally intended for launching lunar ore to construction sites in earth orbit. Mass drivers were a prominent part of the subject matter at two NASA-AMES summer studies on Space Colonization, in 1976 and 1977. The second study also dealt with the feasibility of earth-based launching and a calculation of ablation and energy losses in traversing the atmosphere, by Chul Park and Stuart Bowen of AMES. Their subsequent publication of this calculation represents the first serious consideration of earth-based launching by catapults, as far as we know.

In 1977 Dr.Harry Fair, then chief of propulsion research at ARRADCOM, commissioned Kolm to initiate the first research effort into potential DoD applications of electromagnetic launch technology, at the MIT National Magnet Laboratory. At the same time Fair and Kolm organized a DoD working group and a civilian technical advisory panel to plan and implement a national research program in electromagnetic launch technology, managed by the ARRADCOM group. Four contracts for 6.1 category research hardware were awarded to:

(1) WESTINGHOUSE: for construction of a homopolar-powered railgun to fire 300 gram projectiles at 3 km/s, using an intermediate toroidal storage inductor designed under subcontract by the EML group. This system forms the nucleus of a new EML research laboratory at ARRADCOM, Picatinny Arsenal, Dover NJ.

(2) The CENTER FOR ELECTROMECHANICAL RESEARCH at the University of Texas, Austin: for the development of compact, light-weight, second-generation homopolar generators.

(3) The ELECTROMAGNETIC ACCELERATOR GROUP at the Francis Bitter National Magnet Laboratory at M.I.T.: for a basic study of coaxial launchers, and for development of the first brush-commutated helical railgun launcher for RPV-type vehicles and projectiles.

(4) LOS ALAMOS and LAWRENCE-LIVERMORE: for testing a small railgun powered by an explosively driven flux compression generator.

At the same time, Dr. Fair initiated the construction of an electromagnetic launch research facility at the ARRADCOM laboratory, adding Westinghouse hardware and MIT graduate students to the existing chemical propulsion research group.

Dr. Fair was transferred in June 1981 from ARRADCOM to DARPA, where he is acting head of the Tactical Technology Office (TTO). He was named executive manager of all DoD research in electromagnetic launch technology.

One open national symposium, in San Diego October 1980, and a number of invitational workshop meetings have been held in the electromagnetic launch field. The second such symposium is planned for Boston in October 1983. During 1981-2 there have been several hardware development contracts awarded by the Air Force and Army, and an expanding research program is planned for the coming years.

2.2 STATE OF THE ART

The state of the EM launcher art is summarized in a recent review article, a copy of which is appended:

"An Alternative Launching Medium"
by H. Kolm and P. Mongeau
IEEE Spectrum, April 1982

The next thorough review of the art will be prepared by EML Research Inc. on the basis of the papers to be presented at the EM Launch Technology Symposium to be held in Boston October 10 to 13, 1983.

Railgun technology has advanced to a demonstrated capability of about 5 km/s for kilogram-size vehicles, and 10 km/s for gram-size vehicles, with overall energy conversion efficiencies of the order of 30%.

Scale-up in both projectile mass and velocity requires refinement of railguns and energy supplies. Several groups are working on multi-stage, distributed-energy railguns, and studies are also in progress in alternative energy supply systems. Velocities in the 20 km/s range are clearly feasible, but projectile mass limits at that velocity are unclear at present.

Coaxial launcher technology has reached a demonstrated capability of 5 km/s in the gram range, but only 1 km/s in the kilogram range, mainly because it has received very little serious attention. Ours is the only group working in the field at present. There is virtually no fundamental limit to scale-up in projectile mass, but scale-up in launch velocity is limited by switching capability at high voltage. A compensating advantage to the switching problems is the fact that efficiencies of the order of 90% are possible in coaxial launcher technology at a wide range of launch velocities.

2.3 CRUCIAL TECHNOLOGY BOTTLENECKS

Linear electric motors, like their rotary counterparts, develop a back-emf which is proportional to speed. The voltage at which energy is supplied to any type of launcher must increase with speed not only to buck this rising back-voltge, but also to make the rate of current-rise in the inductance of the launcher match the increasing speed. Thus, the supply voltage must increase roughly as the square of speed. At 10 km/s, the driving voltage for any railgun or coaxial accelerator must be in the 100 kV range. At 20 km/s it is likely to be in the 400 kV range.

Energy at such voltage levels can be supplied by capacitors, or by inductors acting as constant current sources, at least in principle. In practice, however, there are limits imposed by materials and switching devices. The most crucial technology bottleneck involves coils (both drive coils and storage inductors) with insulation capable of withstanding very high forces and very high voltages. Circuit breakers represent another bottleneck component; they are needed to switch railguns into storage inductor circuits, or to trigger the drive coils of coaxial launchers.

A number of switching devices can handle the closure function at hundreds of kilovolts: triggered gas gaps, mercury vapor ignitrons, solid state break-down switches. However, the only devices which seem capable of breaking circuits at the required voltages and speeds are high vacuum metal vapor arc gaps. This technology has never been pushed to the required levels. Our group is engaged in a study, funded by NASA-LEWIS, to determine whether high-vacuum metal vapor arc switches can be made to operate in the range of hundreds of kilovolts, but it is too early to predict their ultimate limits.

Primary energy storage for any type of launcher will be rotating machinery (homopolar or alternator generator) followed by storage inductors. This does not require extrapolation of existing technology, and represents primarily an economic challenge. The rotary storage might be either by dedicated machinery, or by relying on existing capacity (spinning reserve) in a utility power plant. It is interesting to note that a typical large municipal power plant of 1,000 kw capacity generates enough power to launch one ton into low earth orbit every 1.5 minutes. With an 8 km long launcher the launch duration is about 1.5 second. One-ton vehicles could thus be launched using existing generators with a power compression ratio of about 60.

Secondary storage (or second stage power compression) presents somewhat more of a problem. Inductors will serve to energize railguns, but capacitors are the only storage medium capable of releasing energy fast enough to drive the individual coils of a discrete coil coaxial launcher. However, capacitors are too bulky and expensive by a factor of 100 to store the entire launch energy. The solution is to use capacitors several hundred times during the launch cycle, recharging them from an intermediate storage medium such as inductors. Such a system has been implemented on a laboratory scale at Princeton, using electrolytic capacitors for intermediate storage. Full-scale implementation represents an extrapolation which remains to be developed, and therefore represents another technology bottleneck.

SECTION 3. BASIC PRINCIPLES OF COAXIAL LAUNCH TECHNOLOGY

Intelligent assessment of electromagnetic launch technology requires a certain familiarity with the basic engineering concepts, alternatives, and trade-offs involved. It is almost certain that when earth-based launchers are built, they will incorporate several stages, each using a different technology. The present section is intended to supply an overview for the benefit of non-experts.

3.1 INDUCTANCE GRADIENT

The most important single performance parameter of any electromagnetic accelerator is the inductance gradient, or the change in circuit inductance with every meter of projectile travel. Since the energy stored in the circuit is the product of inductance times current squared, the thrust applied to the projectile is the product of this inductance gradient times the current squared. Inductance gradient is thus a figure of merit, indicating how much thrust is generated for a given current squared.

For railguns this inductance gradient is simply one half of the rail circuit inductance per unit length of rail, and it is constant throughout the launch. Thrust for railguns is given by:

$$\text{thrust} = F = \frac{1}{2} L' I^2 \qquad (3.1)$$

where F = thrust force in Newtons,
I = railgun current in amperes, and
L' = dL/dz, railgun inductance in henry/meter

L' has a typical value of 0.5 microhenry/meter for any railgun of reasonable aspect ratio (gap-to-railheight). Thus, the figure of merit for railguns of any size and shape is essentially 0.25.

In the case of coaxial launchers, the inductance gradient is the change in mutual inductance as the projectile coil moves with respect to the stationary drive coil. The thrust in this case is given by

$$\text{thrust} = F = \frac{dM}{dz} I_p I_d \qquad (3.2)$$

where: subscripts p and d denote projectile and drive coil currents, and

dM/dz is the mutual inductance change per meter of projectile coil motion, in henry/meter

In coaxial accelerators the mutual inductance gradient can have peak values as high as 30 microhenry, or a figure of merit 100 times higher than for railguns (note the absence of the factor of 1/2). The catch is that the inductance gradient in the case of coaxial accelerators is not constant as in the case of railguns, but varies from positive to negative values as the projectile coil moves through each drive coil. In coaxial accelerators it is therefore necessary to synchronize one or both of the currents in the form of pulses timed to occur when the projectile coil passes through the region where the current will produce a thrust. It is therefore important to examine the variation of inductance gradient as the projectile moves along the accelerator.

3.2 INDUCTANCE LENGTH

Fig. 3.1 (next page) defines all the crucial parameters of coaxial accelerators. It shows a cross section of one drive coil and one projectile coil, and the mutual inductance M between them as a function of the projectile coil position.

It is useful to think of inductance in terms of its fundamental definition: the self-inductance of one coil is simply the flux linkage per unit current, and mutual inductance between two coils is flux linkage in one coil produced by one ampere in the other coil. As expected, the mutual inductance M is maximal at the point where the projectile coil passes through the plane of the drive coil. This is the point of maximum inductive coupling.

Fig. 3.1 also shows the mutual inductance gradient dM/dz, which is simply the slope of the inductance curve M. The dM/dz curve has two peaks at the inflection points of the M-curve, one positive and one negative. The projectile curve is shown located at one of these peaks. The distance between each peak and the central plane of the drive coil is called the "inductance length" l_m .

The inductance length is a very important parameter because it defines the distance over which acceleration can be produced, and therefore the time-period during which current must flow, or the necessary pulse duration, at a given speed. For practical dimensions, the inductance length is about equal to the radius of the drive coil. This circumstance defines the most important scaling factor for coaxial accelerators: it relates pulse duration to speed and caliber. Pulse duration (half-period of an oscillation) corresponds to the time required for the projectile to traverse a distance equal to the drive coil caliber. Thus, the period of the drive current oscillation corresponds to twice the caliber:

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$$T \simeq \frac{4R_d}{v} \qquad (3.3)$$

where: T = oscillation period in seconds

R_d = drive coil radius in meters, and

v = local projectile velocity in m/s

For intuitive reference: a one meter (3 foot) caliber launcher which achieves 20 km/s must be driven with an oscillating current which accelerates to a period of 0.1 millisecond, or a frequency of 10 kHz by the time the projectile leaves the muzzle.

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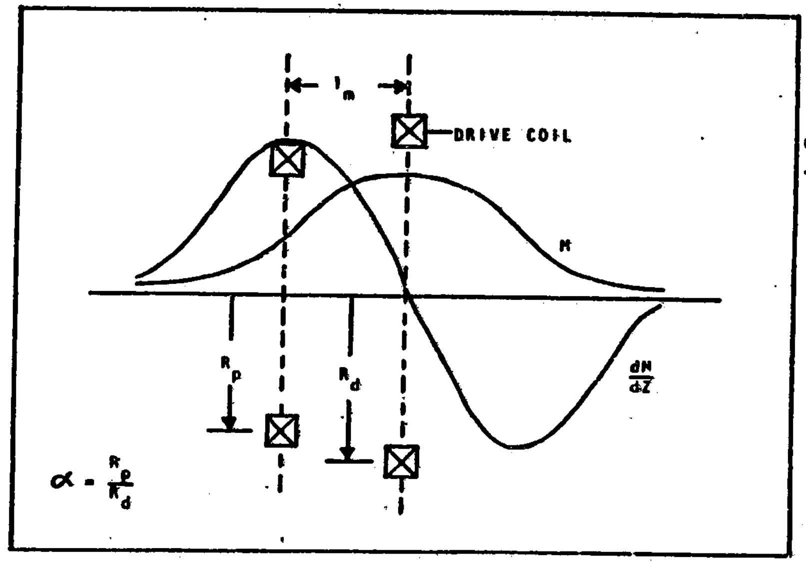
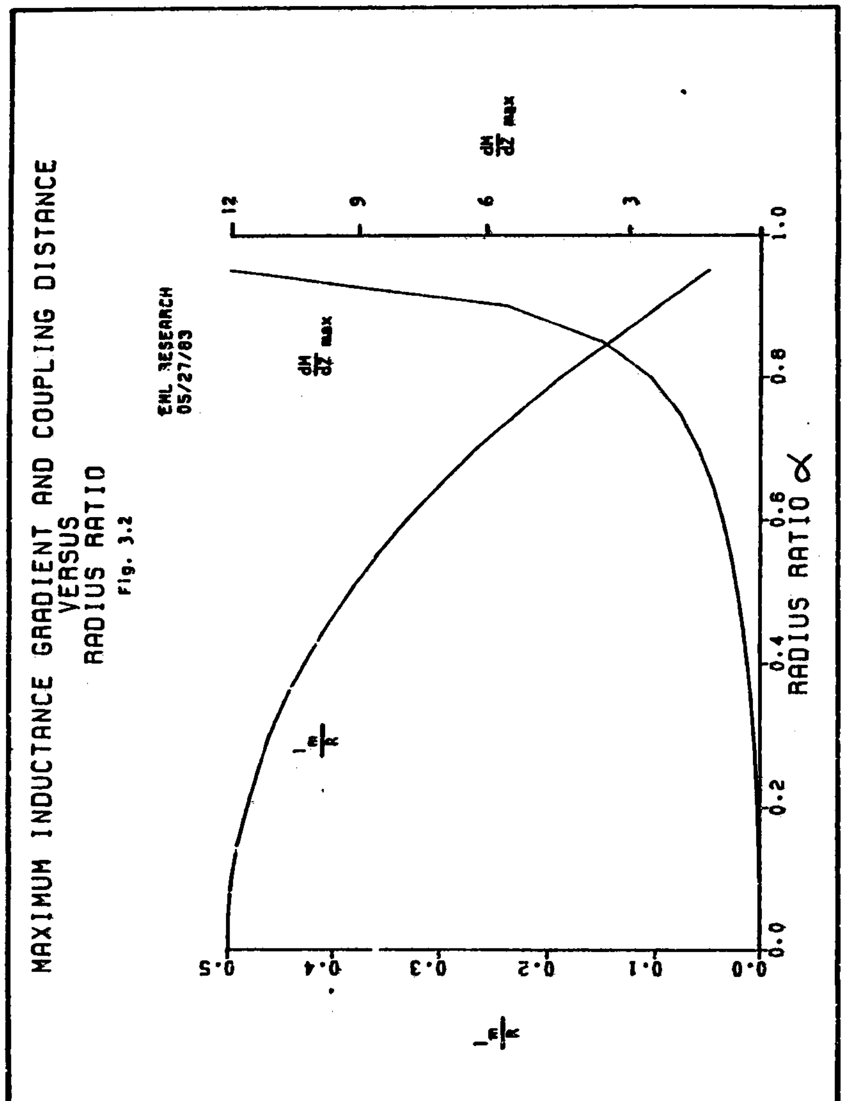


Fig. 3.1 Crucial parameters of coaxial launchers



MAXIMUM INDUCTANCE GRADIENT AND COUPLING DISTANCE VERSUS RADIUS RATIO

Fig. 3.2

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3.4 EXCITATION AND COUPLING

The mutual inductance gradient changes sign as the projectile coil passes through the drive coil. Therefore, if the projectile current is a constant excitation current supplied by sliding brushes, for instance, a sinusoidal drive coil current synchronized with the projectile will produce two pulses of thrust per drive coil: a pull as the projectile coil approaches, and a push as it departs. Push-pull operation of this nature was achieved in the first mass driver designs which were based on superconducting projectile coils carrying a persistent excitation current.

Superconducting projectile coils are not practical except for very large projectiles. Sliding contacts are the most efficient means, but only work up to a certain as yet undertermined speed, probably in the vicinity of 1 km/s. How can coaxial launchers be supplied with projectile current at speeds to the order ofg 10 km/s?

Fortunately there is another alternative: At speeds too high for brush contacts the projectile coils can be made in the form of a series of simple rings, and the excitation current can be induced by the drive pulse itself. This is possible because the L/R time constant of solid copper or aluminum rings is about 20 milliseconds, which corresponds to about 50 Hz. Thus the induced excitation current will serve for several acceleration pulses before it decays, and a new current needs to be induced. Since the induced current always opposes the inducing current, the induced thrust will only produce push. The pull half of the thrust cycle is thus sacrificed, and the coaxial launcher becomes a pulsed induction accelerator. In this mode, the induced projectile coil current is given by

$$I_p = -\frac{M}{L_p} I_d \qquad (3.4)$$

and the resultant thrust is therefore:

$$\text{THRUST} = F = -I_d^2 \frac{dM}{dz}\left(\frac{M}{L_p}\right) \qquad (3.5)$$

where the subscripts p and d again denote projectile and drive coils, and where L and M again denote self and mutual inductance respectively. Optimization for the pulsed induction operating mode will clearly lead to different configurations, since M and L now become significant parameters, contrary to the push-pull mode in which only dM/dz appeared in the expression for thrust force.

Clearly, the best inductive coupling between projectile and drive coils will be achieved when both coils approach a single current line of almost the same radius, i.e., when the build, length, and radial clearance approach zero. Coupling is characterized by the ratio alpha of the radii of circles representing the equivalent current loop of projectile and drive coils (assuming thin coils), defined in Fig. 3.1.

Fig. 3.2 shows the dependence of the inductance length and the peak mutual inductance gradient on this coupling ratio alpha. As alpha approaches unity (perfect coupling), the gradient peak increases, but the inductance length decreases. Thus a given current will produce a higher peak thrust over a shorter distance.

The available compromises can be summarized briefly. Good coupling requires thin build, short length coils, while high performance requires more massive coils. Thus higher current capacity can only be achieved at a sacrifice of coupling. Good coupling makes for higher peak inductance gradient, but shorter inductance length and therefore requires shorter current pulses; this can lead to unrealistic voltage requirements at high speed. It is clear once again that optimization conditions will vary along the acceleration path.

Our selected reference design, to be described later, eliminates some of these opposing requirements.

3.5 CONVERSION AND TRANSFER EFFICIENCY

Conversion efficiency of a launcher system is simply the fraction of electrically stored energy which is converted into kinetic energy of the projectile over the entire launch cycle, the remainder of initially stored energy being lost. This corresponds to what is normally meant by the "efficiency" of a motor.

However, the initially stored energy which is not converted into kinetic energy is not necessarily lost. In this case, we need to define a second kind of efficiency, the transfer efficiency.

Transfer efficiency is the fraction of stored energy converted into kinetic energy during each acceleration cycle. Of the energy not transferred, a certain fraction is lost either resistively, or in the commutation process. This commutation energy needs to be supplied between acceleration cycles.

In the original discrete coil synchronous coaxial launchers known as mass drivers, capacitors were used to store the unused coil energy between acceleration cycles. Fig. 3.3 (next page) shows the original mass driver circuit in which a number of drive coils are energized by one capacitor C, which in turn is re-charged by an intermediate energy store, shown here as a battery with series resistor. The sector capacitor is chosen so that the capacitor-coil circuit oscillates at approximately the frequency corresponding to the local projectile transit time.

Each drive coil is connected to the capacitor circuit by means of tandem SCRs triggered as the projectile coil passes the appropriate position. The capacitor voltage starts at the battery potential. As the first SCR is triggered at instant A,

current flows through the coil, exerting a pull on the approaching projectile coil, and the capacitor voltage swings to a negative maximum at B, which occurs at the instant the projectile coil passes through the plane of the first drive coil. The current then reverses, exerting a push on the departing projectile coil, and the voltage begins to rise, reaching a positive maximum at instant C. At this point the current has fallen to zero and the first drive coil is disconnected from the capacitor circuit. Since the discharge is highly oscillatory, most of the original energy in the capacitor has returned, and only a small fraction has been converted to kinetic energy or lost resistively in the circuit.

Before the capacitor is connected to the next drive coil, its energy is restored to the initial value by current flowing into it from the intermediate storage device shown as a battery, which may be a homopolar generator coupled to an intermediate storage inductor. The energy remaining in the capacitor is thus not lost, but used in the next discharge cycle. Although the transfer efficiency may be as low as 10 percent or even less, the overall conversion efficiency of such discrete coil accelerators or mass drivers can exceed 90 percent.

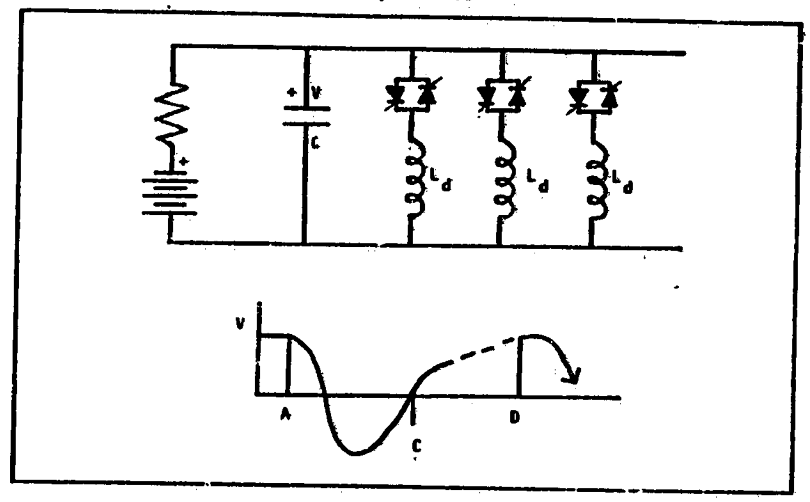

Fig. 3.3

If the recharge cycle requires more time than is available between successive drive coil passages, a multi-phase "Leapfrog" system must be used, in which a given capacitor drives every tenth coil for example, with nine other capacitors handling the intermediate coils.

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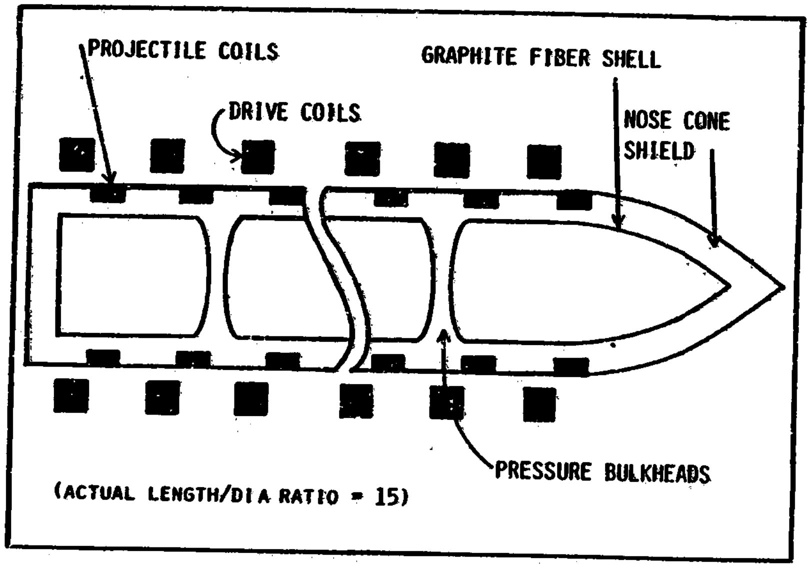

DISCRETE COIL LAUNCHER

FIG. 3.4

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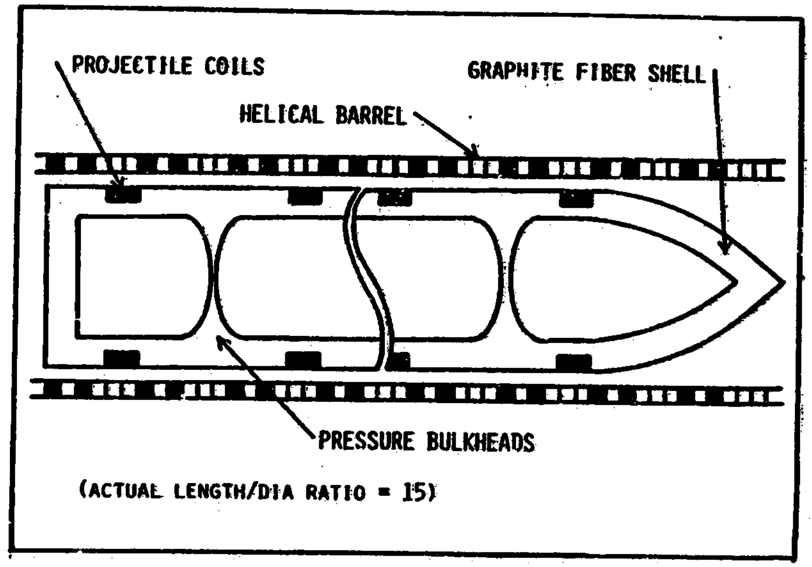

HELICAL BARREL LAUNCHER

FIG. 3.5

Fig. 3.4 shows a schematic cross section of a discrete coil launcher based on the mass driver approach described above. The projectile coils take the form of a series of rings, spaced about one radius apart. All rings are propelled simultaneously by matching drive coils, energized in the manner described above with the aid of sector capacitors tuned to the local transit period.

Fig. 3.5 shows an alternative configuration, a helical barrel launcher. In this case the discrete coils are replaced by a continuous helix, resembling the helix in a brush-commutated helical railgun. However, the helical turns are commutated individually by solid state switches, in such a manner that an energized segment of helix travels in synchronism with the projectile. This configuration will be described in detail in Section 6. At present it will suffice to point out the fundamental advantage of this configuration.

In the case of the continuous helical barrel, each individual turn is so tightly coupled to the remainder of the helix that 99 percent of the commutation energy is stored in the helix itself at the instant a turn is disconnected. Thus only one percent of the energy needs to be stored in commutation capacitors.

In addition to reducing the capacitor requirement, this approach has two other advantages: commutation switch load is drastically reduced, and the propulsion wave advances almost continuously, so that the mutual inductance gradient can be kept very near is peak value.

It is for these reasons that we have selected the helical barrel launcher as the reference design, to be described in Section 6.

3.6 POWER CONDITIONING AND ENERGY STORAGE

Although the launch energy itself is very inexpensive (65 cents per pound, to LEO), the cost of storing it with rapid access can be prohibitive. In other words, it is the needed power, not the energy, which is expensive. The main challenge in electromagnetic launch technology is therefore power conditioning, or more specifically power compression.

Capacitors release energy fast enough to permit synchronization of individual drive coils, but capacitive storage of the entire launch energy is completely out of the question. In the preceding section we described means for storing only a fraction of the commutation energy in capacitors, and using a much less expensive slow energy storage medium to store the launch energy. Large inductors represent the most promising candidate for primary energy storage.

The loss rate of a storage inductor is characterized by its L/R time constant, which is the time required for its current to fall to 1/e or about one third of its initial value when the inductor is short-circuited. In this period of time, about 85% of the originally stored energy will have been lost.

Fortunately the L/R time constant of inductors large enough to store the entire launch energy is several minutes, even if they are made of ordinary conductors at room temperature. A charging time of a few minutes therefore involves losing only about half the energy. Operation at liquid nitrogen temperature would lengthen the L/R time by a factor of about 8, but this complication may not be worth the economy, because energy itself is very inexpensive.

With a charging time of several minutes, the entire launch energy can be derived from existing powerlines or from a dedicated generator of commercially available size.

Fig. 3.6 illustrates several configurations which can be used for the primary energy storage inductor. The simplest is a "Brooks Coil", a coil of equal i.d., build and length, found by its inventor to provide the maximum inductance for a given amount of conductor. To store the energy for a typical mission, a Brooks coil of 60 meter (180 ft) outside diameter is required. It will probably be made of aluminum wire, and kept at room temperature or nitrogen temperature. It will cost about 500 million dollars. Table 3.1 lists the design characteristics of a range of Brooks coil inductors.

More complex storage inductors shown in Fig. 3.6 weigh and cost more than a corresponding Brooks coil, but have certain advantages. Toroidal configurations have very low external magnetic fields so as to minimize electromagnetic noise. Solenoidal configurations are more suitable for cryogenic or superconducting operation.

3.7 SYNCHRONIZATION AND SWITCHING

The advantages of coaxial launchers are purchased at the expense of a complication: the drive current needs to be synchronized with the motion of the projectile, as explained above. This represents a challenge in the form of switching technology, and also an opportunity by providing continuous, accurate, real-time control of the launch cycle.

The simplest means of synchronization is a brush sliding directly on the inside or outside surface of a helical barrel winding, thereby energizing segments of helix which push the travelling projectile coil from behind and pull it from the front. The source of drive current will be a pair of rails which also supply excitation current to the projectile coils. Brush

commutation is likely to fail above some limiting speed, which has not as yet been determined. It is likely to be in the one km/s range; rotary brush testers are limited by centrifugal forces to the range of 100 m/s. Some other switching method must therefore be used for all but the first few meters of a space launcher.

Table 3.2 provides an overview of available switching technology. Listed parameters represent available hardware rather than performance limits. In most cases the technology simply has never been pushed beyond present requirements in the power industry. The state of the art might be summarized as 100 kiloamperes at 100 kilovolts, the limit being vague and soft. Since switching requirements depend on launcher length and projectile mass, it is not possible to express any firm limit in launch velocity.

Beyond the velocity limit of synchronous launchers, it will be necessary to use railguns. It must be emphasized, however, that although railguns involve no switching, they are still subject to a flash-over voltage between rails which depends on rail separation and the medium between rails, presumably the highest vacuum achievable between launches. This voltage is also in the vicinity of 100 kV.

3.8 FAILURE MODES AND REDUNDANCY

As in other forms of air and space transportation, success in electromagnetic launch technology depends on our ability to provide absolute assurance of some minimum acceptable predetermined statistical safety level. This minimum acceptable safety level will of course depend on what we are launching: inert cargo, sensitive cargo (such as high level nuclear waste), or human passengers.

A crucial advantage of electromagnetic over chemical rocket launch technology is its ability to provide any degree of redundancy we are willing to pay for. The potential redundancy of electromagnetic launch technology is even higher than that of aviation, since the powerplants are entirely ground-based and the vehicle is entirely passive. There are, after all, few airborne propulsion devices more infallible than two dozen massive copper rings embedded in a shell of carbon fiber composite.

It may therefore be a mistake to rule out the acceptability of launching nuclear waste without crash-proof shielding, in vehicles small enough to optimize the duty cycle.

To approximate fail-safe operation, it would suffice to provide two or even three times the required number of drive coils, energized by independent supply sources which can be called upon if the first system fails to perform to specified minima.

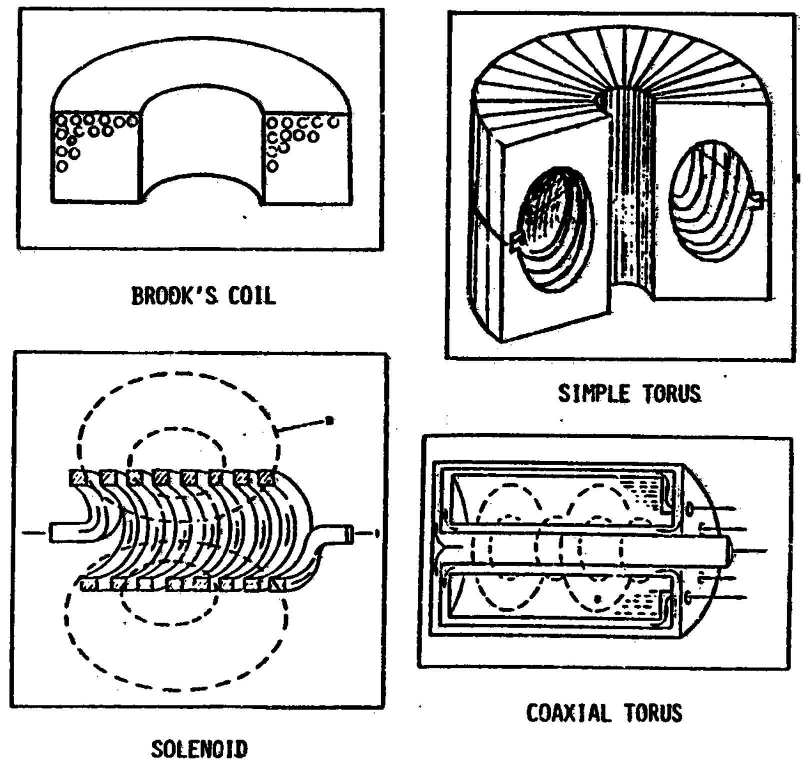

FIG. 3.6

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TABLE 3.1

BROOKS COIL STORAGE INDUCTOR				L/R TIME CONSTANT			
OUTSIDE DIAMETER (METERS)	MASS (METRIC TONS) ALUMINUM	MASS (METRIC TONS) COPPER	ENERGY (GJ)	τ AT 20° C (SECONDS) ALUMINUM	τ AT 20° C (SECONDS) COPPER	τ AT 77° K (SECONDS) ALUMINUM	τ AT 77° K (SECONDS) COPPER
2	17.9	58.9	.017	7.5	12.5	60	100
5	279	920	.259	47	78	375	625
10	2,230	7,360	2.07	187	312	1500	2500
20	17,900	58,900	16.5	750	1250	6000	10,000
50	279,000	920,000	259	4,690	7810	37,500	62,500

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TABLE 3.2

SWITCHING OVERVIEW

	CURRENT (KA)	VOLTAGE (KV)	FREQUENCY (KHZ)	INTERRUPTION TIME (S)
GAS GAP	10^3	10^3	1	10^{-3}
VACUUM GAP	10^2	10^2	10	10^{-5}
THYRATRON	10	10	10	10^{-6}
IGNITRON	10^3	10^2	0.1	10^{-2}
SOLID STATE	10	10	10^5	10^{-6}
FUSES & EXPLOSIVES	10^3	10^3	0	10^{-6}
MECHANICAL	10^3	10^3	0.01	10^{-2}

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SECTION 4. FUNDAMENTAL LIMITS

We are concerned here only with an assessment of the fundamental limits to electromagnetic acceleration, and how these limits are affected by our design decisions. Exact limits will depend on detailed design and are far beyond the scope of this study.

The preceding section provides an understanding of the basic limits, which are imposed by the following requirements:

(1) Heat dissipated in projectile coils, drive coils and leads cannot be removed during the launch cycle; the process is adiabatic: thermal inertia must keep temperature rise to within acceptable limits.

(2) Acceleration force and magnetic pressure on all coils or rails must be contained by the conductors and insulating structure which separates them.

(3) The voltage required to make the drive coil pulses match the transit time must be withstood by the coil insulation.

We shall consider the implication of these three limits.

4.1 THERMAL LIMITS

Energy loss reflected in the conversion efficiency appears as heat during the launch cycle, in fractions of a second. The typical thermal relaxation (equilibration) time of a centimeter-size coil is tens of seconds. Therefore, not much of the heat can be removed in the short time during which it is generated. The launch cycle is adiabatic. The various system components must therefore have enough thermal inertia to keep their temperature from reaching destructive levels. Active cooling is involved only in thermal recovery between successive launches, and will not be considered here.

Several tools are at our disposal for dealing with adiabatic thermal containment:

(1) We can increase the amount of conductor to decrease current density and increase thermal inertia, at a cost of increased coil mass and decreased coil proximity, or coupling.

(2) We can select a conductor on the basis of its adiabatic properties, i.e. specific heat and temperature limit.

(3) We can select insulators capable of withstanding very high temperatures.

(4) We can pre-cool the conductor to store negative heat energy and reduce electrical resistivity.

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The heat energy generated per unit mass per unit time by joule heating is given by:

$$\frac{dQ}{dt} = \rho J^2 \qquad (4.1)$$

where Q = energy density in joule/cubic meter
ρ = electrical resistivity in ohm-meter
J = current density in amperes/square meter

Other heat sources (friction, hysteresis, radiation and conduction) are negligible by comparison. The rate of local temperature rise is given by:

$$\frac{dT}{dt} = \frac{\rho J^2}{C_v} \qquad (4.2)$$

where C_v = specific heat at constant volume, J/kg.

Since both the specific heat and the resistivity are highly temperature-dependent, the temperature rise must be computed by integration of the expression:

$$J^2(t) = \frac{C_v(T)}{\rho(T)} \frac{dT}{dt} \qquad (4.3)$$

with the result:

$$\int_0^t J^2 dt = \int_{T_i}^{T_f} \frac{C_v}{\rho} dT \qquad (4.4)$$

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where subscripts i and f denote initial and final temperatures in a time interval 0-t. The expression at left is an "action integral", the "current integral", which has been tabulated explicitly for various materials. Fig. 4.1 below shows it for copper, at three initial temperatures [Pulsed High Magnetic Fields, H. Knoepfel, p. 87, North Holland publishing Co., NY 1970].

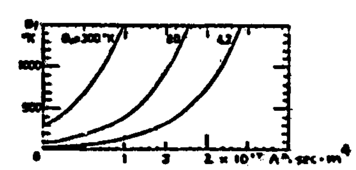

Fig. 4.1 Current integral for copper at three initial temperatures

If we approximate electrical resistivity by a commonly used expression:

$$\rho = \rho_o(T_i)\left[1 + \alpha(T_f - T_i)\right] \quad (4.5)$$

where alpha is a tabulated characteristic, and approximate the specific heat by a constant, then equation (4.4) reduces to:

$$\int_o^t J^2 dt = \int_{T_i}^{T_f} \frac{C_v}{\rho_o} \frac{dT}{1-\alpha(T_f - T_i)} \quad (4.6)$$

or

$$J^2 t = \int_{T_i}^{T_f} \frac{C_v}{\rho_o} \frac{dT}{1+\alpha(T-T_i)} = \frac{C_v}{\rho_o \alpha} \ln\left[1+\alpha \Delta T\right] \quad (4.7)$$

where $\Delta T = T_f - T_i$

We can now obtain a closed analytical expression for the temperature rise:

$$\Delta T = \frac{1}{\alpha}\left[e^{\rho_o \alpha J^2 t / C_v} - 1\right] \quad (4.8)$$

For projectile or drive coil purposes, the limiting temperature is some point substantially below the melting point, imposed by strength or insulator considerations. For orbital transfer reaction engines which expel metal rings as reaction mass, it is permissible and even desirable to let ohmic heating cause the metal rings to melt and vaporize. In this application the current integral can be established as a function of total enthalpy, i.e., as the integral of specific heat over temperature, plus the latent heats of melting and evaporation. The total current integral including these two phasetransitions is tabulated below (Knoepfel, p. 87):

Table 4.1 Current integral from room temperature to four states

	solid at melting point	liquid at melting point	liquid at boiling point	vapor at boiling point
aluminum	3	4	5.9	10.9
copper	8.9	10.5	12.4	19.5

dimensions: $\times 10^8 amp^2$ - sec / cm^4.

With the widespread use of superconductors it will be asked why resistive heating limits cannot be eliminated altogether in projectile and drive coils.

Aside from the obvious complexity of cryogenic insulation and radiation shielding requirements, the reduced proximity and coupling, and the problems of induced heating, there is the often overlooked observation that normal conductors will easily outperform superconductors in current density for periods of the order of 0.1 to 1 second.

More specifically: niobium-tin can sustain superconducting current densities in excess of 200 kA/sq cm, but necessary stabilization conductor will reduce this effectively to 25 kA/sq cm. Copper, on the other hand, has no intrinsic current density limit and can easily sustain any current density as long as thermal inertia (and mechanical support) will allow. For a current integral of 4×10^8 amp^2 sec/cm^4 (corresponding to a temperature rise from room temperature to 300 C) copper will carry 25 kA/sq cm for about half a second. The corresponding period for aluminum is about one third second. In other words, for duty periods of the order of half a second, normal conductors are preferable to superconductors.

The allowable current integral limit can be extended by precooling the conductor. As a rough rule of thumb, the current integral doubles when precooling with liquid nitrogen (77K). Nitrogen is inexpensive and containable in simple styrofoam insulated vessels and pipes.

Additional thermal inertia can be gained by precooling to even lower temperatures, but gains become progressively smaller. More elaborate cryostats and more expensive cryogens are necessary, and specific heats asymptotically approach zero. At liquid helium temperature, specific heats are 2,000 times lower than at liquid nitrogen temperature.

4.2 MECHANICAL LIMITS

Mechanical containment of the pulsed-magnetic pressures and thrusts involved in a launch system is a very complex problem, dealt with at length by Mongeau [P.Mongeau, "Coaxial Air Core Electromagnetic Accelerators", Ph.D. Thesis M.I.T., Physics Department, October 1981]. Simplistic estimates based on quasistatic approximations have been used to assess the mechanical limits, with highly misleading results.

The complexity is related to several features. First, the time scale of force containment is intermediate between static and dynamic so that neither approximation is valid; an exact solution requires detailed design work, and involves subtle strain-rate effects. Second, intuitively non-obvious induction

effects dominate. For instance, placing a projectile ring in close proximity to a pulsed drive coil actually reduces the bursting forces on the drive coil. This effect is directly opposite to the effect of placing a projectile into a chemical gun barrel, which produces an increase in bursting pressure.

There are three distinct force containment areas:

(1) drive coils and their insulation must withstand pulsed hoop stresses and pulsed axial reaction forces;

(2) projectile coils or short-circuited rings must withstand hoop stresses;

(3) acceleration thrust must be transmitted non-destructively from the projectile coils to the entire sabot (if any), launch vehicle and payload.

4.2.1 STATICS

To compute hoop stresses we approximate the drive coils by a long solenoid of thin build having dimensions defined in Fig. 4.2 below:

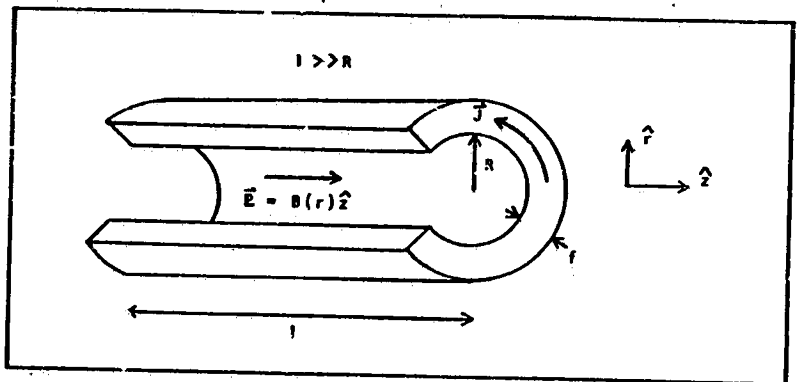

Fig. 4.2 Long thin-build solenoid dimensions

The purely axial field inside can be derived from Ampere's Law:

$$\begin{array}{ll} \text{inside the bore: } r \le R: & B(r) = \mu K \\ \text{in the build: } R+S \ge r \ge R: & B(r) = \mu K\left[\frac{R+s-r}{s}\right] \\ \text{outside: } r > R+S & B(r) = 0 \end{array} \quad (4.9)$$

where is the magnetic permeability and K is the surface current, defined as the current density times the build: K = J s .

The magnetic pressure acting on a thin shell of thickness delta r is:

$$P_{mag}(r) = \frac{1}{2\mu}\left[B^2(r) - B^2(r+\Delta r)\right]\hat{r} \simeq \frac{\Delta r}{2\mu}\frac{dB^2}{dr}\,\hat{r} \quad (4.10)$$

where the factor of 1/2 comes from averaging the field within the build. Stress in the shell is found by balancing magnetic pressure against tangential tension, as indicated in Fig. 4.3 below:

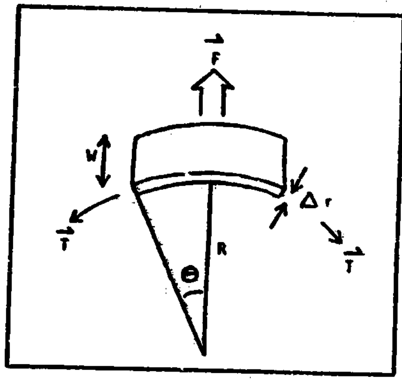

Fig. 4.3

Containment of magnetic pressure by hoop tension forces

The total magnetic force acting radially on the section is

$$F = P_{mag} \times (area) = 2PR\theta w \quad (4.11)$$

This force must be balanced by the radial component of tension in the shell:

$$F = 2T\sin\theta \simeq 2T\theta \quad (4.12)$$

which defines the shell stress:

$$\sigma = \frac{T}{w\Delta r} = P_{mag}\frac{R}{\Delta r} \quad (4.13)$$

assuming the layers are self-supporting (no stress transmittal between layers), we obtain the stress distribution in the build:

$$\sigma(r) = \mu I^2\left[R+s-r\right]r \quad (4.14)$$

which is plotted in Fig. 4.4 (next page).

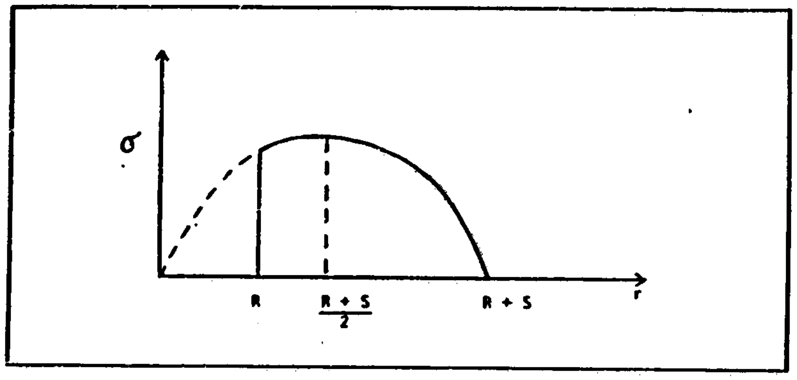

Fig. 4.4 Distribution of hoop stress in the build of a long coil

For thin-build solenoids, i.e., build small compared to radius, we can define an average stress:

$$\sigma_{ave} = \frac{\mu J^2 s R}{2} \qquad (4.15)$$

For comparison, the complete tangential stress including stress sharing among layers can be expressed as

$$\sigma_\theta = \frac{\mu J^2 s}{2(\alpha-1)}\left\{\frac{3+\nu}{8}(\alpha^2+1) - \frac{2+\nu}{3}\left[\frac{\alpha^2(\alpha^2-\alpha)}{\alpha^2-1}+\alpha\right] + \frac{R^2}{r^2}\left(\frac{3+\nu}{8}\alpha^2 - \frac{2+\nu}{3}\frac{\alpha^2(\alpha^2-\alpha)}{\alpha^2-1}\right) + \frac{1+2\nu}{3}\alpha\frac{R}{r} - \frac{1+3\nu}{8}\frac{R^2}{r}\right\}$$

Where $\alpha = \frac{R+S}{R}$ and ν is Poisson's ratio. (4.15a)

Although actual drive coils are neither long nor thin-walled, the above approximation (eq. 4.15) provides an upper limit and scaling relations accurate to within 50%.

Thus, we can deal with magnetic pressure as analogous to gas pressure. For intuitive feel, it is useful to remember that a field of 1 tesla (typical of iron electromagnets) corresponds to a pressure of 60 psi (typical of truck tires). From this reference point, magnetic pressure increases as the square of field intensity B. For example:

0.1 tesla = 0.6 psi = pressure under a glass of water

1 tesla = 60 psi = pressure in truck tire

10 tesla = 6000 psi = pressure in rifle chamber

100 tesla = 600,000 psi = pressure in core of earth.

For static containment of a magnetic field, the yield strength of the coil material must exceed the stress level throughout the winding. Table 4.2 is a collection of relevant properties of all coil materials of interest, as a starting point. Actually static containment is not an applicable approximation since inertial effects play a dominant role. Moreover, the coil material itself will almost certainty be reinforced by pre-stressed high-strength tensile elements. The tabulation should therefore only serve as a very preliminary comparison base.

Tungsten is an intriguing material that has not received the attention it deserves. Its high conductivity and exceedingly high strength combined with superior high temperature characteristics are likely to outweigh its high density, high cost, and poor machinability. Its high density will contribute significantly to inertial containment in the maximum velocity region of the accelerator.

For static analysis, thermal as well as mechanical limits must be considered. By way of design example, we consider a thin-build coil in which the stress is:

$$\sigma = \mu\frac{J^2 s R}{2} = \frac{B^2 R}{2\mu s} \qquad (4.16)$$

The current integral for the coil is:

$$\int J^2 dt = G(T) = \frac{B^2 t}{\mu^2 s^2} \qquad (4.17)$$

where G(T) is the action constant as a function of the temperature rise in the conductor. These two equations define the thermal and mechanical constraints:

$$B = \sqrt{\frac{\mu^2 s^2 G(T)}{t}} \quad \text{thermal} \qquad (4.18)$$

$$B = \sqrt{\frac{2\,\sigma_{yield}\,\mu s}{R}} \quad \text{mechanical}$$

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Table 4.2

Mechanical Properties of Some Metals

(Collected from Metal's Handbook Vol. 1, T. Lyman, ed.)

Material	Conduct -ivity % IACS	density gm/cm^3	Young's modulus x10^6 psi	yield strength x10^{3}psi	tensile strength x10^{3}psi
Copper alloys Cu-OFHC					
annealed	101	8.94	17	10	32
hard-drawn	100	" "	" "	50	55
Chromium-Cu (1%)	80-90	8.7	16.5	43	55
Be-Cu (0.5%)	60	8.4	17	50-75	75-90
Be-Cu (2%)	22	8.23	17	110-130	130-170
Brass (70%Cu-30%Zn)					
annealed	27	8.53	16	11	44
spring stock	" "	" "	" "	65	99
Aluminum alloys EC alloy					
H19 temper	52	2.7	10	24	27
6061					
T0 temper	40	2.7	10	8	18
T6 temper	" "	" "	" "	40	45
6063 T6 temper	53	2.7	10	31	35
7075 T6 temper	30	2.8	10.4	73	83
Stainless St. 304					
annealed	2	7.9	28	35	85
30% cold work	" "	" "	" "	120	140
Tungsten	30	19.3	50		250-600

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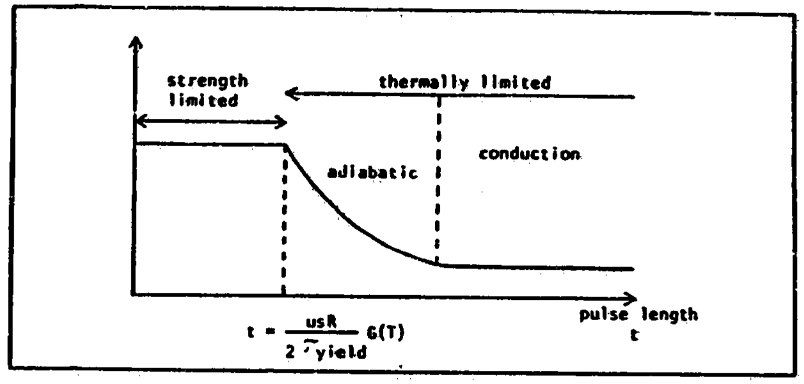

Fig. 4.5 Mechanical and thermal field limits as a function of pulse duration

As might be expected, the coil is strength-limited for short pulses and heating-limited for longer pulses. For very long pulses, ohmic heating will eventually reach equilibrium with heat loss by conduction and radiation. Different parts of a launcher will operate in different regions of this curve. Drive coils will operate in the mechanical limit region, while projectile coils are likely to be thermally limited. A storage inductor, on the other hand, will operate in the steady-state equilibrium domain and is likely to require active cooling or cryogenic precooling.

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4.2.2 QUASISTATICS

As the load is applied to a coil more and more abruptly, we approach a point at which inertial effects are no longer negligible, and the preceding static analysis no longer applies. We define the quasi-static domain as the domain in which both static and inertial containment are important. Eventually the coil is held together entirely by inertia and static containment becomes negligible; we call this the purely dynamic range.

The quasistatic range can be said to begin when the pulse duration becomes less than the length of time it takes for an elastic wave to traverse the coil. This elastic time constant can be defined as:

$$\text{elastic time constant} = \tau_e = \frac{\ell}{c_e} \qquad (4.19)$$

where c_e is the velocity of an elastic wave in the material, and ℓ is the characteristic dimension of the coil.

The quasistatic equation of motion is:

$$\sigma = \rho \ell^2 \frac{\partial \epsilon}{\partial t^2} + E\epsilon \qquad (4.20)$$

where σ is stress and ϵ is strain. For a thin-build coil this expression reduces to:

$$P_{radial} = \rho s \frac{d^2 \Delta r}{dt^2} + \frac{E s \Delta r}{R^2} \qquad (4.21)$$

where s is the build, Δr is the radial yield, R is the initial radius, E is Young's modulus, ρ is density, and P_{radial} is the radial pressure. This equation has a homogeneous solution of the form:

$$\Delta r(t) = A \sin \omega t + B \cos \omega t \qquad (4.22)$$

which defines a characteristic frequency omega:

$$\omega = \sqrt{\frac{E}{\rho R^2}} \qquad (4.23)$$

and a characteristic oscillation period

$$T = \frac{2\pi}{\omega} = 2\pi R \sqrt{\frac{\rho}{E}} \qquad (4.24)$$

which separates the quasistatic from the static regime. For a copper cylinder of 5 cm radius, for instance, the period is 86 microseconds. This can be thought of as the frequency at which it will ring when struck.

$$\Delta r(t) = \frac{P_o R^2}{E s}\left[1 - \cos \omega t\right] \qquad (4.25)$$

If we apply a square pulse of duration tau to a coil of period T, where tau is much greater than T, it will respond like a bell being struck by a single impulse, with an oscillating radial expansion shown in Fig. 4.6, where we have assumed a realistic amount of internal damping:

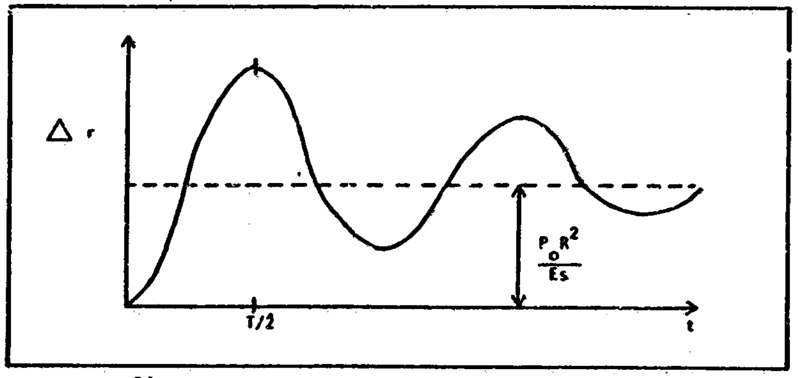

Fig. 4.6 Coil response to a radial impact

Although the final steady state expansion is , as indicated in the figure, the initial expansion can exceed this value by a factor of two. The maximum radial expansion is thus:

$$\Delta r_{max} = \frac{2 P_o R^2}{E s} \quad \text{for } \tau > T/2 \qquad (4.26)$$

and the associated strain is:

$$\epsilon_{max} = \frac{\Delta r_{max}}{R} = \frac{2 P_o R}{E s} \qquad (4.27)$$

Since the maximum allowable stress is:

$$\epsilon_{limit} = \frac{\sigma_{yield}}{E} \qquad (4.28)$$

the maximum allowable field is:

$$B_{max} = \sqrt{\frac{\sigma_{yield}\, \mu s}{R}} \qquad (4.29)$$

If tau approximately equals the half-period T/2, the quasistatic response allows a maximum field only 1/2 that of the static limit (assuming a force rise time much shorter than the period).

For pulse durations tau less than half the response period T/2, the radial expansion will be less, but in this regime the phase becomes important. The total response is given by:

$$\Delta r = 0 \qquad t < 0$$

$$\Delta r = \frac{P_o R^2}{E s}\left[1 - \cos \omega t\right] \qquad 0 \le t \le \tau$$

$$\Delta r = \Delta r_{max} \cos(\omega t + \phi)$$

$$\Delta r_{max} = \frac{P_o R^2}{E s}\sqrt{2(1 - \cos \omega \tau)} \qquad (4.30)$$

where the phase shift is given by:

$$\phi = \tan^{-1}\left[\frac{\sin \omega\tau}{\cos \omega\tau - 1}\right] - \omega\tau \qquad (4.31)$$

The response is plotted in Fig. 4.7.

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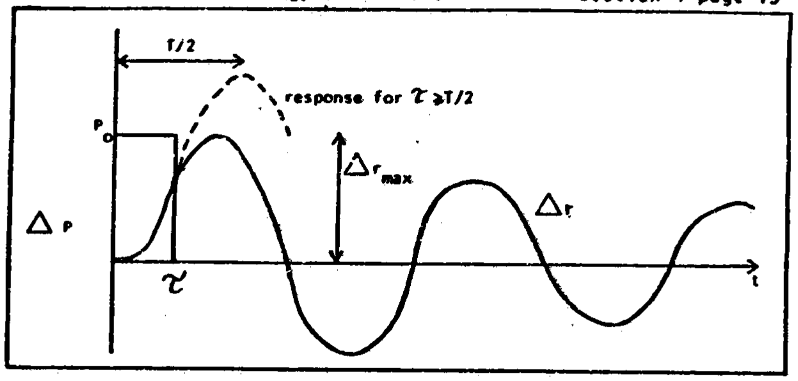

Fig. 4.7 Quasistatic response to a short impulse

The limiting field in this regime is found to be:

$$B_{max} = \left[\frac{2\mu s \sigma_{yield}}{R\sqrt{2(1-\cos\omega T)}}\right]^{1/2} \quad (4.32)$$

As tau decreases, the coil can withstand an increasing field, up to the purely dynamic range, to be discussed below. An approximate expression for the maximum field is:

$$B_{max} = \sqrt{\frac{2\mu \sigma_{yield} s}{R \omega T}} \quad (4.33)$$

The combined quasistatic and static field limit is plotted in Fig. 4.8.

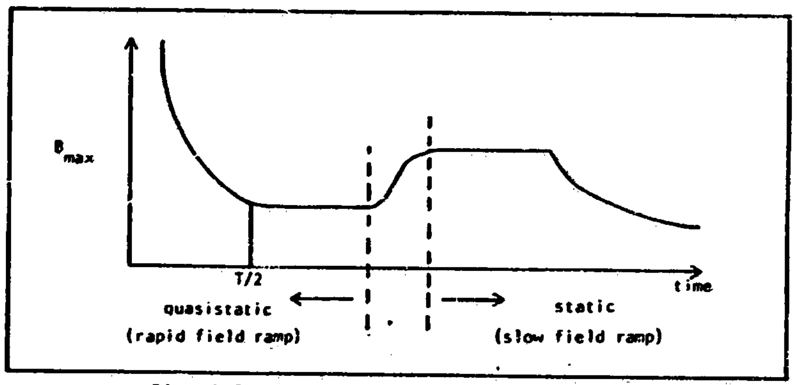

Fig. 4.8 Quasistatic and static response

The yield strength of materials is actually strain-rate dependent, as indicated in Fig. 4.9. It shows the stress-strain curve for maraging steel at different strain-rates. The yield strength is seen to increase with strain rate, but the elastic slope (Young's modulus) is constant.

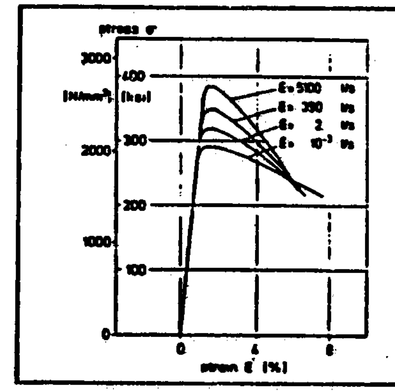

Fig. 4.9

Yield strength as a function of strain rate

[Myers & Murr, p. 58]

Typically, strain rate effects become important above about 10/sec. For our example copper solenoid (s = 1 cm, R = 5 cm), strain rates above 10/sec are achieved with fields of only 2 tesla. This suggests that strain rate effects will be important for fields having rise-times less than the period T.

4.2.3 DYNAMIC STRENGTH

As the rate of force application becomes even more abrupt, local strain rate eventually exceeds the rate at which elastic deformation can distribute the strain, and failure becomes a local event. In this regime a body may withstand either more or less impact than in the static or quasistatic domain, and dynamic strength must therefore be evaluated separately.

In terminal ballistic studies involving impact, it is customary to assume a surface impact velocity and density. By equating the incident kinetic energy density to the strain energy density:

$$\frac{1}{2}\rho V^2 = \int \sigma(\epsilon)\, d\epsilon \quad (4.34)$$

it is possible to relate a critical impact velocity to an elastic strain limit [J.Rinehart and J.Pearson, "Behavior of Metals under Impulsive Loads", Dover, NY 1954, p16]:

$$V_{critical} = \sqrt{\frac{E}{\rho}}\, \epsilon_{limit} = \frac{\sigma_{yield}}{\sqrt{E\rho}} \quad (4.35)$$

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If particle velocity ever exceeds this critical velocity, the material will yield locally. The equality of impulse and momentum:

$$\text{Impulse} = \int \text{force } dt = mv = \text{momentum} \quad (4.36)$$

can be expressed for magnetic pressure:

$$\text{Mag. impulse} = I_{mag} = \int F_{mag} dt = \frac{B^2}{2\mu} \text{Area} \times t \quad (4.37)$$

The impulse mass (as in the case of terminal ballistics) is a time-dependent quantity which depends on the stress wave velocity in the material. Consider for example a solenoid cross section as shown in Fig. 4.10.

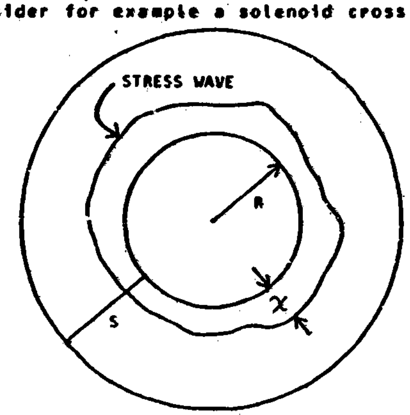

Fig. 4.10

Stress wave propagating into solenoid build from inside surface

Assume the magnetic pressure arises from a very thin skin depth layer along the inside surface. The impulse per unit length of solenoid is

$$i = P_{mag} \times 2\pi R t = B^2 \frac{\pi R t}{\mu} \quad (4.38)$$

The impulsive load generates a stress wave which defines a cylindrical shell of thickness X:

$$X = c_e t \quad (4.39)$$

where c is the elastic wave velocity in the material. The impulse thus couples to an effective mass per unit length:

$$m(t) = 2\pi R \rho X = 2\pi R \rho c_e t \quad (4.40)$$

where we assume x to be much smaller than the radius R. The effective impact velocity is then:

$$v = \frac{B^2}{2\rho c_e \mu} \quad (4.41)$$

Using the critical velocity as an impact velocity limit, we can derive a maximum field intensity of

$$B_{max} = \sqrt{2 v_{crit} \rho c_e \mu} = \sqrt{2\sigma_{yield} c_e \mu \sqrt{\frac{\rho}{E}}} \quad (4.42)$$

which is not time-dependent. In other words, this purely dynamic limit is valid for arbitrarily short impact times. As impact duraction increases, response will enter the quasistatic regime analyzed in the preceding section. This transition will occur when the stress waves have equilibrated throught the bulk of the material. In our example, the dividing point is defined by a pulse duration tau depending on the ratio of build to elastic wave velocity:

$$\Upsilon = \frac{S}{C_e} \quad (4.43)$$

and the response can be considered dynamic below and quasistatic above this impulse period.

Elastic wave velocity is a function of material properties as well as geometry. For example, a long slender rod or wire will support only longitudinal (compression) waves. We can model such a rod as a linear chain of lumped masses and springs, as shown in Fig. 4.11.

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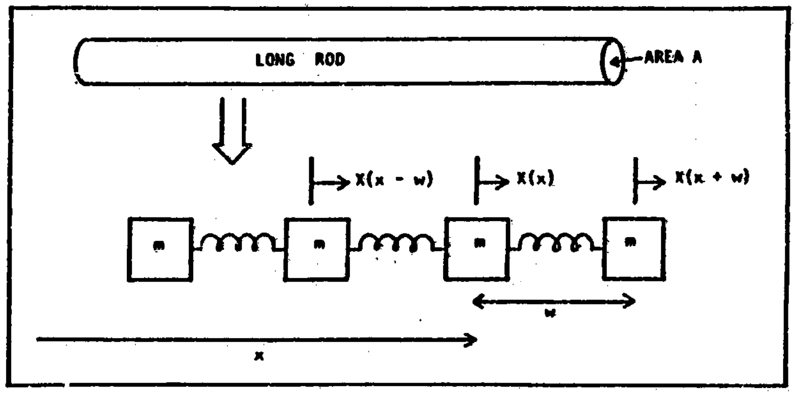


Fig. 4.11 Lumped mass model of slender rod

The equation of motion of an isolated mass lump is:

$$m\ddot{x}(x) = F_{tot} \qquad (4.44)$$

and the mass of this lump corresponds to:

$$m = Aw\rho \qquad (4.45)$$

The total force on this lump is the sum of elastic forces acting on each end:

$$F_{tot} = \frac{AE}{w}\left\{\left[x(x+w)-x(x)\right]-\left[x(x)-x(x-w)\right]\right\} = AE\frac{d^2x}{dx^2}w \qquad (4.46)$$

Solving for X, we obtain the standard wave equation:

$$\frac{\partial^2 X}{\partial t^2} = \frac{E}{\rho}\frac{\partial^2 X}{\partial x^2} \qquad (4.47)$$

which implies a propagation velocity

$$v_c = \sqrt{\frac{E}{\rho}} \qquad (4.48)$$

An infinite medium will support shear as well as longitudinal waves, whose velocities will be:

$$C_e = \left[\frac{3K(1-\nu)}{\rho(1+\nu)}\right]^{1/2}; C_t = \left[\frac{G}{\rho}\right]^{1/2} \qquad (4.49)$$

where k is the bulk modulus, G is the shear modulus, and nu is Poisson's ratio. [see E.Bruno, ec, "High Velocity Forming of Metals", Am. Soc. of Tool and Manuf. Engr, Dearborn, MI 1986, p29].

The three stress velocities for three common materials are tabluated below: compression wave in slender rod, compression wave in infinite medium, and shear wave in infinite medium:

Table 4.3 Elastic Wave Velocities

	compression in slab m/s	shear in slab m/s	compression in rod m/s
aluminum	6,370	3,110	5,050
copper	4,300	2,260	3,630
steel	5,940	3,200	4,940

If we assume the propagation velocity to be an approximate coupling velocity $v_c = \sqrt{E/\rho}$, then the maximum allowable dynamic field is

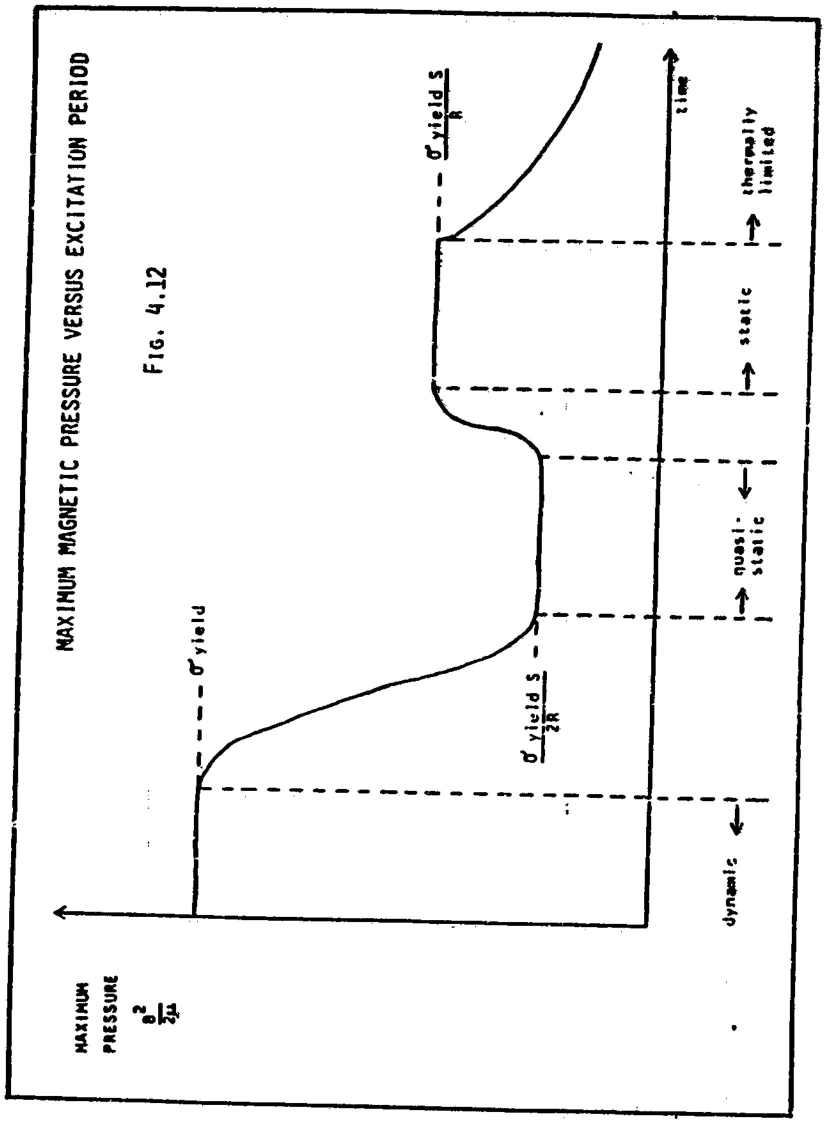
$$B_{max} = \sqrt{2\,\sigma_{yield}\,\mu} \qquad (4.50)$$

which is identical to the static field Limit for an infinite sample, where:

$$P_{max} = \frac{B^2_{max}}{2\mu} = \sigma_{yield} \qquad (4.51)$$

For purely dynamic conditions, the sample will respond according to its bulk properties as if it were under pure hydrostatic compression or tension locally.

The maximum permissible magnetic pressure for all three ranges analyzed is plotted in Fig. 4.12, next page.



MAXIMUM MAGNETIC PRESSURE VERSUS EXCITATION PERIOD

Fig. 4.12

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4.3 ELECTRICAL LIMITS

Electrical limits are not as clearly definable as thermal and mechanical limits because they involve extrapolation of component technology beyond performance regions commonly used in the power industry.

The uncertainty is not in the current and voltage required, but in the ability of insulators and switchgear to deal with them. The nature of the electrical problems can be summarized as follows:

For each type of accelerator, a certain current product is required to generate a required thrust. The required thrust depends on projectile mass, launch velocity, and launcher length.

This required current must be supplied at a voltage which satisfies two requirements: it must overcome the back-emf, and it must ensure an adequately fast rise-time in the case of synchronous launchers. Both the back-emf and the required current rise rate are proportional to projectile velocity.

In table 4.4 below we have summarized all relevant electrical scaling relations in normalized form, for the three electrically distinct types of accelerators:

Railguns, single or multiple segment,

Solenoid guns with projectile coil excitation, including brush-commutated helical railguns and solid state commutated discrete coil mass drivers, and

Pulsed induction guns, in which projectile coil current is induced by the drive coils.

The current I in projectile and drive coils is seen to depend on the required thrust force F, the inductance gradient, the number of coil turns N, and the number of simultaneously active coils (phases) n.

The voltage V required to drive this current depends on the same quantities, plus the local velocity V.

The "back-emf resistance" R(bemf), which is the quotient of the previous two columns, represents the apparent resistance due to the back-voltage which is generated by the moving projectile coil. As in any electric motor, this back-emf is proportional to velocity.

TABLE 4.4

ELECTRIC SCALING RELATIONS

TYPE	CURRENT (I)	VOLTAGE (V)	RESISTANCE (R_{BEMF})
RAILGUN	$\sqrt{F} \times \sqrt{\frac{2}{dL/dZ}}$	$\sqrt{F} \times \frac{V}{\eta}\sqrt{\frac{dL/dZ}{2}}$	$\frac{V}{2\eta}\ \frac{dL}{dZ}$
DC SOLENOID ACCELERATOR	$\sqrt{F} \times \sqrt{\frac{1}{N_{n}\, dM/dZ}}$	$\sqrt{F} \times \frac{V}{\eta}\sqrt{\frac{N\, dM/dZ}{n}}$	$\frac{VN}{\eta n}\ \frac{dM}{dZ}$
PULSED INDUCTION ACCELERATOR	$\sqrt{F} \times \sqrt{\frac{2}{N^2 n \left.\frac{dM}{dZ}\right\vert_{eff}}}$	$\sqrt{F}\ \frac{V}{\eta}\sqrt{\frac{(\dot{L}_{eff})^2 N^2 \pi^2}{n(2\ell_s)^3 \left.\frac{dM}{dZ}\right\vert_{eff}}}$	$\frac{V}{\eta}\ \frac{\dot{L}_{eff} N^2 \pi}{n 2\ell_s}$

N = # OF TURNS

n = # OF ACTIVE PHASES

F = DESIRED FORCE

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4.4 SCALING RELATIONS AND TRADE-OFFS

We have summarized in graphic form the scaling laws which govern some of the crucial design parameters under reasonable assumptions.

Fig. 4.13 shows the inductance gradient for all distinct types of accelerator capable of launching a 6500 kg projectile at 10 km/sec. We have assumed a 2 km launcher length. In the case of the coaxial launchers, we have assumed a 30 cm caliber and shown the inductance gradient as a function of coil separation.

Fig. 4.14 is a log-log plot of the force required to launch a 650 kg projectile with a 2 km long launcher using 100 kilovolts as a function of desired launch velocity, showing also the force available as a function of velocity from the distinct types of launcher. Coaxial launchers have been assumed to have a caliber of 30 cm with reasonable coil separation.

The intersection of each curve with the required force curve (A) represents the velocity limit of the particular type of accelerator, if driven at 100 kilovolts.

The apparent superiority of the railgun is misleading, because the gap of a railgun would not stand off a 100 kilovolt drive voltage unless it were much larger than is reasonable. Once arcing in a railgun is initiated, it cannot be extinguished. Coaxial drive and projectile coils, on the other hand, can be insulated to stand off 100 kV.

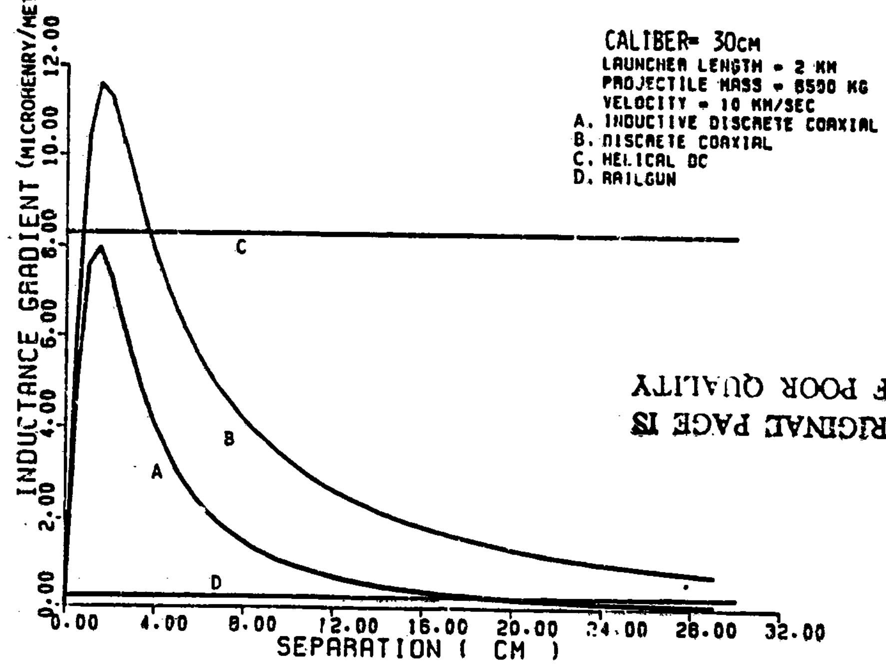
The helical railgun (brush-commutated helical launcher) is also misleading, because brush-commutation is not likely to work at 2 km/sec. The limiting speed for brush commutation remains to be determined.

Fig. 4.15 is a log-log plot of structural mass required as a function of launch velocity for reasonably long projectiles, under the same base assumptions made previously, assuming a maximum stress level of 100 ksi. We also assume that coaxial projectiles have multiple coils (or rings) spaced about one diameter apart. The required projectile structure can serve simultaneously for shielding.

Fig. 4.16 is a semi-log plot of mass fraction versus boost velocity, for a hybrid electro-chemical vehicle. It shows how both total delivered mass and payload mass approach unity as the electric boost velocity approaches orbital velocity of 8 km/s. The ratio of structure to total mass is assumed to be 0.1.

INDUCTANCE GRADIENT
FOR
VARIOUS ACCELERATORS

FIG. 4.13



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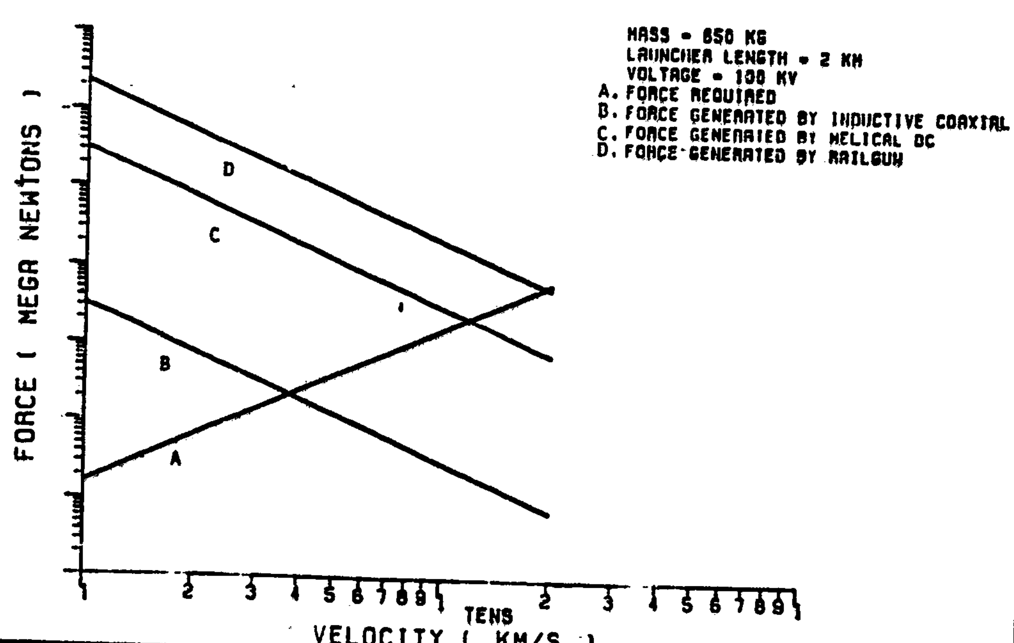
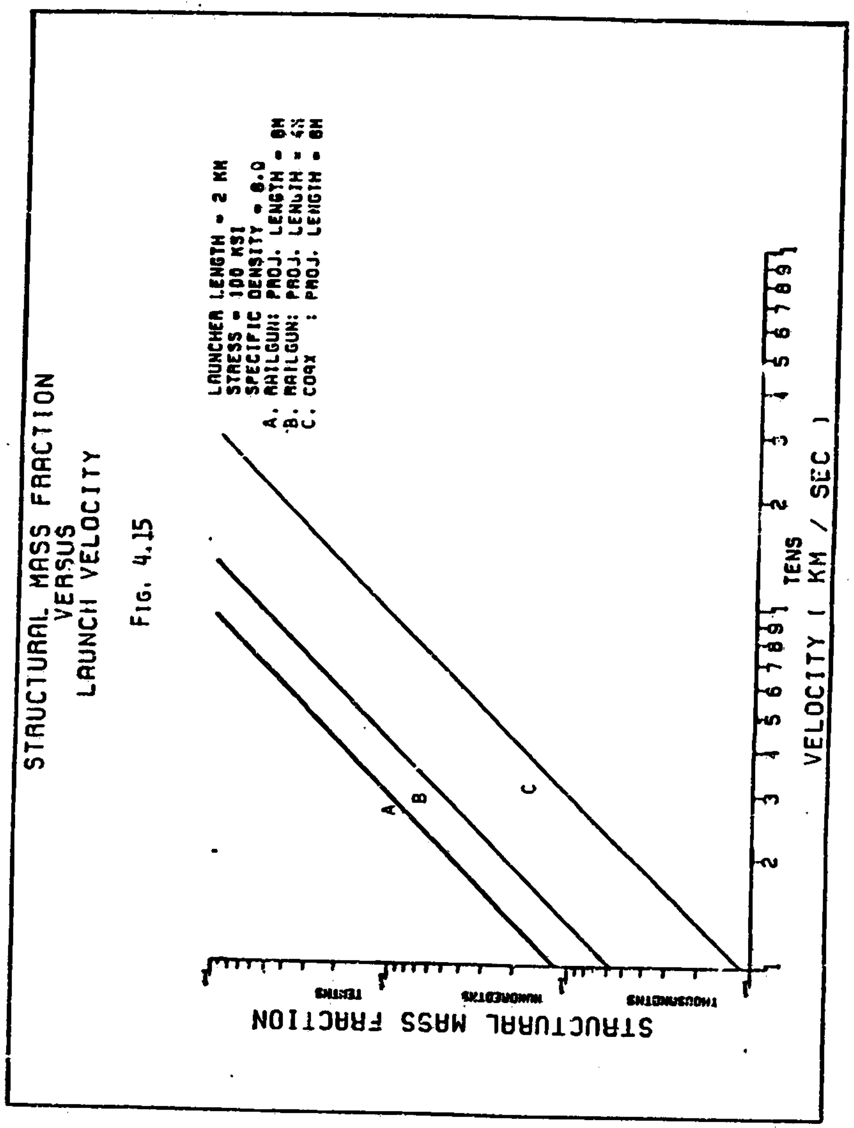
FORCE VERSUS VELOCITY

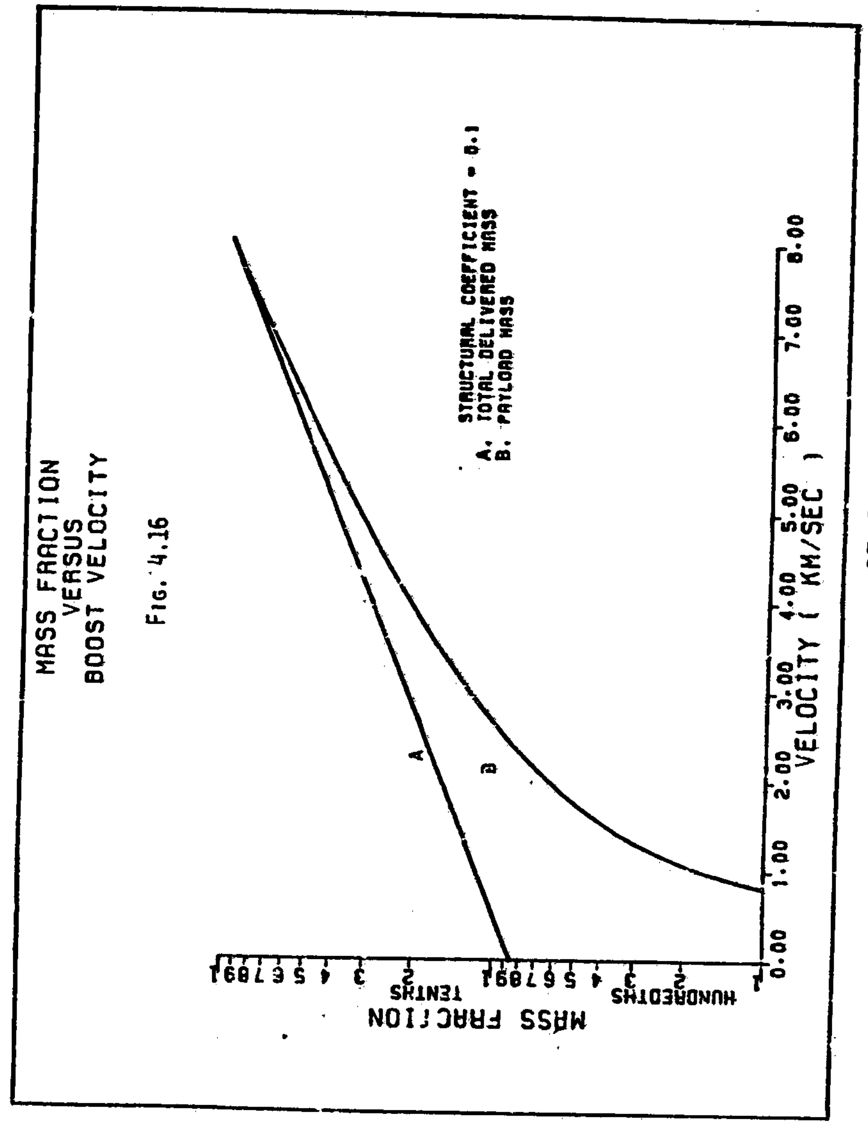
FIG. 4.14





STRUCTURAL MASS FRACTION
VERSUS
LAUNCH VELOCITY

FIG. 4.15



MASS FRACTION
VERSUS
BOOST VELOCITY

FIG. 4.16

Fig. 4.17 (next page) is a plot of ideal rocket efficiency versus velocity, which shows a sharp peak at the velocity which corresponds to the exhaust velocity at the assumed specific impulse of 290 seconds (2.8 km/sec). At this point the reaction mass comes to rest in the inertial frame and absorbs no kinetic energy. The curve shows dramatically the advantage to be gained by a hybrid launch vehicle with an electric first stage capable of reaching only 3 km/sec, as compared to igniting chemical rockets at zero velocity where the efficiency is zero. The hybrid vehicle requires only about one fifth the launch mass of its all-chemical counterpart.

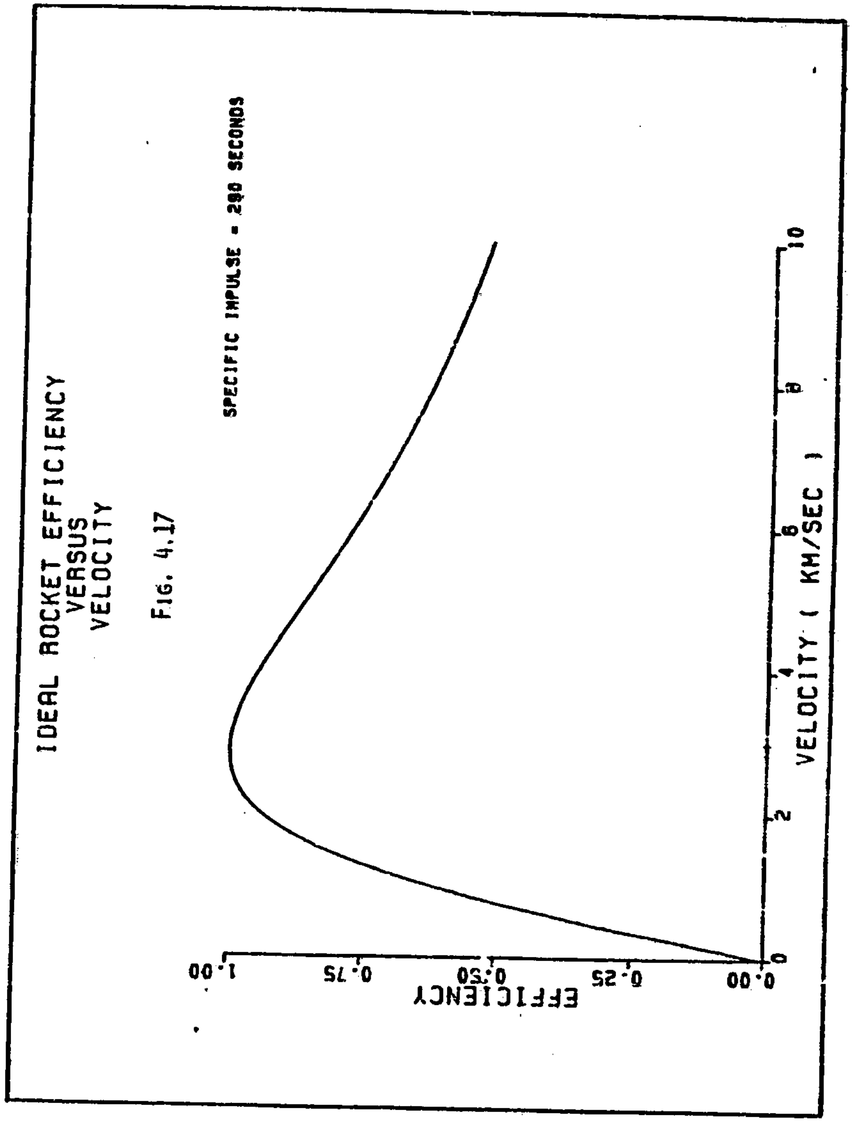


FIG. 4.17

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SECTION 5. ECONOMIC CONSIDERATIONS

Energy is cheap, but the equipment to compress it is expensive. For an intuitive grasp of what is involved, we shall repeat some reference figures.

To launch one ton to earth escape velocity we must store the output energy of a large municipal power plant (1,000 megawatt capacity) for about 1.5 minutes, and release it in about 1.5 seconds. A 60-fold power compression. Although the energy only costs about 50 cents per pound launched, the devices to store that same energy would cost about 5 million dollars, an increase of 8 orders of magnitude.

It is clear that electromagnetic launch technology is heavily capital-intensive. Economics are therefore dominated by several factors which are under our control, subject to certain overriding considerations.

It is important to state these factors.

5.1 ENERGY STORAGE

The amount of necessary energy storage can be minimized by using the smallest possible projectile mass.

Other constraints apply, of course, but energy storage is the dominant cost driving factor. The lower launch energy required for high elevation launches may be a profitable trade-off against the lower construction costs of a more accessible low altitude launcher.

Similarly, shielding requirements may suggest large waste disposal projectiles for the sake of efficiency and better atmospheric penetration. But smaller and less efficient projectiles might prove more expedient, simply because they reduce overall energy storage requirements.

It may also be expedient to use the lowest possible launch velocity, i.e., to use chemical rockets as soon as they become efficient. Hybrid launchers will make sense long before all-electric ones do.

5.2 POWER COMPRESSION AND SWITCHING

The necessary delivered electrical power can be minimized by using the longest possible launcher.

In general the electrical and mechanical loads are reduced as the input power is decreased. Not only will the launcher design be less constrained but the electrical switching devices

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can be of lower power and consequently less costly. Next to the energy storage devices the switches are the most cost intensive and are certainly the most critical in terms of technical risk.

As the launcher power is decreased the energy storage devices can be of low specific power (such as homopolars or batteries) and consequently less expensive. Ultimately, if the power requirements can be reduced sufficiently it is conceivable that energy storage devices can be eliminated almost entirely. The launcher would then operate directly from the prime power source (nuclear power station, power grid, etc.).

5.3 UTILIZATION RATE

The most economical launcher is clearly one which launches the smallest possible projectiles at the highest possible repetition rate 24 hours a day. Opposing constraints include minimum projectile size (shielding, atmospheric penetration, economics of scale) launch windows, and recycling problems.

Since the launch barrel itself is very inexpensive compared to the energy supply, and since these two components do not need to be intimately connected (in the proposed coaxial approach), it will certainly prove expedient to use a multitude of separate launch barrels with a single power compression device. This will speed up the recycling time and will also permit the launching of a wide range of different vehicles for a range of different missions having different launch windows.

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5.4 PROJECTILE SIZE

For atmospheric drag and dynamic stability reasons, the projectile should be long and slender, with length/diameter ratios above 15. Railguns favor short projectiles to reduce launch pressure on the base surface. Coaxial launchers can accommodate long slender projectiles because the launch force can be distributed over a number of rings along the entire projectile length. No sabot is needed for stiffening purposes.

Projectile size will have a lower limit imposed by payload and atmospheric drag considerations. Atmospheric drag permits projectile mass as low as 50 kilograms with reasonable energy and ablation loss. Shielding requirements of nuclear waste impose a larger minimum.

It is justified to ask whether there exists a level of redundancy in drive coils at which reliability is such as to eliminate the need for shielding, particularly if projectiles are so small that an occasional loss over deep ocean water is acceptable.

5.5 LAUNCHER LENGTH

As mentioned above, long launchers require less power compression. They also reduce thrust and thereby drive current and voltage. The longest vertical launch barrel which can be built by present technology (in a drilled well casing) is about 10 kilometers. The technology would have to be stretched to accommodate an access shaft. Launch barrels several times this length could be built on a mountainside, but at an inclination of no more than about 20 degrees. They could also be submerged in deep ocean water near an island launch site.

Long barrels also offer increased opportunity for redundancy in drive coils. In principle it is possible to provide two or even three completely independent launch systems, any one of which is capable of achieving the required launch velocity. This would make the operation extremely reliable and make launching of highly toxic wastes and possibly even human personnel (allowing, of course, for the low acceleration limits) a reasonable endeavor.

SECTION 6. SELECTED REFERENCE DESIGN

Fig. 6.1 is a schematic cross section of the launcher selected as reference design for accomplishing the assigned missions.

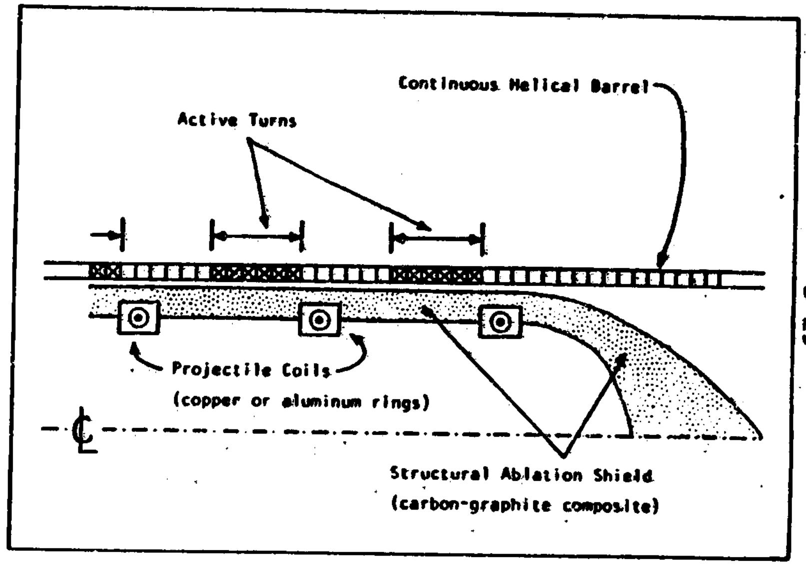

Fig. 6.1 Selected reference design launcher

The projectile coils are copper or aluminum rings spaced at approximately one radius intervals along the entire length of the projectile so as to distribute the thrust uniformly. These rings are imbedded in a carbon-graphite composite which serves as heat shield, ablation shield, friction pad and structural shell.

The drive coils form a continuous winding on the inside of a support barrel, which also serves as evacuated launch tube. The individual turns may be electrically isolated from each other, or they may be connected into a continuous helix. As shown in Fig. 6.1, approximately ten of these individual turns located behind each projectile ring are simultaneously active and thus exert a push on each projectile ring.

These active segments of the drive coil barrel are made to travel in synchronism with the projectile by commutating individual turns of the barrel or helix, by means of solid state devices or other suitable switches. As indicated in Fig. 6.2, the active segment of barrel turns is made to propagate by disconnecting the rearmost turn and connecting a previously inactive turn at the head of the segment. This method of synchronization minimizes the energy to be switched at each commutation step. It has the added advantage that the dM/dz coupling parameter is held constant and near its peak value, which can approach 10 microhenry per meter. The thrust is thus held constant and near its peak value.

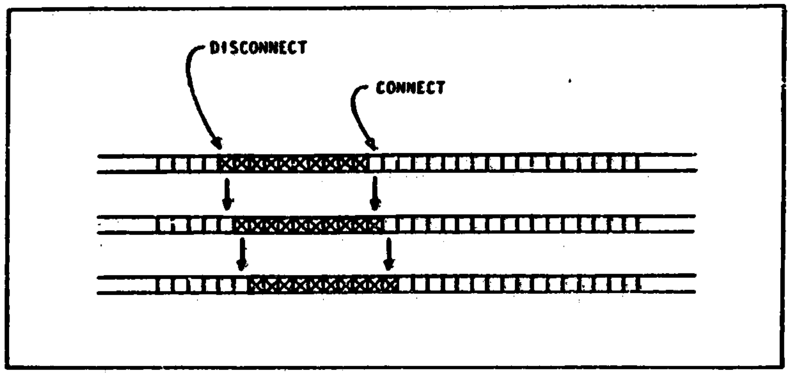

Fig. 6.2 Propagation of active barrel segment

Excitation current is supplied to the projectile rings without physical contact by induction, as will be explained.

Fig. 6.3 defines the critical dimensions of this system, which varies somewhat with individual missions. A typical projectile caliber might be 43 cm = 17 inches, as in the case of the earth orbital launcher for example. At this size, the L/R time constant of the projectile rings is about 20 milliseconds. In other words, a current in the projectile ring will decay to 1/e of its initial value in about 20 milliseconds. This corresponds to a frequency of about 50 Hertz.

Fig. 6.4 shows an idealized current waveform supplied to an active section of drive coil barrel. It is a square wave of somewhat higher frequency than 50 Hertz, say 60 to 100 Hertz. The lower plot shows the waveform of current induced in the projectile rings by this applied square wave drive coil current. There is an initial rise, followed by a gradual decay. The two currents are mutually repulsive, and therefore thrust is exerted on each projectile ring while the two currents flow.

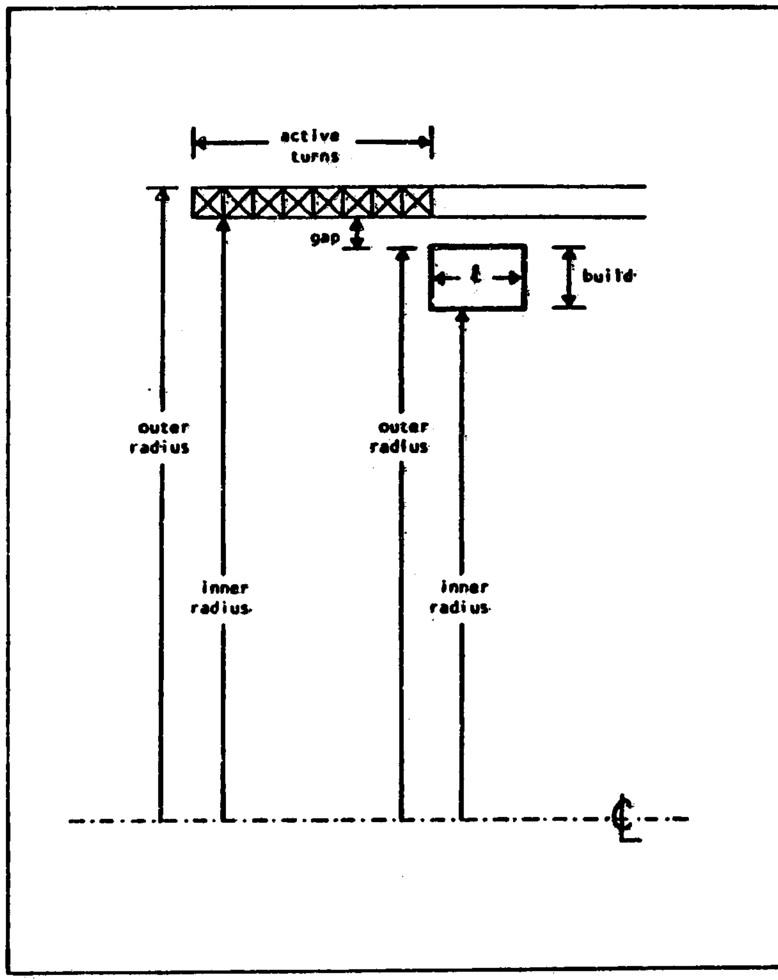

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Fig. 6.3 Dimensions Defined

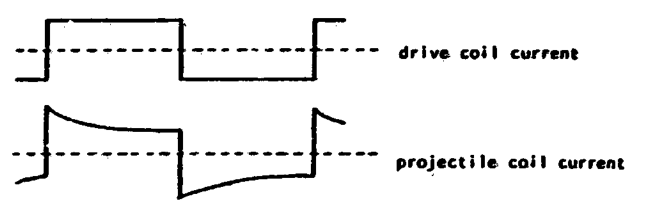

Fig. 6.4 Current waveforms in projectile and drive coils

The projectile is guided by contact between its ablation shield and a liner of similar material which coats the drive coil barrel. However, since the projectile rings always carry an induced current, physical contact can be virtually eliminated by means of thin strips of conducting material applied to the inside surface of the barrel liner. Eddy currents induced in these conducting strips will tend to repel the projectile as contact is approached.

The drive coils are energized by one or several large storage inductors or homopolar generators. The primary store in turn is energized from a powerline or a dedicated set of generators. It should be noted that a storage inductor large enough to store the entire launch energy has an L/R time constant of several minutes, which is long enough to permit charging from an existing powerline without undue resistive energy loss.

Propagation of the magnetic field along the drive coil barrel or helix is assisted by transfer capacitors. These transfer capacitors serve exactly the same function as the capacitor which shunts the breaker points in every automotive ignition distributor since the Model A Ford. They need to store only the commutation energy, which in the case of the selected launcher design amounts to only several kilojoules. The reason this commutation energy is so low is the very tight coupling between each individual turn being commutated, and the remainder of the drive coil barrel. In other words, when an individual current-carrying turn of the barrel is disconnected, most of its magnetic energy is coupled into the remaining active turns, and only the leakage energy needs to be supplied by the commutation capacitor to the next ring to be connected at the head of the active segment. The commutation energy amounts to only about one percent of the energy stored in one active drive coil segment.

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Commutation is accomplished by a switch capable of rapid turn-off, synchronized to operate at or near an induced zero-crossing of the trailing-turn current. There are a variety of switches capable of making and breaking at the required speed, including thyratrons, triggered vacuum gaps, and various solid state devices. Although none of the currently available devices have the combined rating of current, voltage, turn-off time and repetition rate required by some of the designs, commercial devices are within a factor of two or so of the necessary performance specifications. Switching therefore does not appear to be a serious constraint in the context of a ten-year implementation scenario.

This, in general terms, is how a coaxial launcher will operate. It very definitely does not require unrealistic extrapolation beyond present technology and knowledge, nor unrealistic expenditure of capital.

Although many details remain to be implemented, the presently outlined reference design provides sufficient basis for computation of the performance parameters. These are presented in the remainder of this report.

Some comments concerning mission selection

Table 6.1 is a summary of mission requirements as defined in the NASA/LeRC - Battelle study. We wish to comment on certain aspects of this tabulation.

Consider two of the tabulated missions:

Earth Orbital System:
650 kg payload, 6-14 launches/day, 2,500 gee, 7-12 km/s

The railgun projectile proposed for this mission (see Fig. 6.5) contains 2730 kg of payload support structure, in order to permit the high acceleration required for a short launcher. This is an unreasonably high fraction of structure. It also contains 1150 kg of propellant.

Electromagnetic boost - solid rocket vehicles
15,000 kg payload, 3-10 launches/year, 1,000 gee, 2.5 km/s

The launch frequency is too low to justify the investment for such a large payload system.

It seems more expedient to combine these two missions into a more realistic and flexible hybrid launcher using electromagnetic boost to a velocity of about 3 km/sec, followed by solid fuel rockets. Replacing the payload support structure by solid fuel rockets will accomplish the objective, and provide a hybrid launcher suitable for much higher use rate than the one proposed.

Such a hybrid launcher also represents the most logical "first attempt" in the field, and can be recommended in good faith for immediate implementation. A launch velocity of 3 km/sec is well within commercial component capability. Such a launcher can also be made flexible enough to accomplish a broad range of missions and operate at an economically justifiable use-rate.

This one point cannot be overstressed:

In electromagnetic launch technology, the key to economic viability is a high use-rate.

The following sections 7 through 14 summarize the design parameters for the various missions listed in Table 6.1. For reference, a Key (Table 6.2) is included to define all the common variables.

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Fig. 6.5

ESRL

PROJECTILE

— NASA/LeRC — EARTH-TO-SPACE RAIL LAUNCHER FEASIBILITY ASSESSMENT — BATTELLE —

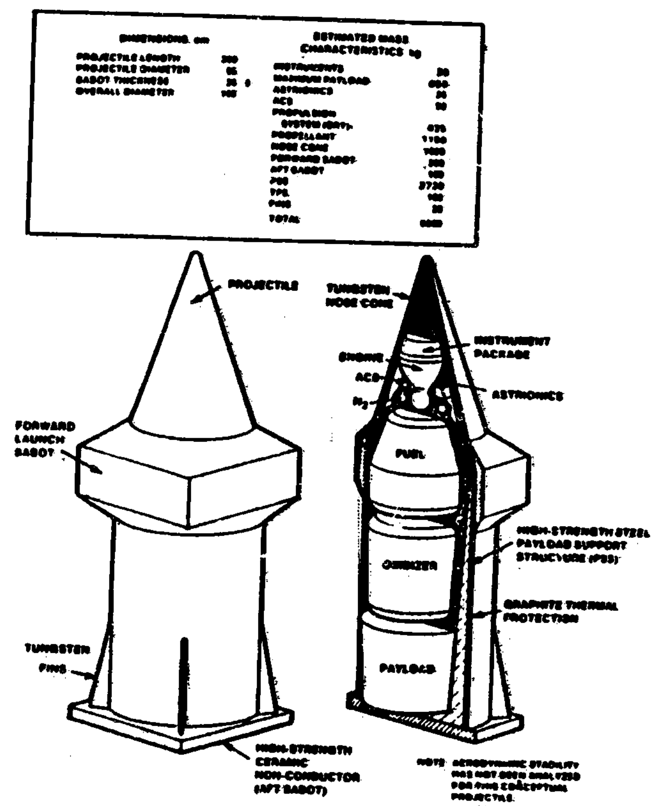

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TABLE 6.1

EML PAYLOAD REQUIREMENTS SUMMARY

NASA/LeRC — SPACE MISSION APPLICATIONS ANALYSIS OF ELECTROMAGNETIC LAUNCHERS — BATTELLE

Mission	Payload Mass (kg)	Launch Frequency	Acceleration Limit (g)	Velocity (km/s)
GROUND-BASED				
• Earth Orbital System	650	6 - 14/day	2,500	7 - 12
• Solar System Escape	250 (HLW)	2/day[(a)]	10,000	20
	285 (TRU)	22/day[(a)]	10,000	20
• Earth Escape	800	1 - 3/yr	10,000	12 - 20
• Suborbital	50 - 2,000	50 - 150/yr	2,500	1 - 5
• Electromagnetic Boost				
- Solid Rocket Vehicle	15,000	3 - 10/yr	20 - 100	2
- TAV/SSTO	900,000	50 - 200/yr	5	0.5
SPACE-BASED				
• Satellite Kick System	5,000	12 - 24/yr	1,000	Up to 2.5
• Nuclear Waste Disposal	1 (HLW)	500/day[(b)]	Limited By Launcher	11

Table 6.2

KEY

v = projectile muzzle velocity
F = average launch force
l = launcher length
t = launch period
U_{KE} = projectile kinetic energy
J_p = projectile coil current density
I_p = projectile coil current
I_d = drive coil current
V_{bemf} = back EMF voltage
U_{comm} = single turn, commutation energy
t_{comm} = commutation period
V_{comm} = commutator voltage
V_R = resistive voltage drop in drive coil
V_{Tot} = total drive coil voltage
U_p = energy dissipated per projectile coil
U_d = energy dissipated per drive coil phase
η = accelerator efficiency
U_{cap} = total capacitive commutation energy

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SECTION 7

MISSION DESCRIPTION

Mission: Ground Based Earth Orbital System

Nominal Launch Parameters

Mass = 650 kg

Velocity = 7-12 km/s

Acceleration = 2500 g's

PROJECTILE

Mass Breakdown:

Instruments =	30 kg
Payload =	650 kg
Astrionics =	25 kg
ACS =	50 kg
Propulsion System =	425 kg
Propellant =	1150 kg
Nose Cone =	110 kg
Payload Support Structure =	200 kg
Thermal Protection System =	200 kg
Fins =	20 kg
Coils =	140 kg
TOTAL =	3000 kg

diameter = 43 cm

length = 650

volume = 863,000 cm^3

average density = 2.7 gm/cm^3

ELECTRICAL CONFIGURATION

40 total active phases

Projectile Coils:

40 rings, 16 cm center to center spacing

material = copper

inside radius = 19.5 cm

outside radius = 21.5 cm

axial length = 3 cm

single turn inductance L_P = 774 nH

single turn resistance R_P = 36.5 uΩ

time constant τ_P = L/R = .021 sec

thermal current integral limit $J^2t = 5 \times 10^8\ A^2s/cm^4$

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 21.8 cm

outside radius = 22.8 cm

active axial length = 8 cm

number of turns in active drive coil section = 10

active drive coil inductance L_D = 70.1 μH

active drive coil resistance R_D = 2.98 mΩ

time constant τ_D = .0241 sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 4.75 μH/m

commutating effective single end turn inductance = 722 nH

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SYSTEM PARAMETERS

Design Parameters

v	F	l	t	U_{KE}	J_P	I_P	I_d	V_{bemf}
km/s	MN	km	sec	GJ	kA/cm^2	kA	kA	kV
7	73.50	1	.286	73.50	41.81	250.9	154.2	83.4
8	"	1.3	.327	96	39.10	234.6	164.9	89.14
9	"	1.65	.367	121.50	36.91	221.5	174.6	94.69
10	"	2.04	.408	150.0	35.01	210.0	184.2	99.75
11	"	2.47	.449	181.5	33.37	200.2	193.2	104.6
12	"	2.94	.490	216.00	31.94	191.7	201.8	109.3

Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tot}	U_P	U_d	η	U_{cap}
km/s	kJ	µs	kV	kV	kV	MJ	MJ	%	kJ
7	8.58	1.14	48.7	0.5	132.6	0.66	20.2	98.9	343
8	9.82	1.0	59.6	0.5	149.2	0.66	26.47	98.9	392.8
9	11.0	.89	70.9	0.5	166.4	0.66	33.31	98.9	440.0
10	12.3	.80	83.47	0.5	183.7	0.66	41.21	98.9	492
11	13.5	.73	96.1	0.6	201.3	0.66	49.89	98.9	540
12	14.7	.67	109.3	0.6	219.2	0.66	59.4	98.9	588

Comments

- The projectile's length to diameter ratio has been increased over the reference design to accommodate several projectile coils. This has the added benefit of greatly reducing structural support mass requirements.
- Although the switching loads are high, (up to I_D = 201 kA and V_{Tot} = 219 kV), they are not more than a factor of about two beyond present devices.

SECTION 8

MISSION DESCRIPTION

Mission: Solar System Escape, High Level Waste

Nominal Launch Parameters

Mass = 250 kg

Velocity = 20 km/sec

Acceleration = 10,000 g's

PROJECTILE

Mass Breakdown:

Waste Form =	250 kg
Shield/Container/Coils =	1140 kg
Nose Cone =	440 kg
TPS =	25 kg
Instruments =	50 kg
Fins =	10 kg
TOTAL =	1915 kg

diameter = 51 cm

length = 170 cm

volume = 2.66 x 105 cm^3

average density = 7.2 gm/cm^3

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ELECTRICAL CONFIGURATION

5 total active phases

Projectile Coils:

5 rings, 20 cm center to center spacing

material = copper

inside radius = 25.5 cm

outside radius = 19.5 cm

axial length = 8 cm

single turn inductance L_p = 594 nH

single turn resistance R_p = 5.0 uΩ

time constant $\tau_p = L/R$ = 0.119 sec

thermal current integral limit J^2t = 5×10^8 A^2s/cm^4

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 26 cm

outside radius = 27 cm

active axial length = 16 cm

number of turns in active drive coil section = 16

active drive coil inductance L_D = 175 μH

active drive coil resistance R_D = 4.53 mΩ

time constant τ_D = .039 sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 2.33 μH/m

commutating effective single end turn inductance = 1.02 μH

SYSTEM PARAMETERS

Design Parameters

v	F	l	t	U_{KE}	J_P	I_P	I_d	V_{bemf}
km/s	MN	km	sec	GJ	kA/cm^2	kA	kA	kV
20	188	20	0.204	383	49.5	2380	424	1770

Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tot}	U_P	U_η	η	U_{cap}
km/s	kJ	μs	kV	kV	kV	MJ	MJ	%	kJ
20	92	0.5	433	1.9	2210	5.8	166	95 8	459

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Comments

- Only the high level waste system was examined. The heavy shielding masses (12 cm wall thickness) precludes increasing the projectile length to diameter ratio. Essentially, one is reduced to accelerating shield mass rather than waste.
- Although the coaxial approach is possible in principle, the high switching loads make it impractical for the forseeable future.
- The TRU waste mission is not appreciably different.

SECTION 9

MISSION DESCRIPTION

Mission: Ground-Based Earth Escape

Nominal Launch Parameters

Mass = 600 kg

Velocity = 12-20 km/sec

Acceleration = 10,000 g

PROJECTILE

Mass Breakdown:

Payload =	600 kg
Nose Cone =	50 kg
Thermal Shield/Support Structure =	60 kg
Coils =	40 kg
TOTAL =	750 kg

diameter = 28 cm

length = 450 cm

volume = 2.46×10^5 cm^3

average density = 2.7 gm/cm^3

ELECTRICAL CONFIGURATION

30 total active phases

Projectile Coils:

30 rings, 14 cm center to center spacing

material = aluminum

inside radius = 12 cm

outside radius = 14 cm

axial length = 3 cm

single turn inductance L_p = 418 nH

single turn resistance R_p = 38.5 $\mu\Omega$

time constant $\tau_p = L/R$ = .011 sec

thermal current integral limit $J^2t = 2 \times 10^8$ A^2s/cm^4

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 14.3 cm

outside radius = 14.8 cm

active axial length = 7 cm

number of turns in active drive coil section = 10

active drive coil inductance L_D = 41.4 μH

active drive coil resistance R_D = 4.44 $m\Omega$

time constant τ_D = .009 sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 3.37 $\mu H/m$

commutating effective single end turn inductance = 517 nH

SYSTEM PARAMETERS

Design Parameters

v	F	l	t	U_{KE}	J_P	I_P	I_d	V_{bemf}
km/s	MN	km	sec	GJ	kA/cm^2	kA	kA	kV
12	73.5	.734	.122	54.00	40.5	243	299	98.3
15	"	1.148	.153	84.38	36.2	217	335	109.7
18	"	1.653	.184	121.5	33.0	198.0	367	120.2
20	"	2.041	.204	150.0	31.3	188	387	126.6

Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tot}	U_P	U_d	η	U_{cap}
km/s	kJ	μs	kV	kV	kV	MJ	MJ	%	kJ
12	23	0.583	132	1.3	232	0.277	48.4	97.4	690
15	29.0	0.467	185.4	1.5	297	0.277	76.2	97.4	870
18	34.8	0.389	243.8	1.6	366	0.277	110	97.4	1044
20	38.7	0.350	285.7	1.7	414	0.277	135.7	97.4	1161

Comments

- The high launch velocity again requires high performance switches.

SECTION 10A

MISSION DESCRIPTION

Mission: Ground-Based Sub-orbital

Nominal Launch Parameters

Mass = 25 kg

Velocity = 1-5 km/s

Acceleration = 2500 g's

PROJECTILE

Mass Breakdown:

payload =	25.0 kg
nosecone =	1.8 kg
coils =	2.7 kg
thermal shield/support structure =	4.0 kg
fins =	0.5 kg
TOTAL =	34.0 kg

diameter = 9.2 cm

length = 140 cm

volume = 12,600 cm^3

average density = 2.7 gm/cm^3

ELECTRICAL CONFIGURATION

15 total active phases

Projectile Coils:

15 rings, 9 cm center to center spacing

material = aluminum

inside radius = 4.35 cm

outside radius = 4.6 cm

axial length = 3 cm

single turn inductance L_p = 110 nH

single turn resistance R_p = 22.4 $\mu\Omega$

time constant τ_p = L/R = .0049 sec

thermal current integral limit $J^2t = 2 \times 10^8$ A^2s/cm^4

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 5.3 cm

outside radius = 5.8 cm

active axial length = 4 cm

number of turns inactive drive coil section = 4

active drive coil inductance L_D = 2.05 μH

active drive coil resistance R_D = 474 $\mu\Omega$

time constant τ_D = .0043 sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 1.81 μH/m

commutating effective single end turn inductance = 144 nH

SYSTEM PARAMETERS

Design Parameters

v	F	l	t	U_{KE}	J_P	I_p	I_d	V_{bemf}
km/s	kN	m	sec	MJ	kA/cm^2	kA	kA	kV
1	833	20.41	.0408	17	210	157.5	48.7	1.14
2	833	81.6	.0816	68	148	111.4	68.85	1.61
5	833	510	.204	425	93.9	70.45	108.9	2.55

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Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tot}	U_p	U_d	η	U_{cap}
km/s	kJ	μs	kV	kV	kV	kJ	kJ	*	kJ
1	.171	10	.351	.023	1.514	22.7	45.87	94.3	2.57
2	.342	5	.994	.033	2.637	22.7	183.3	95.7	5.13
5	.856	2	3.93	.052	6.532	22.7	1147	99.7	12.84

* For comments, see end of section 10C

SECTION 10B

MISSION DESCRIPTION

Mission: Ground-Based Sub-Orbital

Nominal Launch Parameters

Mass = 160 kg

Velocity = 1 - 5 km/sec

Acceleration = 2,500 g's

PROJECTILE

Mass Breakdown:

Payload =	160	kg
Nose Cone =	12.2	kg
Thermal Shield/Support Structure =	20	kg
Coils =	31.8	kg
Fins =	1	kg
TOTAL =	225	kg

diameter = 20 cm

length = 280 cm

volume = 83,300 cm^3

average density = 27 gm/cm^3

ELECTRICAL CONFIGURATION

20 total active phases

Projectile Coils:

20 rings, 25 cm center to center spacing

material = copper

inside radius = 9 cm

outside radius = 10 cm

axial length = 3 cm

single turn inductance L_p = 294 nH

single turn resistance R_p = 33.8 $\mu\Omega$

time constant $\tau_p = L/R$ = .0087 sec

thermal current integral limit J^2t = 5×10^8 A^2s/cm^4

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 10.2 cm

outside radius = 11.2 cm

active axial length = 7 cm

number of turn in active drive coil section = 7

active drive coil inductance L_D = 12.6 μH

active drive coil resistance R_D = 800 $\mu\Omega$

time constant τ_D = .016 sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 2.63 $\mu H/m$

commutating effective single end turn inductance = 264 nH

SYSTEM PARAMETERS

Design Parameters

v	F	l	t	U_{KE}	J_P	I_P	I_d	V_{bemf}
km/s	MN	m	sec	GJ	kA/cm^2	kA	kA	kV
1	5.513	20.41	.0408	0.1125	111	332.1	45.09	6.113
2	"	81.6	.0816	0.950	78.3	234.8	63.76	8.65
5	"	510	.204	2,813	49.5	148.5	100.8	13.67

Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tot}	U_P	U_d	η	U_{cap}
km/s	kJ	μs	kV	kV	kV	kJ	kJ	%	kJ
1	0.268	10	0.5944	.036	6.74	152	66.4	96.3	5.36
2	0.5366	5	1.681	.051	10.38	152	265.4	98.2	10.73
5	1.342	2	6.65	.081	20.40	152	1659	98.7	26.84

* For comments, see end of section 10C

SECTION 10C

MISSION DESCRIPTION

Mission: Ground-Based Sub-orbital

Nominal Launch Parameters

Mass = 1000 kg

Velocity = 1-5 km/s

Acceleration = 2500 g's

PROJECTILE

Mass Breakdown:

payload =	1000 kg
nosecone =	80 kg
thermal shield/support structure =	50 kg
coils =	65 kg
fins =	5 kg
TOTAL =	1200 kg

diameter = 32 cm

length = 550 cm

volume = 4.44×10^5 cm^3

average density = 2.7 gm/cm^3

ELECTRICAL CONFIGURATION

25 total active phases

Projectile Coils:

25 rings, 20 cm center to center spacing

material = copper

inside radius = 15 cm

outside radius = 16 cm

axial length = 3 cm

single turn inductance L_p = 573 nH

single turn resistance R_p = 55.2 $\mu\Omega$

time constant $\tau_p = L/R$ = .014 sec

thermal current integral limit $J^2t = 5 \times 10^8$ A^2s/cm^4

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 16.2 cm

outside radius = 17.2 cm

active axial length = 10 cm

number of turns in active drive coil section = 10

active drive coil inductance L_D = 42.6 μH

active drive coil resistance R_D = 1.68 mΩ

time constant τ_D = .025 sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 3.68 μH/m

commutating effective single end turn inductance = 551 nH

SYSTEM PARAMETERS

Design Parameters

v	F	l	t	U_{KE}	J_P	I_P	I_d	V_{bemf}
km/s	MN	m	sec	GJ	kA/cm^2	kA	kA	kV
1	29.4	20.4	.0408	0.6	111	332.1	96.23	12.22
2	29.4	81.6	.0816	2.4	78.3	234.8	130.1	17.28
5	29.4	510	.204	15.4	49.5	148.5	215.2	27.32

Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tot}	U_P	U_d	η	U_{cap}
km/s	kJ	μs	kV	kV	kV	MJ	MJ	%	kJ
1	2.55	10	2.65	0.161	15.03	0.248	0.634	90.5	63.75
2	5.11	5	7.495	0.228	25.00	0.248	2.534	97.2	127.8
5	12.76	2	29.65	0.361	57.33	0.248	15.84	97.5	319.0

Comments

- The payload range of 25-1000 kg has been broken up into 3 specific designs at 25, 160 and 1000 kg.
- All of these designs can be done with present state of the art technology.

SECTION 11

MISSION DESCRIPTION

Mission: Electromagnetic Boost - Solid Rocket Vehicle

Nominal Launch Parameters

Mass = 15,000 kg

Velocity = 2 km/s

Acceleration = 100 g's

PROJECTILE

Mass Breakdown:

Total mass = Payload and Rocket Structure/Coils and Propellant

diameter = 80 cm

length = 16 m

volume = 8.0×10^6 cm^3

average density = 1.9 gm/cm^3

Rocket Specifications:

Propellant specific impulse = 300 sec

Exhaust velocity = 2940 m/s

Structural coefficient = 0.0867 which is equal to:

$$\frac{\text{mass structure}}{\text{mass structure + mass propellant}}$$

ELECTRICAL CONFIGURATION

80 total active phases

Projectile Coils:

80 rings, 20 cm center to center spacing

material = aluminum

inside radius = 38 cm

outside radius = 40 cm

axial length = 6 cm

single turn inductance L_P = 1.55 μH

single turn resistance R_P = 57.9 $\mu\Omega$

time constant $\tau_P = L/R$ = 0.027 sec

thermal current integral limit $J^2t = 2 \times 10^8$ A^2s/cm^4

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 40.2 cm

outside radius = 41.2 cm

active axial length = 10 cm

number of turns inactive drive coil section = 10

active drive coil inductance L_D = 148 μH

active drive coil resistance R_D = 4.35 $m\Omega$

time constant τ_D = 0.034 sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 7.54 $\mu H/m$

commutating effective single end turn inductance = 1.4 μH

SYSTEM PARAMETERS

Design Parameters

v	F	l	t	U_{KE}	J_P	I_P	I_d	V_{bemf}
km/s	MN	km	sec	GJ	kA/cm^2	kA	kA	kV
2	14.7	2.04	2.04	30	3.08	40	60.9	2.63

Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tot}	U_P	U_d	η	U_{cap}
km/s	kJ	μs	kV	kV	kV	MJ	MJ	%	kJ
2	2.61	5	8.6	0.27	22.8	0.2	16.5	95.7	209

Comments

- This combination of chemical rocket boost and electromagnetic boost appears to be the most reasonable of all the missions. There is nothing unique about this particular payload and total mass. This system could easily be scaled for larger or smaller net projectile masses with no increase in technical difficulty. By way of example, the payload mass has been traded off against propellant mass for the given total projectile mass of 15,000 kg. Figure 11.1 shows the variation in final velocity versus paylod mass for single, two and three stage solid rockets.

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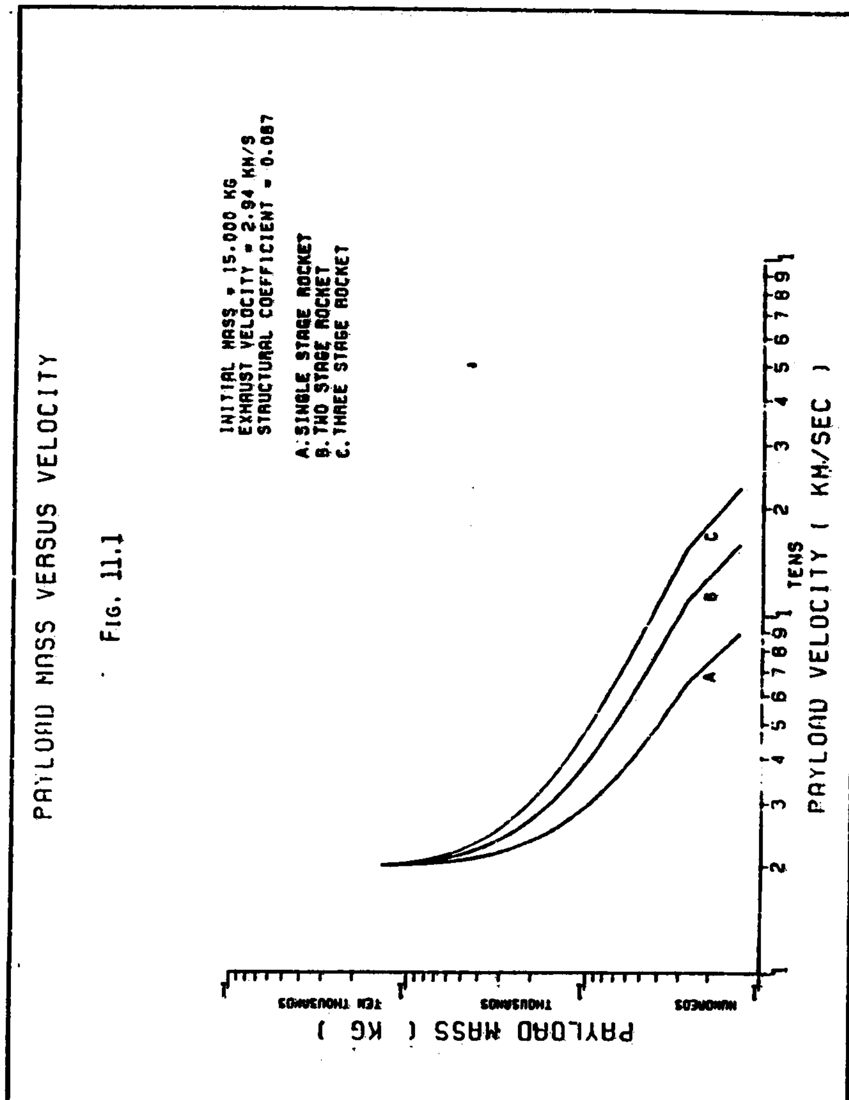


Fig. 11.1

SECTION 12

MISSION DESCRIPTION

Mission: Electromagnetic Boost TAV/SSTO

Nominal Launch Parameters

Mass = 900,000 kg

Velocity = 500 m/s

Acceleration = 5 g's

PROJECTILE

Mass Breakdown:

payload =	900,000 kg
coils =	4,060 kg
sled =	16,500 kg
TOTAL =	921,000 kg

diameter = 5.2 m

length = 21 m

volume = 4.5×10^8 cm^3

average density = 2 gm/cm^3

ELECTRICAL CONFIGURATION

100 total active phases

Projectile Coils:

100 rings, 20 cm center to center spacing

material = copper

inside radius = 248 cm

outside radius = 250 cm

axial length = 3 cm

single turn inductance L_P = 16 μH

single turn resistance R_P = 215 $\mu\Omega$

time constant $\tau_P = L/R$ = 0.074 sec

thermal current integral limit $J^2t = 5 \times 10^8$ A^2s/cm^4

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 250.5 cm

outside radius = 252.5 cm

active axial length = 10 cm

number of turns in active drive coil section = 10

active drive coil inductance L_D = 1.46 mH

active drive coil resistance R_D = 17 $m\Omega$

time constant τ_D = 0.086 sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 50.6 $\mu H/m$

commutating effective single end turn inductance = 6.88 μH

SYSTEM PARAMETERS

Design Parameters

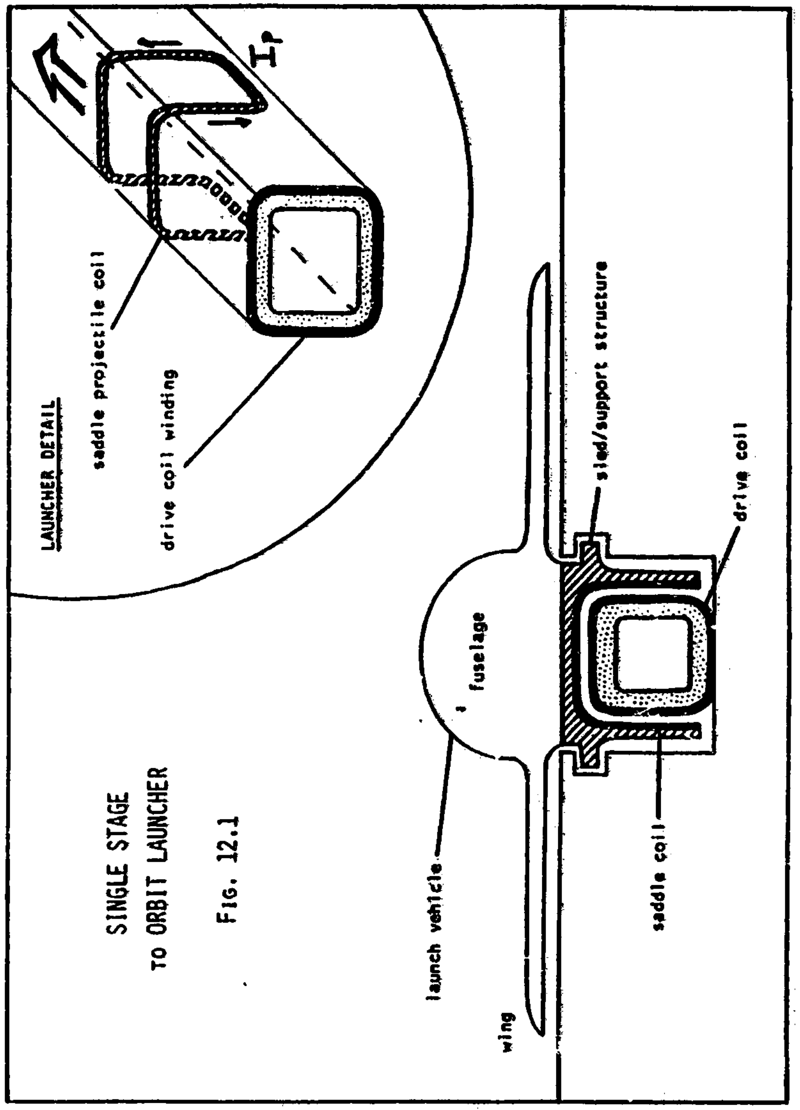
v	F	l	t	U_{KE}	J_P	I_P	I_d	V_{bemf}
km/s	MN	km	sec	GJ	kA/cm^2	kA	kA	kV
2	45.1	2.55	10.2	115.1	7	42	21.3	10.6

Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tct}	U_P	U_d	η	U_{cap}
km/s	kJ	μs	kV	kV	kV	MJ	MJ	%	kJ
2	1.56	20	3.66	0.36	14.6	3.9	74.8	93.5	156

Comments

- This system uses a split saddle coil launcher. In this configuration (see Fig. 12.1), the drive coil is supported in a trough the length of the launcher. The moving armature consists of a linear array of split saddle coils. These coils are attached to a sled support structure which transmits the launch force to the vehicle. Appropriate sled guides are used to support off-axis loads.



SINGLE STAGE
TO ORBIT LAUNCHER

Fig. 12.1

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SECTION 13

MISSION DESCRIPTION

Mission: Space Based Satellite Kick System

Nominal Launch Parameters

Mass = 5000 kg

Velocity = 2.5 km/sec

Acceleration = 1000 g's

PROJECTILE

Mass Breakdown:

Payload Mass = 5000 kg

Coil Mass = 100 kg

Support Structure = 300 kg

Total Mass = 5400 kg

diameter = 56 cm

length = 8.1 m

volume = $2 \times 10^6 cm^3$

average density = $2.7 gm/cm^3$

ELECTRICAL CONFIGURATION

27 total active phases

Projectile Coils:

27 rings, 30 cm center to center spacing

material = aluminum alloy

inside radius = 26 cm

outside radius = 28 cm

axial length = 4 cm

single turn inductance $L_P = 1.05\ \mu H$

single turn resistance $R_P = 90.1\ \mu\Omega$

time constant $\tau_P = L/R = .01165$ sec

thermal current integral limit $J^2t = 2 \times 10^8\ A^2s/cm^4$

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 28.5 cm

outside radius = 29.5 cm

active axial length = 10

number of turns in active drive coil section = 10

active drive coil inductance $L_D = 93.3\ \mu H$

active drive coil resistance $R_D = 3.1\ m\Omega$

time constant $\tau_D = .03$ sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 5.22×10^{-6} H/m

commutating effective single end turn inductance = $0.987\ \mu H$

SYSTEM PARAMETERS

Design Parameters

v	F	l	t	U_{KE}	J_p	I_p	I_d	V_{bemf}
km/s	MN	km	sec	GJ	kA/cm2	kA	kA	kV
2.5	53	0.319	0.255	16.88	28	224	168	29.2

Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tot}	U_p	U_d	η	U_{cap}
km/s	kJ	μs	kV	kV	kV	MJ	MJ	%	kJ
2.5	13.9	4	15.5	0.52	45.2	1.15	22.3	96.4	375

SECTION 14

MISSION DESCRIPTION

Mission: Space Based Nuclear Waste Disposal

Nominal Launch Parameters

Mass = 1000 gm

Velocity = 11 km/s

Acceleration = 10,000 g's

PROJECTILE

Mass Breakdown:

waste form = 1000 gm

coils = 225 gm

glass coating = 275 gm

TOTAL = 1500 gm

diameter = 4.5 cm

length = 12 cm

volume = 191 cm^3

average density = 7.9 gm/cm3

ELECTRICAL CONFIGURATION

4 total active phases

Projectile Coils:

4 rings, 3.7 cm center to center spacing

material = copper

inside radius = 1.75 cm

outside radius = 2.25 cm

axial length = 1 cm

single turn inductance L_p = 48 nH

single turn resistance R_p = 42.7 $\mu\Omega$

time constant τ_p = L/R = 0.0011 sec

thermal current integral limit $J^2t = 5 \times 10^8$ A^2/cm^4

Drive Coils:

one single layer of rectangular copper alloy wire

inside radius = 2.4 cm

outside radius = 2.9 cm

active axial length = 2 cm

number of turns inactive drive coil section = 4

active drive coil inductance L_D = 902 nH

active drive coil resistance R_D = 595 $\mu\Omega$

time constant τ_D = 0.0015 sec

Coupling Parameters:

effective mutual inductance gradient of drive coil segment and projectile coil = 1.95 μH/m

commutating effective single end turn inductance = 53.7 nH

SYSTEM PARAMETERS

Design Parameters

v	F	l	t	U_{KE}	J_P	I_P	I_d	V_{bemf}
km/s	kN	m	sec	MJ	kA/cm^2	kA	kA	kV
11	147	617	0.112	90.8	66.8	33.4	141	2.87

Performance Parameters

v	U_{comm}	t_{comm}	V_{comm}	V_R	V_{Tot}	U_P	U_d	η	U_{cap}
km/s	kJ	μs	kV	kV	kV	kJ	MJ	%	kJ
11	0.53	0.45	8.3	0.08	11.2	5.3	1.33	94.4	2.1

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Comments

- Switch loads are very conservative due to the chosen acceleration limit of 10,000 g's. Acceleration could be increased to 20,000 - 30,000 g's without much extrapolation of present technology.
- The waste form pellet has been modified to accommodate multiple projectile coils (Fig. 14.1).

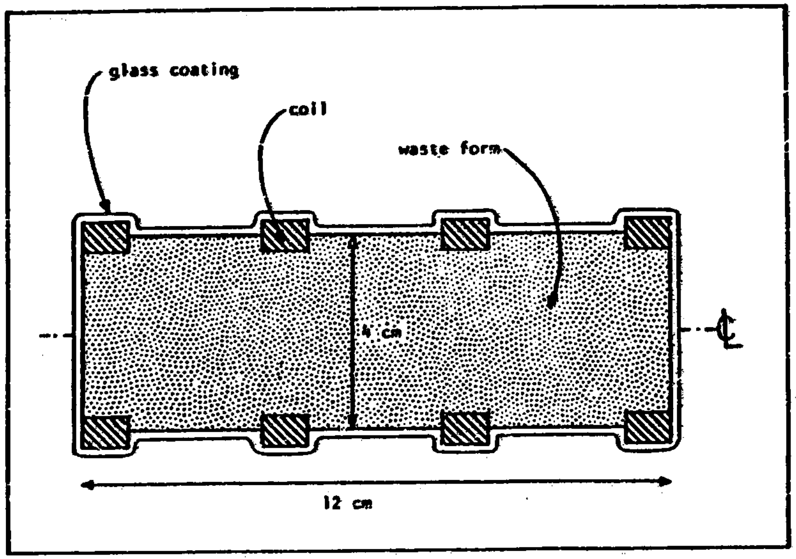

Fig. 14.1 Modified Waste Pellet Projectile

APPENDIX E

AUGUST 1983

FREQUENCY CONTROLLED COIL DRIVER

Report Submitted By

COLLABORATIVE PLANNERS, INC.

Work performed under

Subcontract 4610 - 23 (U of P)
Contract NAS 3-22662
with NASA-Lewis

I INTRODUCTION

At a recent meeting at Battelle's Columbus (in June 1983), two types of electromagnetic launchers proposed to NASA for possible space applications were reviewed. The present report contains background information as well as the recommended design parameters for the Collaborative Planners' scheme proposed to meet the requirements of the two missions specified in Battelle's study.

To fix the ideas, the study calls for a hybrid EM launcher/rocket where the EM launcher will replace the first stage of a chemical launch vehicle. In addition, an Earth -Orbital launcher is also required. The specifications for the two launchers are summarized below.

	EARTH ORBITAL	HYBRID
Maximum Acceleration:	2500 g	100 g
Launch Velocities:	5-12 km/s	1-2 km/s
Launch Elevation:	20° from horizontal	40°- 90° from Horizontal
Projectile Mass:	600 Kg/6500 kg	15,000 kg
Launch Frequency:	8/day	variable

On the basis of aerodynamic considerations, the projectile is given a "telephone-pole" - like shape with approximate overall dimensions that are 3.60 m in length and with a diameter of 0.90 m.

The report is divided in four parts which treat in succession the procedure used to optimize the magnetic forces, the configuration of the electromagnetic launch system, the method employed to find the design parameters and finally the evaluation of these parameters for the two systems needed.

II FORCE OPTIMIZATION

It is well-known from classical electromagnetic theory that the force between two coaxial cylindrical coils (Fig. 1) C_1 and C_2 in which circulate currents I_1 and I_2 respectively is given by the expression

$$F = I_1 I_2 \frac{\partial M}{\partial s} \qquad (1)$$

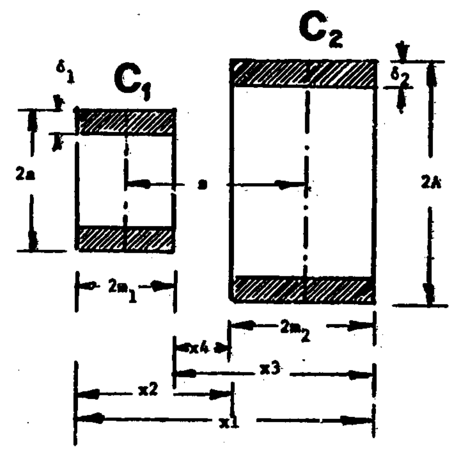

Fig. 1 Notation Used in Text

In the above equation, M stands for the mutual inductance between the two coils and s is the relative distance between their planes of symmetry. The quantity $\frac{\partial M}{\partial s}$ changes sign as the location C_1 with respect to C_2 is altered from the one shown in Fig. 1, to another one in which C_1 is to the right of C_2 i.e., when

one chooses the axis of C_2 as reference, s goes from a negative to a positive value.

If one imagines the coil C_2 to remain fixed while C_1 to move, then in view of $\frac{\partial M}{\partial s}$ reversing its sign as s goes from a positive to a negative value, it is clear that the product $I_1 I_2$ too, must change its polarity in order for the direction of the repulsion force on C_1 to remain unchanged. For reasons to be explained later, it is advantageous to keep the direction of the current I_1 unaltered (i.e. d.c. current), and to make I_2 reverse its sign with s.

The mutual inductance M is a complicated function of x_1, x_2, x_3, x_4, a, m_1, m_2, s and A. The meaning of these symbols is easily found by inspection of Fig. 1. The functional dependence of M on these qualities has been given in tabular form by Grover[1].

In order to develop a simpler computer program that will yield the relative dimensions for optimum force the expression for the mutual as given by Grover (Ref. 1) was rewritten in the form

$$M = 0.002 \frac{\pi^2}{4m_1} a^2 N_1 N_2 f \quad (\mu \text{ Henries}) \qquad (2)$$

where N_1, N_2 are the number of turns of C_1 and C_2 and $f = f(x_1/m_2, x_2/m_2, x_3/m_2, x_4/m_2, m_1/m_2, A/m_2, s/m_2)$. One now locates the values of $s/m_2 = \zeta$ at which $\frac{\partial M}{\partial s} = \max$. These maxima as a function of $m_1/m_2 = \xi$ and $A/m_2 = \beta$ are plotted in Fig. 2. This figure is most valuable in that it yields at once the normalized force i.e. the force per unit turn, per unit current for specific values of relative lengths and radii of the coils.

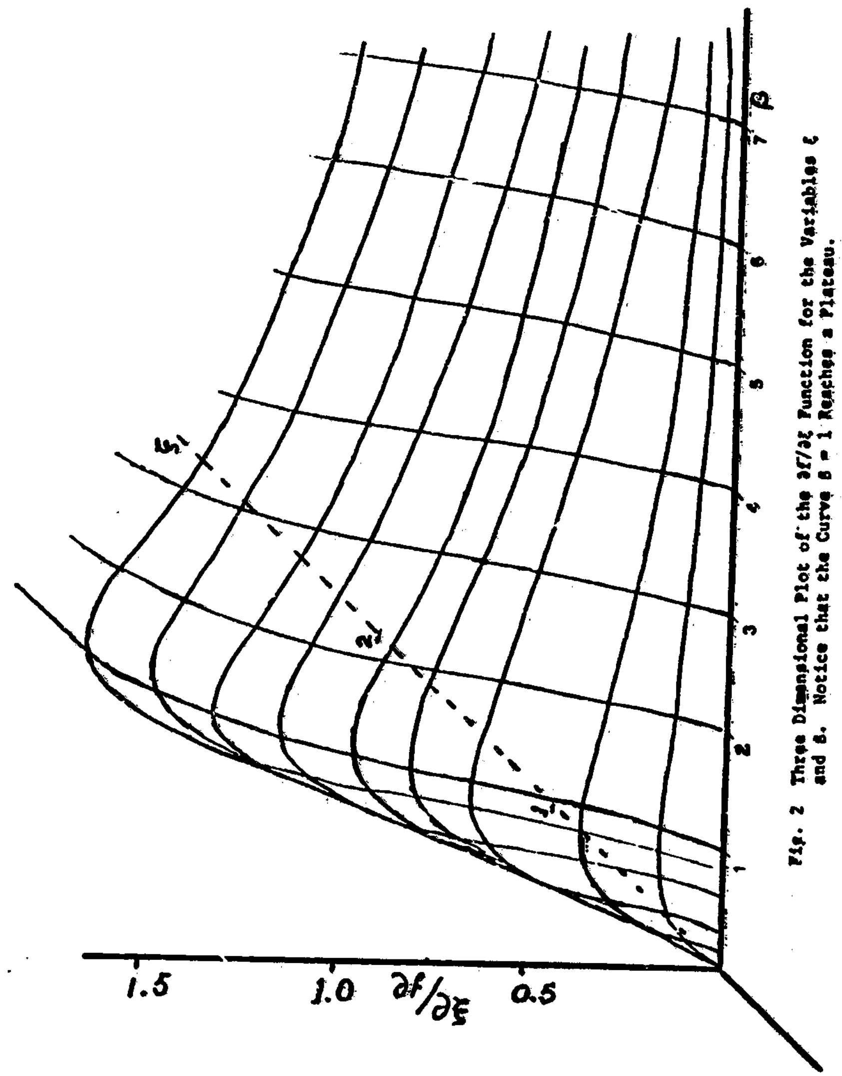

Fig. 2 Three Dimensional Plot of the $\partial f/\partial \xi$ Function for the Variables ξ and β. Notice that the Curve $\beta = 1$ Reaches a Plateau.

formally, the maximum force per coil pair per unit current is:

$$0.001 \frac{\pi^2}{2} a^2 \frac{N_1 N_2}{m_1 m_2} \left.\frac{df}{d\zeta}\right)_{max} \times 10^6 \text{ Newtons/amp}^2$$

In the above expression a is expressed in cms and $\left.\frac{df}{d\zeta}\right)_{max}$ is as shown in the aforementioned Fig. 2.

III CONFIGURATION OF LAUNCHER SYSTEM

The general configuration of the electromagnetic launcher is shown schematically in Fig. 3 below. A series of coils C_1, C_2,.....C_n are arranged on a linear track. These stationary coils will be referred to as the driver coils. The stationary coils form a long string that stretches to approximately 2 km.

Inside the chain C_1, C_2,... another set of a much smaller number of coils D_1, D_2,...D_q which are rigidly attached to the projectile are nested inside the C coils. The D coils will be referred to as the driven coils. The excitation current for the D coils is fed to these coils through brushes and guide rails. As the projectiles - and the D coils - pick up speed, the mechanical stability of the brushes worsens. For this reason, it is essential to maintain the current density in the brushes at a low safe value in the range commonly used in conventional electrical machine design. The D coils are electrically connected in series.

By a proper arrangement of the direction of the current in the C coils, the total forces on the D coils add arithmetically. Consequently, if there are "q" D coils, then the total force is qF where F is the force between one pair of C and D coils. The sequence of the excitation currents is indicated in Fig. 4, where the magnetic field lines produced by the C coils (rather than current sense) have been shown.

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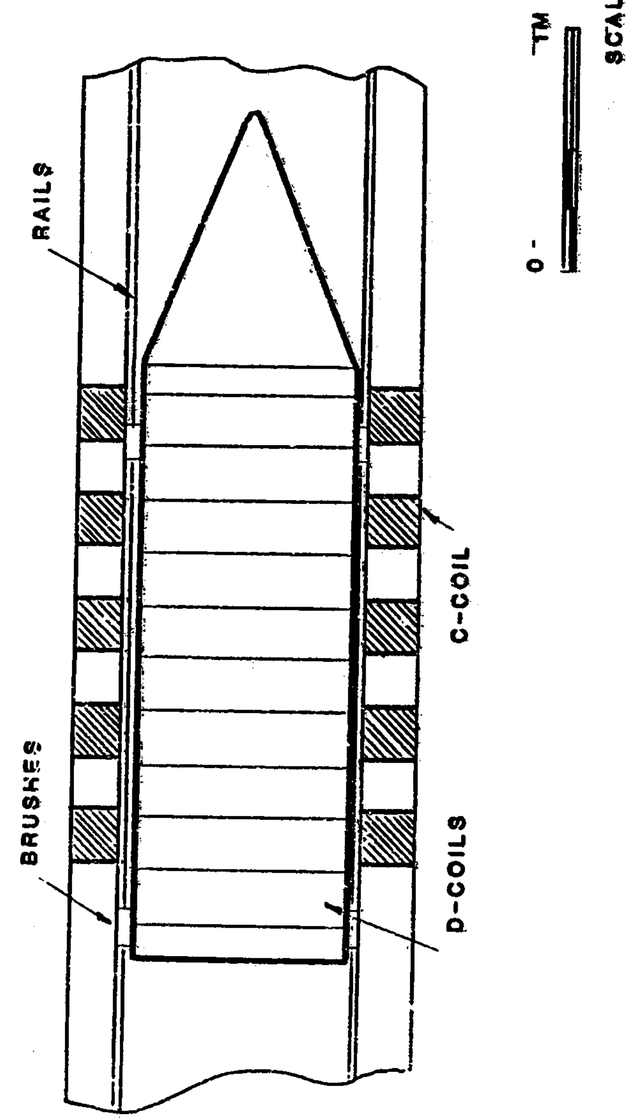

DRIVER COIL SYSTEM
collaborative planners, inc.
FIG. 3

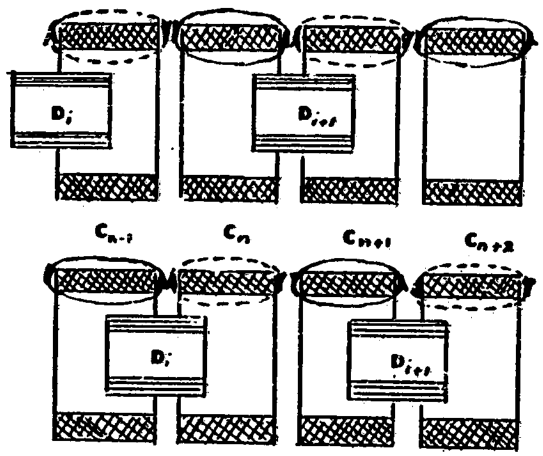

Fig. 4 Schematic Representation for the Distribution of Current in the Driving Coils. A Set of "q" Driver or C-coils are Excited with the Same Current in Series. The Coils are Connected Such That the Current in the set $C_{n-1}, C_{n+1}, \ldots\ldots C_{n+q-1}$ Flows in the Opposite Direction to the Current in the set $C_n, C_{n+2}, \ldots C_{n+q}$.

An advantage in connecting the driver coils in the manner described in Fig. 4, is that the number of switches needed to make the connections is reduced by q. The logic for the circuit used is indicated in Fig. 5.

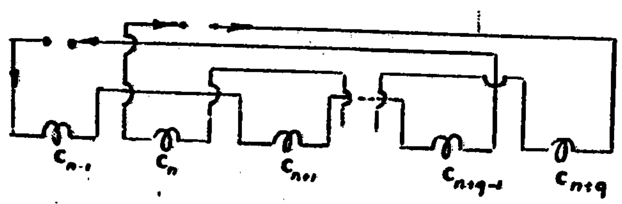

Fig. 5 Connection of the C-Coils Used in the Launcher.

The transit time of the projectile in the launcher is estimated to be of the order of a second. The energy required to propel the projectile is estimated to be in the vicinity of a hundred gigajoule. The corresponding power is enormous. It is most practical, therefore, to use a storage scheme in which energy can be stored at a reduced rate (since the launch frequency will never exceed 8/day) and which has a fast energy transfer time constant.

There are two possible options that appear most promising. The first one consists of an inductive storage system coupled to a fast Graetz[2] inverter. The arrangement has been studied at great length by Peterson and Boom[3] in connection with inductive storage for power applications. Actually, the Graetz circuit is used as a high voltage converter-inverter able to handle fairly large currents (up to 10,000 amps) at high voltage. The inverter uses SCR's which have proved to be rugged components with a high life and high reliability.

As the projectile accelerates to high speeds the transit time of a D-coil in a C-coil becomes shorter and shorter reaching a minimum of a tenth of a millisecond. Now, the inductive voltage drop across the C-coil increases inversely with the transit time. As a result, a very high voltage develops across the C-coils. The Graetz inverter is, therefore, very well suited to the electromagnetic launcher. An additional feature in favor of the Graetz inverter-converter (I/C) is its ability of producing an alternating current source of reasonably good wave form. As pointed out before, the voltage drop increases linearly with the frequency. This means that the presence of harmonics could lead to undesirable spikes of high voltage.

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The Graetz I/C is fundamentally a grid controlled reversible AC/DC bridge converter. Its frequency, therefore, can be controlled to fit the speed of the projectile along its travel. The overall system associated with the first option is shown in schematic form in Fig. 6.

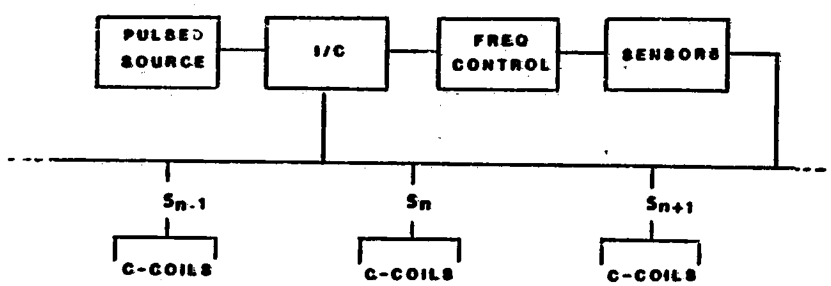

Fig. 6 Schematic Arrangement for the Power Connection to the Driver Coils.

Groups of "2q" C-coils are serviced by a same pulsed source. The number of these coils is to be evaluated on the basis of economics. The output of the pulsed source is modulated by a Graetz I/C whose frequency is made to follow the velocity of the projectile down the chain of D-coils. Sensors distributed along the chain relay the information to the frequency controller. The current to the D-coils is gated by the switches S_{n-1}, S_n,...that are also controlled by the same sensors. As will be shown later, one pulsed source may service 50-60 C-coils.

A second option consists in substituting the inductor of the pulsed source with a pulsed air core homopolar machine[4]. Early versions of this machine have yielded very encouraging results in terms of time constant, as well as, current intensities. A serious limitation to this machine is associated with the diffusion time of the magnetic field[5] out of the massive rotor. A solution that is bound to improve the performance of the machine by at least one order of magnitude consists in constructing the rotor out of laminated high conductivity copper sheets (or wedges). The state of these machines is such that a substantial amount of development is needed in order to obtain a design for the rotor that will yield a low diffusion time coupled with low electrical resistance and high mechanical strength to withstand the centrifugal stresses corresponding to high rotational speeds (30,000 rpm or better).

IV GOVERNING EQUATIONS FOR LAUNCHER PERFORMANCE

The starting point of the derivation is the relation for the conservation of momentum under the assumption that "q" D-coils attached to the projectile interact with q C-coils. This relation is

$$m \frac{dv}{dt} = q\, I_1 I_2 \frac{\partial M}{\partial s} \qquad (2)$$

where in the notation of Eq.(1), I_2 is the current in the C-coils, I_1 that in the D-coils, M is the mutual inductance between a C and D coil, m is the mass of the projectile and v its velocity.

The above equation can be rewritten as

$$m \frac{dv}{ds} \frac{ds}{dt} = m \frac{d}{ds} v^2/_2 = q\, I_1 I_2 \frac{dM}{ds} \qquad (3)$$

Hence the equation can be integrated. If the limits of integration are a displacement of half the length of a C-coil, i.e. from s = 0 to s = m_2 (See Fig. 1), then

$$\frac{m}{2}(v_a^2 - v_b^2) = q\, I_1 I_2\, (M_a - M_b) \tag{4}$$

When the limits of integration are selected to correspond from $s = m_2$ to $s = 2m_2$ one finds

$$\frac{m}{2}(v_b^2 - v_{a,1}^2) = q\, I_1 I_2\, (M_a - M_b) \tag{5}$$

Notice that the RHS has remained unchanged because the current I_2 has changed signs. Combining (4) and (5)

$$\frac{m}{2}(v_a^2 - v_{a,1}^2) = 2q\, I_1 I_2\, (M_a - M_b)$$

Now the value of $M_b << M_a$, so one can write approximately

$$\frac{m}{2}(v_a^2 - v_{a,1}^2) = 2q\, I_1 I_2\, M \tag{6}$$

Here M is the maximum value of the mutual, which is that for the coils being completely nested within each other. Had one repeated the integration for the 2^{nd}, 3^{rd},....n^{th} coil we would find

$$\frac{m}{2}(v_f^2 - v_a^2) = 2q\, M I_1 \sum_1^n (I_{2,1} + I_{2,2} + \ldots .. I_{2,n}) \tag{7}$$

where v_f^2 is the exit velocity at the n^{th} coil, v_a^2 is the initial velocity and $I_{2,1}$, $I_{2,2}$,...are the currents in the first, second,.....n^{th} coil.

Now, if the pulsed source excites the C-coils as a constant voltage source at a voltage V, then from circuit considerations

$$V = L_2 \frac{dI_{2,n}}{dt} + I_{2,n}R + v\, I_1 \frac{\partial M}{\partial s} \tag{8}$$

In the preceding equation L_2 is the self inductance of the D-coil, R its resistance, $I_{2,n}$ the current in the n^{th} coil and the last term is the back e.m.f. due to the motion.

Once one fixes the dimensions of the coils, then the system of equations (2), (7) and (8) completely defines the launcher. This system is difficult to solve in closed form and the solution is best performed numerically.

An approximate solution, however, can be obtained which will be most helpful in providing a sense for the performance and overall values of the design parameters. To that effect one writes for (8)

$$V = \omega L_2 I_{2,n} \tag{9}$$

where we have used for ω an effective frequency defined by

$$\omega = \frac{2\pi v}{2m_2} \tag{10}$$

The justification of the approximation of (9) is that the D-coils will be cryogenically cooled so that the inductive term will dominate. This approximation also permits to estimate approximately the distribution of the velocity along the launcher. Indeed, combining Eqs. (2), (9) and (10) one can show that

$$v = s^{1/3}\left(\frac{3m_2 q\, I_1 V \frac{\partial M}{\partial s}}{2\pi m L_2}\right)^{1/3} \tag{11}$$

This allows one to express Eq.(7) by

$$\frac{m}{2}(v_f^2 - v_a^2) = m^{1/3}\left(\frac{qI_1 V m_2}{\pi L_2}\right)^{2/3} \frac{2M}{(3m_2 \partial M/_{\partial s})^{1/3}} \sum_1^n \left(1 + \frac{1}{2^{1/3}} + \frac{1}{3^{1/3}} + \ldots \frac{1}{n^{1/3}}\right) \tag{12}$$

But it can be shown that the sum can be evaluated by means of a well known[6] procedure in which the finite series is approximated by an integral. Hence one finds

$$m^{2/3}(v_f^2 - v_a^2) = \left(\frac{qI_1 V m_2}{\pi L_2}\right)^{2/3} \frac{4M}{(3m_2 \partial M/_{\partial s})^{1/3}} \left(1 + \frac{3}{4} n\right)^{2/3} \tag{13}$$

The last expression (13) together with (2) and (8) formally provide all the relations to be used in the design of the launcher. Thus (2) will allow one to select the value of $\frac{\partial M}{\partial s}$ (shown in Fig. 2) to satisfy the acceleration requirements, (8) defines the source voltage and (13) yields the number of coils needed to achieve the escape velocity v_f.

V DESIGN PARAMETERS

The procedure developed in the previous section will be used to obtain the design parameters for the two reference designs indicated in the introduction.

To minimize the mass of the driven coils, these coils are given a Brooks' configuration[7]. The proportions for the D-coils become as shown in Fig. 7.

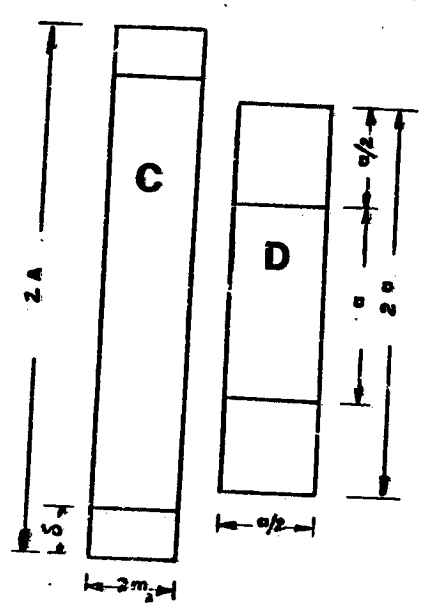

Fig. 7 D-Coil Configuration with Proportions Satisfying Brooks' Conditions. A C-Coil is Shown Also for Reference.

The D-coils, obviously, are not refrigerated. The maximum allowable current density in the wire is therefore, that for pulsed coils. If we denote this density by $j_{c,1}$, then

$$\left(\frac{a}{2}\right)^2 = \frac{N_1 I_1}{j_{c,1}} \phi \qquad (14)$$

where ϕ is a packing factor. Now the radius a is given. In fact, it is approximately the radius of the pole-like projectile viz. a = 0,45 m. Since the current I_1 is selected to meet the conditions of an acceptable current density in the brushes Eq.(14) defines the allowable number of turns N_1. We thus find

$$j_{c,1} = 4000 \text{ amps/cm}^2$$
$$\phi = 0.95$$
$$N_1 I_1 = 2.13\text{x}10^6$$
$$I_1 = 1000 \text{ amps}$$
$$N_1 = 2130 \text{ turns}$$

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In order for a D-coil to be able to slide freely inside a C-coil, it is clear that by inspection of Fig. 7 one obtains

$$A - \delta \geq a \qquad (15)$$

Now, by manipulation of Eq.(15),

$$\frac{a}{m_1}\,\frac{m_1}{m_2} \leq \frac{A}{m_2} - \frac{\delta}{m_2} \qquad (16)$$

which can be rewritten in the notation of Section II as

$$\frac{a}{m_1}\,\xi \leq \beta - \frac{\delta}{m_2} \qquad (17)$$

From Brooks' conditions $\frac{a}{m_1} = 4$. Hence (17) becomes

$$4\,\xi \leq \beta\,\frac{\delta}{m_2} \tag{18}$$

Another subsidiary equation for the maximum current in a C-coil yields an additional relation between δ and m_2. In fact

$$2m_2\delta \leq \frac{N_2 I_2}{j_{c,2}} \tag{19}$$

If we select the cross-section of the C-coil to be equal to that of a D-coil then to meet the requirement of the necessary force given by Eq(2) the current density $j_{c,2}$ will have to be much greater than $j_{c,1}$. This dictates the need for cryogenically cooling the C-coils. Should one decide not to do that, a much larger cross-section (at an added cost) will be needed.

Therefore from

$$\frac{a^2}{8m_2\delta} = 1 \tag{20}$$

it follows that

$$\frac{a^2}{8m_1^{\,2}}\,\frac{m_1^{\,2}}{m_2^{\,2}} = \frac{\delta}{m_2}$$

i.e.

$$\frac{\delta}{m_2} = 2\,\xi^2 \tag{21}$$

By referring to Fig. 2, there is an advantage in selecting $\xi = 1$ since the forces maximize for that value of ξ.

It is thus found from (21) and (18)

$$m_2 = m_1 = 11.25\ \text{cm} \tag{22}$$

$$\delta = 2m_2 = 22.50\ \text{cm} \tag{23}$$

$$\beta = \frac{A}{m_2} = 6 \tag{24}$$

$$A = 0.675\ \text{m} \tag{25}$$

Since ξ, β are found it is readily obtained (Fig. 2) that the "f" function is equal to 0.65 while $\frac{\partial f}{\partial \xi} = 0.09$. This means that

$$M = 1.44\ N_1 N_2 \times 10^{-6}\ \text{H} \tag{26}$$

$$\frac{\partial M}{\partial z} = 0.80\ N_1 N_2 \times 10^{-6}\ \text{H/m} \tag{27}$$

The value of the current I_2, of course, varies along the chain of the coils being in fact approximately given by

$$I_2 = \frac{V}{\omega L_2} = \frac{Vm_2}{\pi v L_2}$$
$$= (2n)^{-1/3}\,\frac{Vm_2^{\;2/3}}{\pi L_2} \tag{28}$$

where n is the numerical location of the n^{th} C-coil.

The mass "m" of the projectile actually consists of the mass "m" of the payload augmented by that of the driven coils. If we denote the mass of one D-coil by m_c, then

$$m = m_o + qm_c \tag{29}$$

But

$$m_c = \rho\,\left(N_1\,2\pi\,a\,\frac{I_1}{j_{c,1}}\right) \tag{30}$$

where ρ is the density of the wire material.

The driven coils can be constructed out of copper or of aluminum. The lighter aluminum, however, has a low electrical conductivity, as well as, low heat capacity per unit mass. As a result, the energy saving in bringing the projectile up to speed may be compensated by the added heat losses. To fix the ideas, we estimate these various quantities.

The resistance of the coil is given by

$$R_1 = (\rho_e N_1 2\pi a j_{c,1}/I_1) \qquad (31)$$

while the temperature rise ΔT is found from

$$(I_1{}^2 R_1)t_1 = m_c C_p \Delta T \times 4.2 \qquad (32)$$

In the above expression t_1 is the transient time of the D-coil in the launcher, C_p is the specific heat, and ρ_e is the electrical resistivity. One thus finds for a transit time of the D-coil of the order of a second the following values for the case of $q = 10$ and $m_o = 6500$ kg.

	Cu	Al
ρ gm/cm^3	8.89	2.699
ρ_e ohm cm	1.69×10^{-6}	2.83×10^{-6}
C_p Cal/gm/^{o}C	0.092	0.214
m_c kg	1338	406
R_1 ohm	3.88	6.82
Δ T^{o}C	7.5^{o}C	16.70^{o}C
$m = m_o + q\, m_c$	19,880	10,560
$(I^2 R_1 t_1)/mv^2$	1.91×10^{-4}	$.65\times10^{-4}$

There appears to be a definite advantage (energy wise) in using aluminum for the D-coil material.

We are now in a position to complete the design of the launchers for the two options mentioned in the introduction.

<u>Earth-Orbital: Option I</u>

If the total mass of the projectile is not to exceed 6500 kg, then the pay load is reduced and is found from

$$6500 = m_o + q(406) = m \qquad (33)$$

The initial velocity v_a which appears in Eq.(13) is actually zero. If we select the follwing values for the launcher

$$N_2 = 50$$
$$N_1 = 2130$$
$$M = 0.153 \text{ H}$$
$$\frac{\partial M}{\partial a} = 0.085 \text{ H/m}$$
$$L_2 = 0.058 \text{ H}$$
$$V = 10^5 \text{ volts}$$
$$I_1 = 10^3 \text{ amps} \qquad (34)$$

We therefore obtain for the number of coils needed the expression

$$n = (.54\times10^{-9})v_f^3 \frac{m}{q} \qquad (35)$$

Here v_f stands for the final velocity. The length of the launcher is given by

$$\ell = n(2m_2) = 0.225\, n \quad \text{meters} \qquad (36)$$

The ohmic losses in the launchers consist of the energy dissipated in the C-coils plus that in the D-coils.

It is readily shown that this energy E_L is given by

$$E_L = (\ell n\, n\, (\tfrac{m}{q})^{2/9}\, 1.05\times10^4 + q\, 6.82\times10^6) \text{ joules} \qquad (37)$$

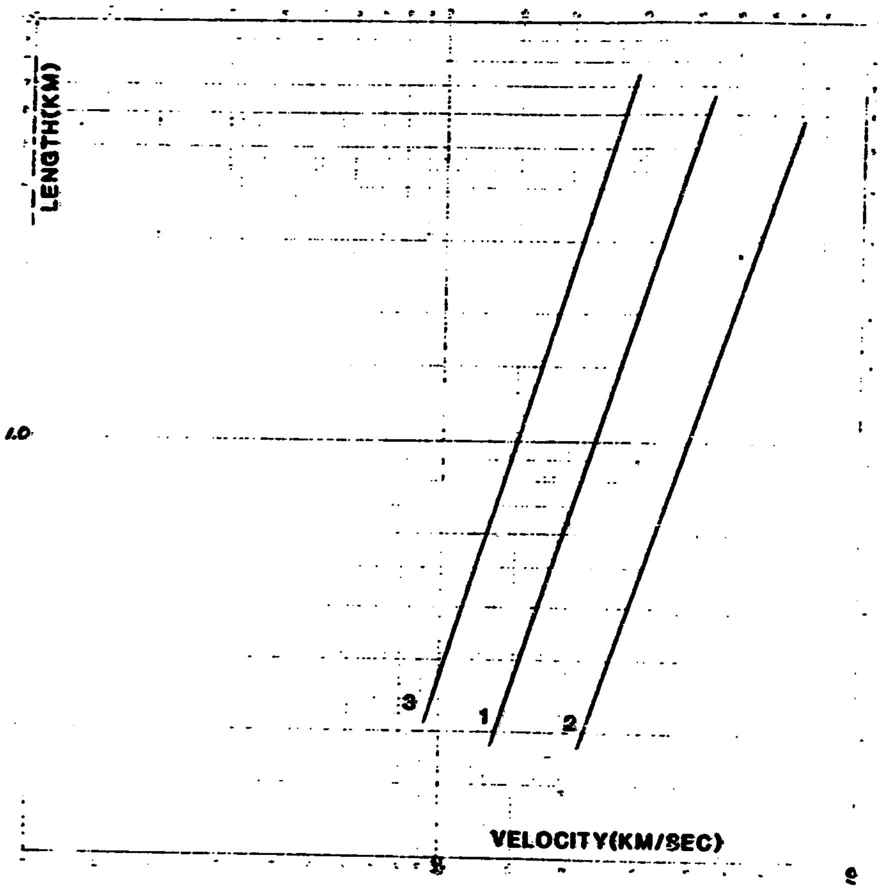

Fig. 8 Dependence of the Launcher Length on the Exit Velocity.
Curve 1 is for a Mass of 6500 kg and q = 10.
Curve 2 is for a Mass of 1500 kg and q = 10.
Curve 3 is for 650 kg, q = 1 and with 2000 amps in in the D-Coil and a Source Voltage of 200,000 volts.

The input energy to the system is

$$E_1 = E_L + \frac{m}{2} v_f^2 \qquad (38)$$

The functional dependence of n (or the length) on the exit velocity is shown in graphical form in Fig. 8. Because of the very rapid rise of the length with the exit velocity it does not appear that within the assumptions set forth in (34) high velocities can be reached within reasonable lengths of the launcher. A hybrid system is indicated in which a chemical rocket provides the last state of acceleration to reach the 5-10 km/sec goal.

Some typical values for the conditions outlined in (35) are given below.

m(kG)	q	v_f(km/sec)	ℓ(km)	E_f(Joules)
650	1	4.8 km	2	$1.3x10^9$
1000	2	3.3 km	2	$2x10^9$
6500	10	2.9 km	2	$1.3x10^{10}$

For the lighter masses (650 kg) a velocity of 5 km/sec can be achieved provided the current I_1 in the driven coil and/or the voltage V is increased. Inspection of (13) shows that these two quantities appear as a product and n is inversely proportional.

One possible way of increasing I_1 and of avoiding the problem of the brushes is to use a procedure similar to that followed by Yoshikawa in his Tokomak work[8]. The method is to cool down the D-coils to superconducting temperatures, induce a current in them and then launch the projectile. In this manner, velocities of 5 km/sec can be reached within 2 km for a projectile of m = 650 kg.

E-11

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OPTION II

The requirement that the acceleration should not exceed 100 g. indicates (from (2) and (13) that the product VI_1 should be less than $2.42x10^9$. Hence, V_1, I_1 remain unchanged. Substituting the mass m= 15,000 v = 2 km/sec in Eq. 35 we find n = 6,480. This corresponds to a length of 1.46 km which is acceptable.

A hybrid system is therefore essential. Using a launching length of 2 km, we find that n = 8888 and the exit velocity by means of (35) works out to be 2.22 km/sec.

CONCLUSIONS

OPTION I

Using the parameters of (34) a hybrid launcher (EML cascaded to a chemical rocket) is an easy solution. However, by resorting to a sophisticated cooling of the D-coils one single system can be constructed. The system can be used up to 4.8 km/sec. For higher speeds a hybrid system is mandatory. Fig. 8 summarizes the results in graphical form.

OPTION II

For the heavy mass and low acceleration one single system may be used.