FINAL TECHNICAL REPORT

ON

PRELIMINARY FEASIBILITY ASSESSMENT FOR EARTH-TO-SPACE ELECTROMAGNETIC (RAILGUN) LAUNCHERS

TO

NATIONAL AERONAUTICS AND
SPACE ADMINISTRATION
LEWIS RESEARCH CENTER

(Contract Number NAS3-22882)

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16. Abstract

The Preliminary Feasibility Assessment of Earth-to-Space Electromagnetic (Railgun) Launchers (ESRL) for launching material into space was a study to determine the viability of developing an electromagnetic rail launcher system to launch material into space. Potential ESRL applications were identified and initially assessed to formulate preliminary system requirements. The potential applications included nuclear waste disposal in space, Earth-orbital applications, deep space probe launches, atmospheric research, and boost of chemical rockets. Based upon the requirements and input from subcontracted railgun and projectile experts, prime ESRL concept options were selected and a Reference Concept was defined. The Reference Concept was developed and the requirements were revised before evaluation of the ESRL system concept. The ESRL system concept consisted of two separate railgun launcher tubes (one at 20° from the horizontal for Earth-orbital missions, the other vertical for solar system escape disposal missions) powered by a common power plant. Each 2040-m launcher tube would be surrounded by 10,200 homopolar generator/inductor units to transmit the power to the rails. Projectile masses envisioned would be 6500 kg for Earth-orbital missions and 2055 kg for nuclear waste disposal missions. For the Earth-orbital missions, the projectile requires a propulsion system, leaving an estimated payload mass of 650 kg. For the nuclear waste disposal in space mission, the high-level waste mass was estimated at 250 kg. This preliminary assessment included technical, environmental, and economic analyses. These analyses indicated that the ESRL system appeared to be feasible and potentially economically beneficial. More work would be needed to develop the concept, including experimental and systems studies. Some environmental effects would be expected for development and construction, but none should prevent a go ahead. Space transportation costs perhaps could be reduced by an order of magnitude in comparison with conventional systems. Based upon these analyses, it was concluded that an ESRL system appears to be technically feasible in the 2020 time frame. A supporting research and technology evaluation indicated areas of necessary development. It was recommended to proceed with further study and development of ESRL system technology at a moderately low level of funding.

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FOREWORD

This Battelle Columbus Laboratory study was sponsored by NASA's Lewis Research Center under NASA Contract Number NAS3-22882, for the purpose of providing a preliminary feasibility assessment of Earth-to-space electromagnetic (railgun) launchers. Work was conducted from May, 1981 through June, 1982. Battelle's assessment involved: (1) the development of a Reference Concept for an Earth-to-Space Rail Launcher (ESRL) system; (2) a preliminary economic assessment; (3) a preliminary environmental assessment; and (4) an initial assessment of technology needs. Emphasis was placed on system concept development and the economic assessment. To support the system concept development, subcontracts were given to three of the nation's leading railgun experts: Dr. Richard Marshall, University of Texas at Austin; Dr. John Barber, IAP Research, Inc., Dayton Ohio; and Mr. Ron Hawke, Lawrence Livermore National Laboratory (LLNL). In addition to these people, NASA's Lewis Research Center supported: Dr. John Lee, Aeronautical and Astronautical Research Laboratory, The Ohio State University, Columbus, Ohio; Dr. Al Buckingham, LLNL; and Mr. Hal Swift, PAI Corporation, Dayton, Ohio; for the purpose of obtaining information in technical areas relating to projectile design, sabots, and aerodynamics.

Information developed during the study period is contained in this final report. Inquiries regarding this study should be addressed to:

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Dr. Richard G. Jung	Mr. Albert E. Weller

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Special appreciation is given to Mr. William R. Kerslake, the NASA/LeRC study monitor and Mr. Fred F. Terdan, NASA/LeRC, manager of the Propulsion Systems Technology Section, for their interest and guidance throughout the study effort.

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1.0 EXECUTIVE SUMMARY

The technical findings of this "Preliminary Feasibility Assessment for Earth-to-Space Electromagnetic (Railgun) Launchers are documented in this final report. The study background, objectives, approach, principal assumptions, summary of major results, conclusion, and recommendation are presented in this section (Section 1.0--Executive Summary). Technical details are given in Sections 2.0 through 8.0. Appendix A provides references; Appendix B provides definition of acronyms and abbreviations; and Appendix C provides metric to English unit conversion factors. Appendices D through H provide an overview of some of the material presented by subcontractors at the 12-13 August 1981 Concept Defintion Meeting, held at Battelle. Appendix I provides the distribution list for this report.

1.1 Study Background

NASA's Lewis Research Center, Cleveland, Ohio, has an ongoing interest in all forms of advanced space transportation and propulsion systems. Because of this interest, current activity by other government organizations, and the promise of railgun technology, this study was conducted to assess the feasibility of an Earth-to-Space Rail Launcher system employing the railgun concept. The paragraphs below provide background on what a railgun is.

Electromagnetic rail launcher concepts (railguns) have existed since the early 1900's. The recent work of Rashleigh and Marshall (1978)*, and Barber (1972), gave credence to the potential of proposed railgun accelerators. Electromagnetic rail launchers offer a potential means of propelling massive projectiles at hypervelocities and could result in other advantages over conventional launchers. A major application is seen in the general field of impact physics, including the specific possibility of nuclear fusion by impact. Other potential applications of this technology include ballistic weapons, space propulsion, and Earth-to-space launchers.

Electromagnetic rail launcher (railgun) research was revived at the Australian National University about ten years ago, using a single large inductor as the power source. More recently, a Lawrence Livermore National Laboratory/Los Alamos National Laboratory team has successfully demonstrated the use of explosively driven magnetic flux compression generators to power a variety of rail launchers.

An electromagnetic rail launcher consists of two conducting rails (electrodes) between which a conducting element with an attached projectile is placed. Electric current is passed along one electrode, through the conducting portion (armature) of the conductor/projectile, and back along the other electrode (see Figure 1-1). The current, I, flowing through the projectile armature interacts with the magnetic flux generated by the current loop resulting in an I x B force in the direction indicated. As the projectile is free to slide along the rails, it will be accelerated by the I x B force as

*References are given in Appendix A.

long as current continues to flow in the circuit and the conductor remains in electrical contact with the rails.

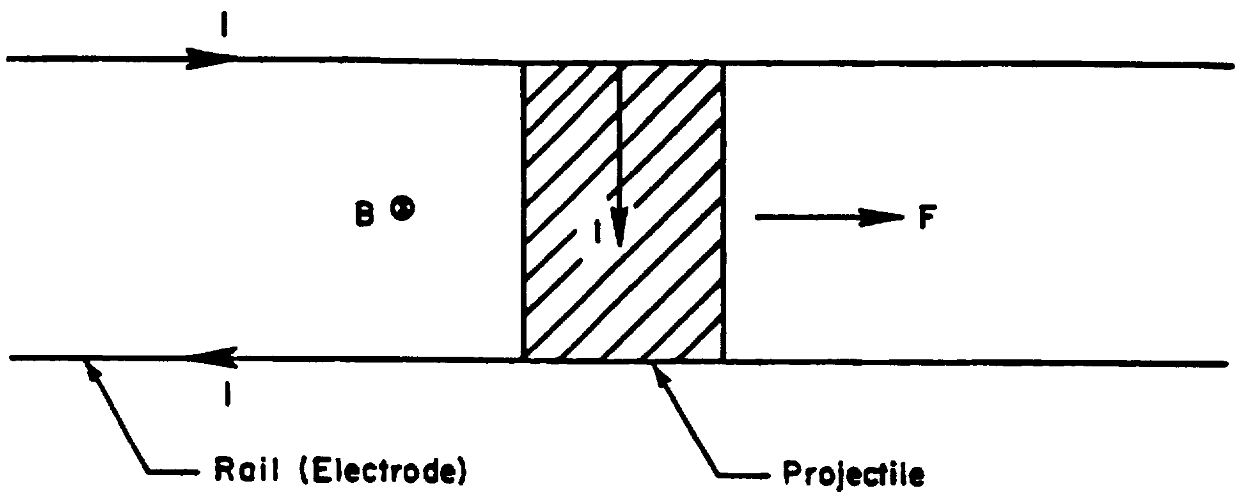

FIGURE 1-1. BASIC PRINCIPAL OF A RAILGUN

Rail-launcher-produced velocities of the order of 10,000 m/s have been achieved for small mass samples. Initial velocities greater than 20,000 m/s for objects having proper characteristics are believed necessary for an Earth-surfaced rail-launched payload to escape the solar system. Currently, NASA/LeRC is sponsoring work at LLNL with the goal of achieving 15,000 m/s.

1.2 Study Objectives

The overall objectives of this study were: (1) to provide NASA/LeRC with sufficient preliminary information in various areas, such that the potential feasibility and benefits of (Earth-to-space electromagnetic railgun launchers) could be determined; (2) to define a reference system concept; (3) to conduct preliminary assessments of system safety, economics, and environmental impact; and (4) to recommend areas of technology development.

1.3 Approach

The study approach emphasized the assessment of important factors which would determine the potential feasibility and benefits of an Earth-to-Space Rail Launcher (ESRL). Important factors included: system requirements and definition, safety, environmental impact, costs, and technology status.

To assure proper development of a reference concept for an ESRL concept, appropriate interaction among the study participants was necessary. A concept definition working group was formed. It was comprised of the study manager, appropriate Battelle staff members, outside consultants (railgun experts), interested parties, and NASA personnel. The attendees of the ESRL Concept Definition Meeting were as follows:

Dr. John P. Barber, IAP Research
Mr. Ralph E. Best, DOE/ONI
Dr. Joe H. Brown, Jr., BCL
Dr. Alfred C. Buckingham, LLNL
Mr. Richard W. Earhart, BCL
Dr. Donald S. Edgecombe, BCL
Dr. Harold M. Epstein, BCL
Mr. Ellis L. Foster, BCL
Mr. William E. Galloway, NASA/MSFC
Mr. William A. Glaeser, BCL
Mr. Mike N. Golovin, BCL
Mr. R. S. Hawke, LLNL
Mr. Raymond E. Hess, BCL
Mr. Terry E. Hill, BCL
Dr. L. E. Hulbert, BCL
Mr. W. R. Kerslake, NASA/LeRC
Dr. John D. Lee, OSU
Dr. Richard Marshall, UT
Ms. Lisa A. Miller, BCL
Dr. Dennis Peterson, LANL
Mr. Richard F. Porter, BCL
Dr. P. K. Ray, Tuskegee Institute
Mr. Warren D. Rayle, UP
Dr. Eric E. Rice, BCL
Mr. Hal F. Swift, PAI
Mr. Fred F. Terdan, NASA/LeRC
Mr. Guy C. Throner, BCL
Dr. Victor P. Warkulwiz, Analytical Serv.
Mr. Bert E. Weller, BCL

This group met once at Battelle (on 12-13 August, 1981) and agreed on the definition of a basic ESRL Reference Concept. From that, Battelle developed the concept further.

Figure 1-2 outlines the specific study activities, the interrelationships between each, and the overall flow of data in the study. The various assessments conducted by Battelle were based upon the ESRL Reference Concept that was ultimately defined.

1.4 Principal Assumptions

The principal assumptions that were used in the performance of this study included:

- Battelle and its subcontractors to make maximum use of related studies and other associated data, as appropriate.
- Nuclear waste material would be the prime candidate for "disposal" launches, but other applications to be identified and briefly assessed.
- For a nuclear waste disposal mission model, only the waste from U.S. commercial reactors to be considered.
- Only consider peaceful uses/applications of an ESRL concept.
- Only railgun technology considered for the launcher.
- All costs to be in 1981 $.
- Study activity scoped to follow allocated funding resource.

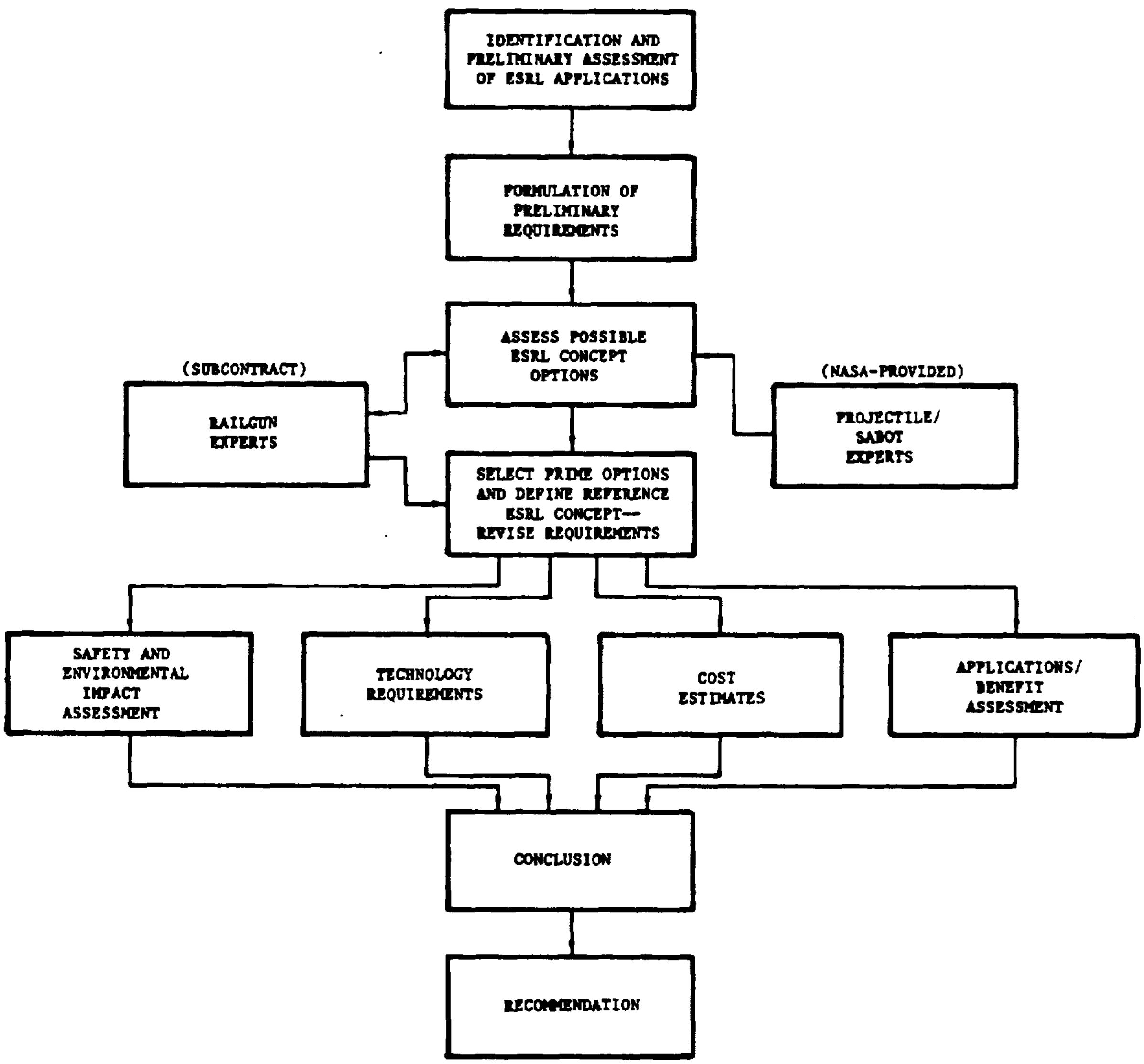

FIGURE 1-2. OVERVIEW OF STUDY FLOW AND STUDY AREAS

1.5 Basic Requirements

The Earth-to-Space Rail Launcher (ESRL) system is envisioned to be a multi-purpose space launcher with a primary application of high-level nuclear waste disposal in space (NWDS). Secondary applications would include the launching of planetary probes, low Earth orbit satellites, and basic materials for use in space. Additional applications could involve the conduct of high altitude research projects (suborbital launches).

Early in the study, system requirements were generated to guide the development of the ESRL Reference Concept; later they were updated. As a result, the updated general performance requirements for the Earth-to-Space Rail Launcher (ESRL) system are as follows:

Launch Site:	Remote island near equator
Maximum Acceleration:	10,000 g's for nuclear waste disposal in space (NWDS) missions 2,500 g's for Earth orbital missions
Launch Velocities:	20 km/s for NWDS missions 5-10 km/sec for Earth orbital missions
Launch Azimuth:	90° E
Launch Elevation:	90° for NWDS missions (vertical) 20° for Earth orbital missions
Payloads:	0.5 MT/day HLW for NWDS missions
Launch Frequency:	As few as possible to accommodate given launch mass requirements
Mission Reliability:	As high as possible (>0.999)
Safety:	System to include safety systems and recovery contingency for hazardous payloads
Reusability:	The system should exhibit high reusability to maintain low life-cycle costs
Launch Constraints:	Cloud cover, rain, wind direction, air traffic, and space traffic
Materials Use:	No significant impact on world-wide materials use.

Also, system safety design requirements were defined for the ESRL nuclear waste disposal in space mission. These requirements provided the guidelines against which a rail launched nuclear disposal in space may be considered acceptable from a radiological safety point of view. The safety objectives for the disposal mission are to: (1) contain the solid radioactive waste material; (2) limit the exposure of humans and the environment to radioactive waste materials; and (3) mitigate the potential non-radiological environmental effects of operations. For normal operations, complete containment and minimum radiological exposure are required; for potential accident situations, the degree of containment and degree of interaction should result in an acceptable risk to humans and the environment.

The safety requirements for the ESRL system, as applied to the ESRL nuclear waste disposal in space mission (described in detail in Section 2.3), involve consideration of the following safety aspects:

- Non-Radiological Safety
- Radiation Explosure
- Containment
- Accident Environments
- Criticality
- Postaccident Recovery
- Monitoring Systems
- Isolation.

1.6 Reference Concept Definition

This section briefly describes the Earth-to-Space Rail Launcher (ESRL) system that has evolved over the course of the study, and is the basis for this preliminary feasibility assessment. The Reference Concept is very preliminary and considerable additional analytical work is necessary to develop an optimum and detailed system description. However, it does represent a pooling of railgun expert opinion, engineering judgment, and properly defined mission requirements. The Reference ESRL Concept consists of two basic launchers: (1) a vertical launcher for NWDS missions; and (2) a launcher inclined 20° from the horizontal for Earth-orbital application missions.

Figure 1-3 provides an overview of the Earth-to-Space Rail Launcher nuclear waste disposal in space mission scenario. Fuel rods from commercial nuclear power plants would be taken to a nuclear waste processing and projectile/payload fabrication facility located in the United States. At this facility the nuclear waste would be processed into various components, some of which would go to a mined geologic repository, and others which could go to space disposal. Waste for space disposal would be aged in storage for a period of time then encapsulated into the nuclear waste payload to become part of the projectile that would be launched by the ESRL system.

The mission scenario for Earth orbital applications would be similar to the NWDS mission, but projectiles and their payloads would not require the stringent safety procedures. Propellants and fluids required for on-orbit propulsion would be loaded at the launch site.

Projectiles and payloads would be fabricated and then transported by a rail car to an ocean seaport where they would be loaded onto a cargo vessel. The ship would then deliver the projectiles/payloads to a remote island launch site. They would be off-loaded onto a rail car and transported to a storage and checkout facility. Here nuclear waste projectiles would be removed from the shipping cask and placed in individual auxillary shields and stored; Earth

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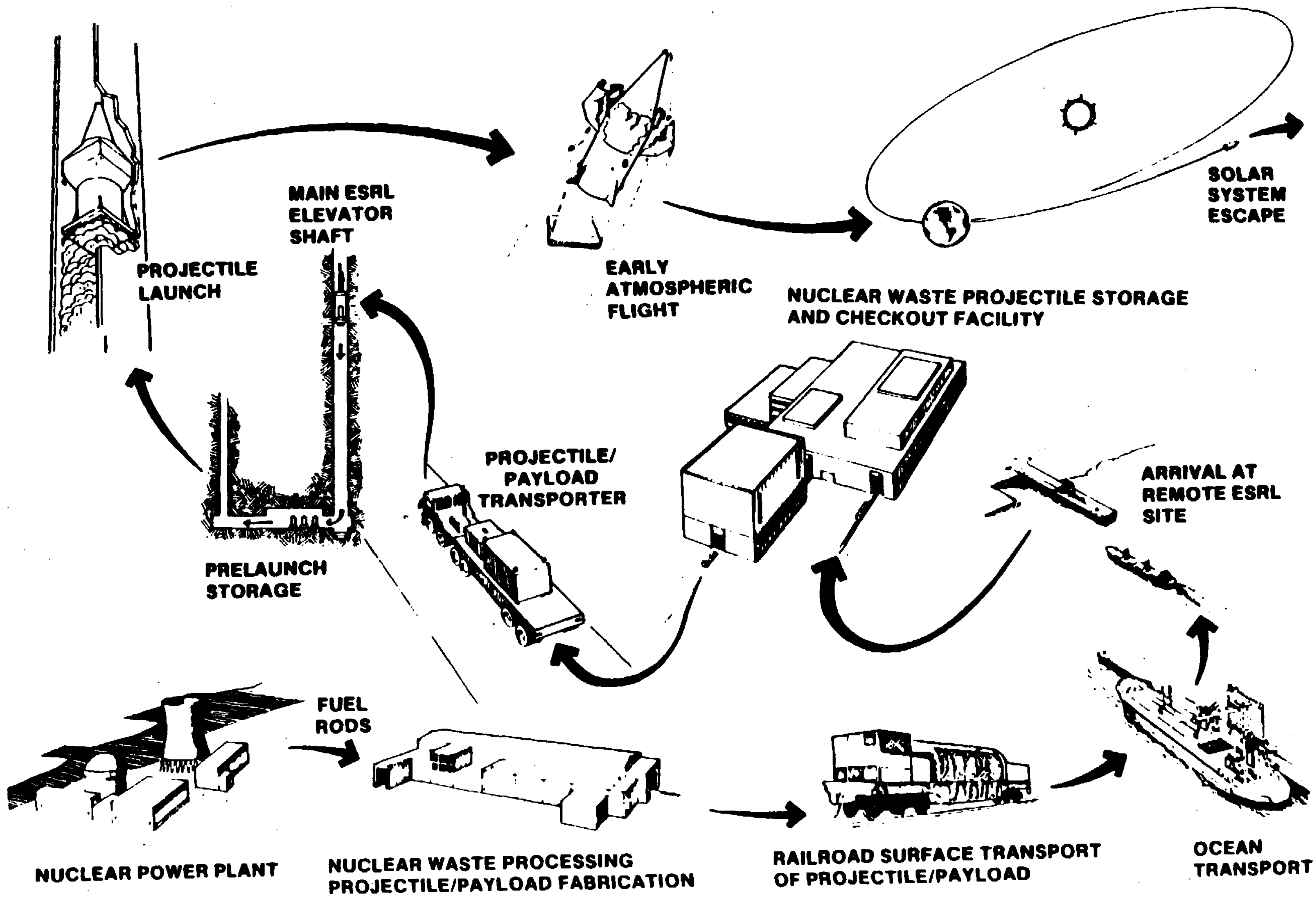

FIGURE 1-3. OVERVIEW OF ESRL NUCLEAR WASTE DISPOSAL IN SPACE MISSION SCENARIO

orbital projectiles would be stored and prepared for launch. At the proper time, the projectile, with its payload, would be placed on a flatbed truck and driven to the main ESRL elevator shaft where it would be lowered into the prelaunch storage area below the surface. For nuclear waste disposal missions, at the scheduled time for launch, the auxiliary shield would be removed and the projectile would be placed in the muzzle of the launcher tube. After the launcher system has been fully charged and prepared for launch, the projectile would be launched. For nuclear waste disposal, the muzzle velocity would be 20,000 m/s and the destination would be solar system escape. For low-Earth orbit missions, the muzzle velocity would be approximately 7,000 m/s, with a provision for 2100 m/s Δv at a 500 km orbit altitude.

Figure 1-4 shows an artist's concept of the remote island launch facility, along with the shipping/receiving, storage facilities, launcher systems, and other facilities shown. Indicated in the figure are the launcher muzzles, the underground launch control facility, radar tracking facility, the water/liquid nitrogen and hydrogen/oxygen storage area, the liquid, gas, and water plant production plant, the nuclear power plant, the industrial area, the airfield, the administration and engineering facilities, and the community living area. The community living area would be located at the greatest distance away from the launcher muzzles to reduce impact caused by sonic booms.

Figure 1-5 shows a cutaway of the island in the plane of the launcher tubes and the overall concept of the railgun launcher system, along with the Earth orbital and the nuclear waste projectile concepts. Detailed discussion of these systems is provided in Sections 3.1 and 4.0 of this report.

1.7 Summary of Major Results

The paragraphs below summarize the major results of the assessment by study area:

Mission Applications, Traffic and Requirements

Of missions identified, nuclear waste disposal, deep space probes, and Earth orbital missions appear attractive as system concept drivers. For the 2020-2050 time period, perhaps as much as 3.0 MT per day of bulk material could be launched to Earth orbit (8 launches/day) and approximately 0.5 MT per day of high-level nuclear waste could be launched to solar system escape. When considering Earth-orbital missions, the payload should be as large and dense as practical. For space station logistics missions, a zero inclination orbit is required for more than one launch per day from one launcher tube. As the target altitude increases, required muzzle velocity goes up and on-orbit Δv requirement goes down.

For nuclear waste disposal missions, the payload should be as large (in diameter) as practical for shielding efficiency. Reducing shielding mass by permitting the allowable radiation dose rate to increase from 10 to 100 rem/hr at 1 meter only doubles the possible waste payload. Launch window considerations (4-6 hr) require dawn launches at an equatorial launch site and

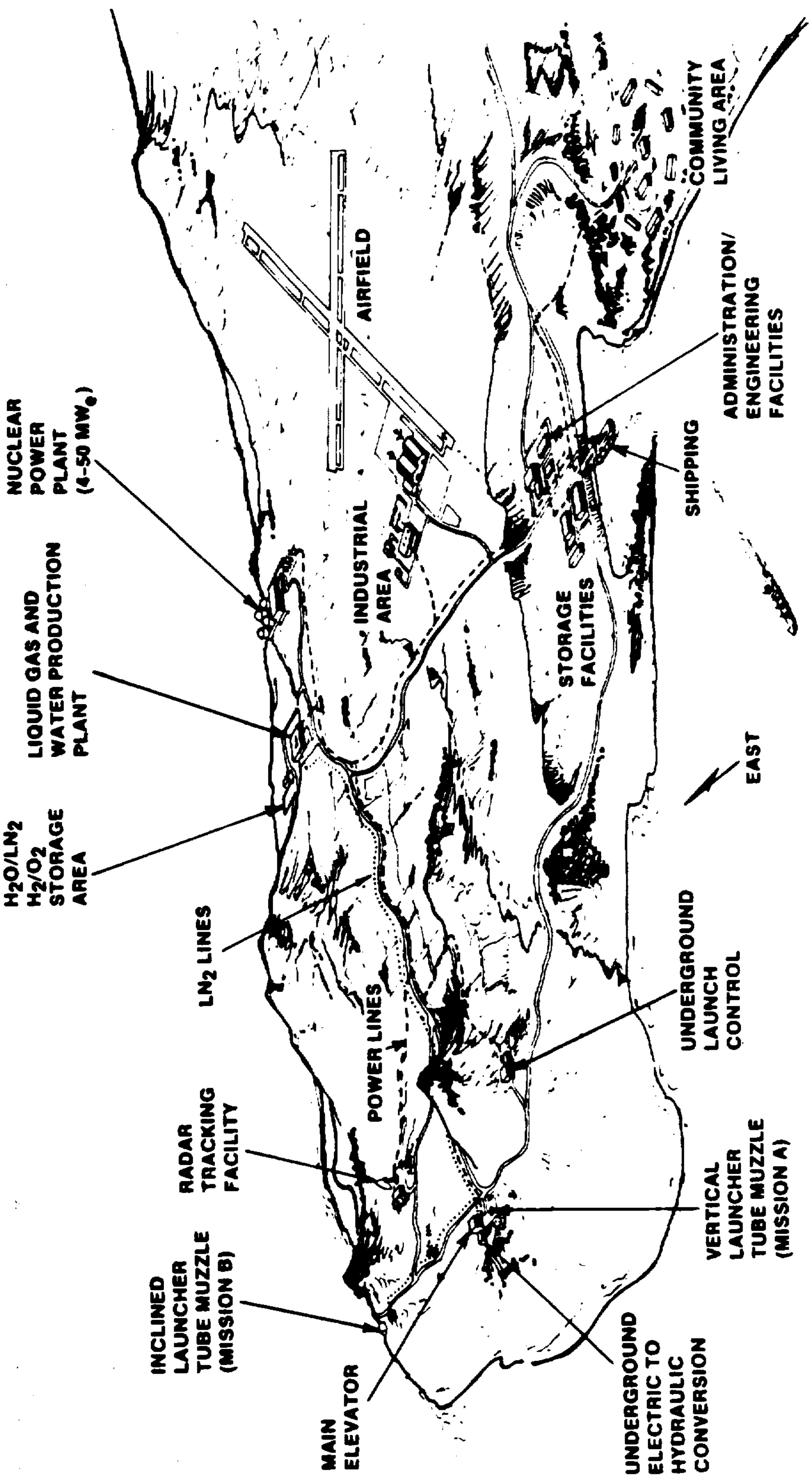

FIGURE 1-4. ARTIST'S CONCEPT OF THE ESRL REMOTE ISLAND LAUNCH SITE

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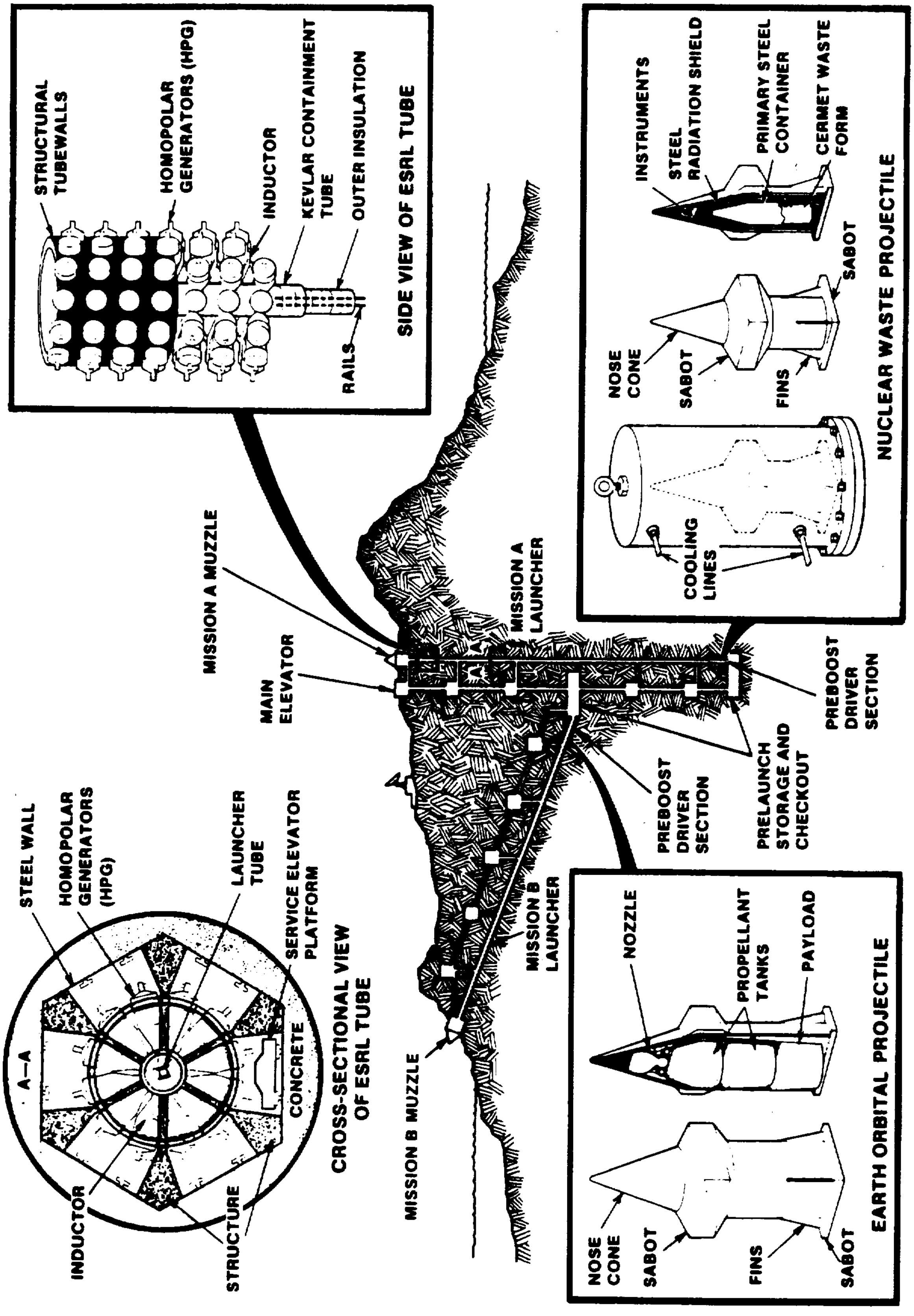

FIGURE 1-5. OVERVIEW OF ESRL LAUNCHER SYSTEM AND PROJECTILES

a launch velocity of 20 km/s to account for off-optimal launch and atmospheric drag. It may be possible to launch all commercial and defense nuclear waste to solar system escape and eliminate need for mined geologic repositories altogether. This aspect requies further study.

ESRL Projectiles

ESRL Projectiles in general require a jettisonable sabot, a low drag, high ballistic coefficient design (cone dart), fins for stabilization, and a metal nose tip.

Stagnation point ablation on the nose is expected to be on the order of 1 to 2 cm, depending upon material selection; more analysis is required. Tungsten is currently shown as the nose cone material; however, steel is currently recommended based upon cost, material availability, and the expected ablation rates.

Preliminary evaluations indicated that flight through clouds and/or rain is expected to have catastrophic consequences, therefore, weather constraints are likely to be imposed. For the nominal ESRL projectile concept, a drag coefficient of 0.1 and a high ballistic coefficient are needed to keep velocity losses below about 5 percent.

For Earth orbital missions, a meaningful projectile mass was approximately 6.5 MT, and an on-board propulsion system is required. Projectile propulsion systems require 3-axis control, a high density, and high specific impulse propellant. A hydrazine/chlorine trifluoride pressure-fed liquid system was selected for study purposes, however, others are possible, including solids. In any case, simple storage and simple ignition start is most desirable.

Launcher Systems

An all-azimuth, movable launcher tube was given consideration, but it appears not to be feasible; it is too large and long to move. Analysis indicates that the launcher system would be large, if meaningful sized payloads are to be successfully launched.

Three types of railgun launch systems were investigated: (1) the single energy store (SES); (2) the segmented distributed energy store (SDES); and (3) the integral distributed energy store (IDES). The IDES was selected for Reference Concept because of its potential performance, not its simplicity. Table 1 discusses the advantages and disadvantages of each. Switching for SDES and IDES systems is complex and additional work is necessary to evaluate performance. SES switching is simple, but the launcher tubes are longer. In DES systems, tailoring of current is possible to allow improved system life.

A preboost is required for extended rail life. A preboost velocity of approximately 1000 m/s has been determined by experiment.

Currently the Reference concept includes a "square bore", however, a round bore launcher appears somewhat attractive. Experiments conducted at LLNL provide basis for this finding. Based on this study, it is just too early to commit to any specific ESRL design. Much more work is required to develop the optimal systems concept.

Safety/Environmental Impact

Various safety and environmental impact issues were evaluated in this assessment. Some environmental effects are expected for ESRL development/construction. Sonic boom is not a "show stopper", however, localized effects are expected. Ear drum rupture is expected to occur at less than 100 m from the muzzle.

Rough estimates of risk for nuclear waste disposal indicate that for comparable risk with standard space disposal, overall system launch reliability would need to be on order of 0.999 to 0.9999 or better. Accident risks involving Earth orbital projectiles are expected to be no worse than current space activities.

Environmental impact benefits (although not great) may be possible by reducing the number of Space Shuttle flights. The quantity of effluents, the frequency of launch noise and sonic boom occurrence, and the impact to the ozone layer, by the HCl-Shuttle emission would be reduced.

Cost Estimates

Preliminary cost estimates for the ESRL system are given in Table 2. The costs are broken down into low, expected and high cost categories. It is assumed that both the Earth-orbital and nuclear waste missions would be accommodated by the ESRL system. The expected investment cost is given at \$5.4 B, with an annual operating expense of \$58 M, not including costs for projectiles and payloads. The cost for homopolar generators is the major investment cost item. Reference 1 discusses costs in detail.

For Reference Concept, with 10 launches per day, projectile costs dominate capital costs; the tungsten nose cone dominates projectile costs (we recommend substitute-steel); and propulsion system costs are significant (we recommend return via the Shuttle Orbiter to provide up to 20 reuses of the system).

While additional system trade studies and concept detail are needed to define the most cost-effective system, this preliminary analysis indicates that potential exists for more than an order of magnitude reduction in space transport costs for bulk materials over conventional systems (e.g., Space Shuttle, etc.). Table 3 provides a comparison of space transportation costs for different missions and space transport systems. Significant reduction in the cost of space disposal of nuclear waste make it now feasible to consider launching all the waste, thus possibly eliminating the need for mined geologic respositories.

Technology Requirements

Work is needed in many areas to improve the ESRL concept and cost estimates. These areas include: experimental research, system studies, and special studies. Battelle has recommended that a 5-year $3 M supporting research and technology program be conducted by NASA to further evaluate the potential and benefits of ESRL systems.

1.8 Conclusion

Based upon this preliminary assessment, Battelle concludes that Earth-to-Space Electromagnetic (railgun) Launchers appear to be technically feasible and environmentally/economically beneficial. However, needed progress in technology indicates that an operational ESRL system of the size contemplated in this study is expected to be achievable after the year 2020.

1.9 Recommendations

Battelle recommends to NASA's Lewis Research Center, on the basis of this study, that NASA should proceed with a moderate 5-year SR&T program to resolve unknowns and to reassess concept feasibility at the end of the 5-year period.

2.0 EARTH-TO-SPACE RAIL LAUNCHER (ESRL) SYSTEM REQUIREMENTS

This section defines the requirements developed for the ESRL system. A brief mission summary is provided, followed by a list of general requirements for the ESRL system. Lastly, a discussion of likely safety requirements for nuclear waste disposal in space is provided. These requirements have guided the development of the ESRL reference concept discussed in Section 4.0.

2.1 Mission Summary

Earth-to-Space Rail Launcher (ESRL) system is envisioned to be a multi-purpose space launcher, operating from an uninhabited region near the Earth's equator. The primary application would be to dispose of high-level nuclear waste in space. Secondary applications would include the launching of planetary probes, low Earth orbit satellites, and basic materials for use in space. The launch of basic materials could involve: structural materials used in the fabrication of large space structures; chemicals used in space-based manufacturing; propellants for orbit transfer operations; and substances for life support functions. Additional applications involve the conduct of high altitude research projects (suborbital launches). The ESRL system requires two separate launch tubes, each having a variable launch velocity capability to accommodate the wide range of possible use of the system.

2.2 General ESRL System Requirements

The general performance requirements for the Earth-to-Space Rail Launcher (ESRL) system are provided below:

(1) Launching site to be located near the equator, in a remote location, with clear weather.

(2) The maximum acceleration for payloads will be 10,000 g's.

(3) Delivered rail launcher velocities shall be: (1) variable between 5 and 10 km/s for Earth orbit and suborbital applications, and (2) for space disposal of nuclear waste, have a velocity at the surface of 20.0 km/s. The Earth orbital rail launcher will be fixed at an azimuth of 90°E with an elevation angle of 20° from the horizontal; the waste launcher will be fixed at an elevation angle of 90° from the horizontal.

(4) The ESRL system for launching waste is to be designed such that 0.5 MT of unshielded cermet HLW (with Cs and Sr removed) can be disposed per average day. The use of multiple shots can be considered if one single shot per day is not technically feasible. Proper shielding of the cermet waste form along with atmospheric flight thermal protection system TPS will be

required. Table 2-1 provides data showing the availability (1989 through 2000) of waste form for space disposal.

(5) The launched payload must be designed to properly survive atmospheric flight, with design consideration given to the flight stability problem.

(6) At a minimum, the reference mission facility should be able to support one launch per day of nuclear waste (meeting the requirements of Item #4).

(7) To maintain low-life cycle costs, the system will require as little maintenance and refurbishment as possible.

(8) Launch constraints related to cloud cover and wind direction are likely to be imposed, especially for waste launches.

(9) Materials for the construction of the ESRL system and payloads shall not adversely impact the availability of the materials to other users.

(10) The ESRL system shall have provisions for rescue and recovery of nuclear waste payloads, if a launch failure occurs.

TABLE 2-1. U.S. COMMERCIAL MODIFIED HIGH-LEVEL NUCLEAR WASTE AVAILABLE FOR SPACE DISPOSAL (IN CERMET FORM)

Year Waste Available	Kilograms of HLW (Cs and Sr Removed), Cermet Form
1989	279,000
1990	85,000
1991	100,000
1992	115,000
1993	131,000
1994	149,000
1995	164,000
1996	166,000
1997	188,000
1998	198,000
1999	206,000
2000	212,000
	1,993,000

Source: Adapted from data in Rice et al, 1982.

2.3 System Safety Design Requirements for Nuclear Waste Disposal ESRL Concept

This section defines system safety design requirements for the reference ESRL nuclear waste disposal in space mission. These requirements provide the guidelines against which a rail launched nuclear disposal in space may be considered acceptable from a radiological safety point of view.

The general safety design objectives for the disposal mission are: (1) to contain the solid radioactive waste material; (2) to limit the exposure of humans and the environment to the radioactive waste materials; and (3) mitigate the potential non-radiological environmental effects of operations. For normal operations, complete containment and minimum radiological exposure are required; for potential accident situations, the degree of containment and degree of interaction shall result in an acceptable risk to humans and the environment.

The following subsections describe the general and specific system design requirements for the ESRL nuclear waste disposal in space mission.

2.3.1 General System Safety Design Requirements for Nuclear Waste Disposal Mission

The general system safety design requirements for the ESRL system, as applied to the ESRL nuclear waste disposal in space mission, involve considering of the following:

(1) Non-Radiological Safety

(2) Radiation Exposure

(3) Containment

(4) Accident Environments

(5) Criticality

(6) Postaccident Recovery

(7) Monitoring Systems

(8) Isolation.

The following paragraphs define the requirements that should be followed for the reference system concept design activity.

2.3.1.1 Non-Radiological Safety

Consideration of the non-radiological safety aspects of the use of the ESRL concept shall be given to the design and siting of the ESRL launcher facilities. Appropriate precautions shall be taken to minimize worker impacts and hazards as well as those imposed upon worker's families located near by. Protection against sonic boom overpressure is the major concern. Also, the launch site location should be selected such that ground tracks for launched payloads shall be over the open ocean and be distant from populated areas such that sonic boom is not a significant impact.

2.3.1.2 Radiation Exposure

Radiation exposure limits for normal operations for the public and ground crews will be those contained in ERDA-MC0524 (U.S. DOE, 1975) and shown in Table 2-2. The normal radiation exposure limits for the current terrestrial transportation of nuclear waste materials would also apply to ground transportation of nuclear waste payloads. The radiation limits (49 CFR 173.393)* are given as:

- 1 m from external container surface...1000 mrem/hour
- External surface of transport vehicle...200 mrem/hour
- 2 m from external surface of transport vehicle...10 mrem/hour
- Normally occupied position of transport vehicle...2 mrem/hour.

For accident conditions of terrestrial transport, dose rates are limited to 1000 mrem/hour at 1 meter from the external surface of the waste package. For launch/reentry accidents, higher dose limits are expected because of the anticipated lower probability and remote locations for these accidents.

A general guideline for the waste package launched into space is that the radiation dose at 1 meter from the flight radiation shield surface is not greater than 10 rem per hour. The shield is to be carried all the way to solar system escape.

2.3.1.3 Containment

The containment requirements are different for the various portions of the ESRL disposal mission. For the reference ESRL mission, four different types of containment configurations are used: (1) shipping cask/auxiliary shielding/flight radiation shield/container/waste form, (2) auxiliary shielding/flight radiation shield/container/waste form, (3) launch sabot/TPS/structure/flight radiation shield/container/waste form, and (4) TPS/structure/flight radiation shield/container/waste form. For all normal operations, the systems will be designed such that no release of radioactive material occurs.

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*Note: Existing United States Nuclear Regulatory (NRC) regulations are quoted frequently in this section. 49 CFR 273 refer to Part 273 Part 49, Code of Federal Regulations.

TABLE 2-2. NORMAL OPERATIONS EXPOSURE LIMITS FOR INDIVIDUALS IN CONTROLLED AND UNCONTROLLED AREAS

INDIVIDUALS IN CONTROLLED AREAS:

Type of Exposure	Exposure Period	Dose Equivalent (Dose or Dose Commitment[a], rem)
Whole body, head and trunk, gonads, lens of the eye[b], red bone marrow, active blood forming organs.	Year Calendar Quarter	5[c] 3
Unlimited areas of the skin except hands and forearms). Other organs, tissues, and organ systems (except bone).	Year Calendar Quarter	15 5
Bone.	Year Calendar Quarter	30 10
Forearms.[d]	Year Calendar Quarter	30 10
Hands[d] and feet.	Year Calendar Quarter	75 25

INDIVIDUALS IN UNCONTROLLED AREAS:

Annual Dose Equivalent or Dose Commitment (rem)[e]

Type of Exposure	Based on dose to individuals at points of maximum probable exposure	Based on an average dose to a suitable sample of exposed population
Whole body, gonads, or bone marrow	0.5	0.17
Other organs	1.5	0.5

(a) To meet the above dose commitment standards, operations must be conducted in such a manner that it would be unlikely that an individual would assimilate in a critical organ, by inhalation, ingestion, or absorption, a quantity of a radionuclide(s) that would commit the individual to an organ dose which exceeds the limits specified in the above table.
(b) A beta exposure below an average energy of 700 Kev will not penetrate the lens of the eye; therefore, the applicable limit for these energies would be that for the skin (15 rem/year).
(c) In special cases with the approval of the Director, Division of Operational Safety, a worker may exceed 5 rem/year provided his average exposure per year since age 18 will not exceed 5 rem per year.
(d) All reasonable efforts shall be made to keep exposures of forearms and hands to the general limit for the skin.
(e) In keeping with ERDA policy on lowest practicable exposure, exposures to the public shall be limited to as small a fraction of the respective annual dose limits as is practicable.

Source: U.S. DOE, 1975.

Configuration (1) must survive probable land and ocean shipping accidents without major release. Configuration (2) must survive probable handling accidents without major containment breach. Configuration (3) must survive all handling, and most launch facility accidents without major containment breach. Configuration (4) must be designed to survive the normal launch trajectory with no release and off-nominal trajectories with atmospheric reentry with no major release. Malfunction of the rail launcher should result in no major risks to man. The accident environments for which the designs of these generic configurations must survive are given below.

2.3.1.4 Accident Environments

The accident environments that need to be considered in the design of containment and other auxiliary systems are as follows:

- Shipping accident
- Ground handling accident at the ESRL Launch Facility
- Accidents/malfunctions during the acceleration of the payload
- Reentry accidents.

2.3.1.4.1 Shipping Accident Environments (for Configuration 1). DOT and NRC regulations, as defined in 49 CFR 170 to 179 and 10 CFR 71, will be assumed for the ground shipment of nuclear waste payloads to the rail launcher site. The following sequential test environments for shipping cask accidents are given below. Initial conditions are to be assumed the same as the normal condition.

- A 9-m drop in worst orientation onto an unyielding surface
- A 1-m drop in the worst orientation onto the end of 15-cm-diameter, 20-cm-high bar (mild steel)
- A 30-minute ground fire at 800 C followed by 3 hours of no artificial cooling; with a cask emissivity of 0.9 and cask absorbtivity of 0.8
- An 8-hour emersion in 0.9 m of water.

At the end of this test, surface radiation of the shipping cask should not exceed 1 rem/hour at 1 m from the surface, the contents must remain subcritical, and only minute radioactive material releases are allowed (see 10 CFR 71).

2.3.1.4.2 Handling at Launch Facility (for Configurations 2 and 3). The payload systems, auxiliary support equipment and facilities must be designed to minimize the occupational radiation exposure to workers (see Table 2-2). Care must also be taken to insure that if certain subsystem failures occur during handling at the Launch Facility, radiation exposure is kept to as low as reasonably achievable (ALARA). The handling area at the Launch Facility will be designed to be a total containment vessel.

2.3.1.4.3 Accidents/Malfunctions During the Acceleration of the Payload (for Configuration 3). The payload package must be designed to withstand the following nominal accidents that can occur after the acceleration phase has begun without a major breach of primary containment. Initial conditions are assumed to be the normal condition.

- Rail launcher structural failure
- Sabot/payload structural failure
- Plasma breakdown ahead of payload, followed by rapid deceleration
- Insulation failure with current through the body of the payload
- Failure of the rail's nominal environmental support subsystems
- After misfire, the payload falls upon its starting position.

2.3.1.4.4 Reentry Accidents (for Configuration 4). The TPS/structure/flight radiation shield/container/waste form configuration must be able to withstand reentry into the Earth's atmosphere and without the dispersion of significant quantities of waste form into the atmosphere. The design reentry environment is defined as a reentry trajectory which provides the maximum heating flux possible. The payload shall be designed to be a high drag reentering body (low drag in departure) such that survivable heating and survivable impact velocities are predicted.

2.3.1.5 Criticality

The radioactive waste package shall be subcritical (calculated K-effective $+3\sigma \leq 0.95$) for normal operations or any possible credible accident during processing, fabrication, handling, storage, or launch to the space destination. Calculations should show that any credible change in waste form geometry and any credible grouping of packages will not cause K-effective $+3\sigma$ to exceed 0.95.

2.3.1.6 Post-accident Recovery

Post-accident recovery teams will be made part of the operational disposal system. They will be responsible for all accident recovery operations, including accidents involving processing, payload fabrication and railroad or ship transport, payload preparation at the rail launcher site, the launch, and possible accidental reentry.

2.3.1.7 Monitoring Systems

Various monitoring systems will be used for the overall system such that overall mission safety can be assured. Examples of such systems include devices for measuring radiation, and temperature, and instruments to provide data for tracking the payload after it leaves the Earth's atmosphere. Permanent labeling will specify the waste contents history and radiation projection.

2.3.1.8 Isolation

The nominal space destination should insure, at a minimum, an expected isolation time from the Earth's biosphere in excess of one million years, and should not adversely interfere with normal space operations projected to be carried out by future generations. Careless contamination of celestial bodies should be avoided. Solar system escape with proper launch windows satisfies this requirement.

2.3.2 Specific System Safety Design Requirements for Nuclear Waste Disposal ESRL Concept

The following paragraphs define specific system design requirements established for the elements of the reference ESRL concept.

2.3.2.1 Waste Processing and Payload Fabrication Facilities

The design and operation of these facilities will follow current proposed regulations, as specified for nuclear waste reprocessing plants. It will be assumed that the waste is processed within the continental U.S.

2.3.2.2 Payload Nuclear Waste Mix and Form

The waste mix is defined as a high-level Purex waste with 95 percent of the Cs and Sr removed (Rice et al, 1982). The waste form will be the ONRL iron-mixed based cermet. For normal and accident conditions, the cermet fabrication temperature of 1050 C shall not be exceeded (Rice et al, 1982). Properties of the cermet waste material are as follows (Rice et al, 1982):

- Structural properties = similar to Hastelloy C
- Density = 6.5 g/cc
- Thermal conductivity = 9.5 W/m-C
- Specific heat = 0.14 calories/g-C
- Heat generation rate = 1.00 W/kg of cermet (based on aged waste cited, Rice, 1982).

2.3.2.3 Payload Primary Container

For normal conditions, the primary stainless steel container shall not exceed a temperature of 416 C (Rice et al, 1981). No chemical and physical interaction will occur between the cermet waste form and the container. For typical accident conditions, the primary container must not exceed the temperature of 1280 C (90 percent of melt absolute temperature--see Rice et al, 1982).

2.3.2.4 Payload Flight Radiation Shielding

Radiation shielding for flight systems will be designed to limit radiation to no more than 10 rem per hour at 1 meter from the shield surface under normal conditions. Auxiliary shielding will be designed such that radiation exposure limits (see Table 2-2) for ground personnel are not exceeded during operations.

The primary candidate material for the radiation shield is high-strength steel. For normal conditions, the temperature limit for the flight radiation shield is 416 C for steel. For accident conditions, the radiation shield should not exceed a temperature of 1280 C for steel (see Rice et al, 1981).

2.3.2.5 Payload Thermal Protection

The payload thermal protection systems must include provisions to adequately survive the expected launch and reentry environments.

2.3.2.6 Sabot System

The sabot system will be properly configured to be highly reliable during the acceleration phase and break up and off the payload when leaving the rails.

2.3.2.7 Payload Instrumentation Systems

The payload should include provisions for a transmitter operable in the ocean and in space which can be used for tracking and/or rescue.

2.3.2.8 Launched Payload Mass

The total average daily launched, unshielded, waste form payload mass shall be 0.5 MT. Within this constraint and others, the nominal payload mass shall be determined.

2.3.2.9 Shipping Casks and Ground Transport Vehicles

Shipping casks and ground transport vehicles will comply with DOT and NRC regulations. The maximum outside diameter of the shipping cask will be 3.05 meters (10 feet).

2.3.2.10 Rail Launch Facilities

It is desirable that the launch facilities used for the reference nuclear waste disposal mission will be a dedicated ocean remote facility (island, platform, ship) located a reasonable distance from human population centers.

2.3.2.11 Rail Launcher System

The system shall be designed to be as efficient and reusable as possible within the projected SOA (beyond the year 2000). The launcher shall not provide payload accelerations in excess of 10,000 g's. The rail launcher for nuclear waste missions (and planetary probes) will be pointed straight up (90 degrees from the horizontal).

2.3.2.12 Reentry High-Drag Device

The payload will be designed to reenter the Earth's atmosphere, in the event of an aborted mission, as a high-drag projectile, such that it can survive atmospheric flight and ground impact without breach of containment.

2.3.2.13 Space Destination

The space destination will be solar system escape with an excess solar system escape velocity of at least 1 km/s. The ideal vacuum minimum velocity requirement from the Earth's surface for this mission is 16.67 km/s, including the excess velocity. To allow for drag and non-ideal launch conditions, a launch velocity of 20 km/s will be required for the system.

2.4 Accident and Malfunction Contingency Plans

Accident and malfunction contingency plans for the general phases of the ESRL system applied to the space disposal mission are listed and addressed below:

- Surface transportation from the payload fabrication sites to the launching site
- Preflight operations prior to the launch signal

- Rail launch operations from the launch site to the achieving of the destination.

2.4.1 Surface Transportation

Ground and ocean transport (via rail and ship) of the shipping cask would be assigned to the U.S. Department of Energy (DOE) which would supply the necessary accident recovery plans and systems. At least two types of incidents must be considered: loss of cooling to the waste container and possible breach of the waste container with a loss of radioactive material.

In case of cooling loss, provisions must be made to have self-contained, auxiliary cooling units on line within an appropriate time such that no waste melting condition is met. Monitoring equipment for both container temperature and radiation will be required during all ground transport operations.

A continuous capability to cope with a container breach will be necessary. A specially trained accident recovery crew will always be ready to act, if necessary.

2.4.2 Handling at the Launcher Site

Contingency plans must be provided for potential malfunctions and accidents that could occur while the waste payload is at the Rail Launcher Facility, being configured in the ESRL itself and awaiting launch. Accidents and contingency plans would be similar to those discussed in Section 2.4.1, above.

2.4.3 Launch Operations

Contingency plans, procedures and systems envisioned to minimize the launch hazard, are given below. These plans, procedures and systems would minimize the probability of a release of radioactive material into the environment and/or reduce its effect upon the human population.

- Stringent containment systems designs to maximize the probability of surviving the possible hostile accident environments.

- The use of a waste form not easily dispersed under adverse conditions.

- The application of appropriate space disposal mission launch constraints (e.g., wind direction) to reduce human radiological exposure resulting from a potential containment breach.

- The use of a payload recovery team ready to rescue the payload at sea or on land.

- Restrictions on the use of air and sea space in the vicinity of the launch site.

- The use of redundant (backup) systems where possible to ensure a high level of system performance.

- Payload tracking via ground based systems, as the payload leaves the launcher muzzle.

- On-orbit payload tracking via satellite, as the payload leaves the Earth.

- The conduct of proper trajectory analysis to insure: (1) that no significant orbital perturbation with other planetary bodies in the solar system will occur, and (2) that other Earth orbit satellites and space program operations will not be threatened by a possible collision.

- The proper application of thermal protection materials on the outside of the payload to reduce the risk of waste containment breach and atmospheric dispersal.

- The use of a high-drag device to reduce reentry velocities in the atmosphere as well as provide a survivable terminal velocity at the ground.

- The use of high-melting point radiation shield and container material to reduce the risk of atmospheric disposal.

- Provisions may be made to rescue the nuclear waste payload in solar orbit in the event of failure to reach solar system escape. The approach is to rendezvous and dock a rescue orbit transfer vehicle with an "uncooperative" payload and place the payload in a "safe" disposal orbit (likely to be a solar orbit at 1.15 AU orbit).

3.0 ESRL SYSTEMS ANALYSES

This section discusses a variety of Earth-to-Space Rail Launcher (ESRL) systems analysis that was conducted during this study. This section discusses activities conducted by Battelle Columbus Laboratories, the University of Texas at Austin, Lawrence Livermore National Laboratory, IAP Research, Inc. and PAI Corporation in Dayton, Ohio, and The Ohio State University. Work reported here and conducted by Battelle involved four technical areas: (1) identification of concept options, (2) radiation shielding analysis, (3) launch velocity requirements and launch window analysis, and (4) preliminary conceptualization of projectiles/payloads. Analysis conducted by the University of Texas at Austin by Dr. Richard Marshall concentrated on railgun analysis for distributed energy stores (DES). Analysis conducted by Lawrence Livermore National Laboratory was conducted by Mr. Ron Hawke, who assessed the multi-stage segmented railgun system, and work done by Dr. Alfred Buckingham on the aerothermal and drag aspects of projectile concepts for the ESRL system. Dr. John Barber from IAP Research, Inc. in Dayton, Ohio, conducted analysis for a single-stage railgun for Earth-to-Space rail launched projectiles. Mr. Hal Swift of the PAI Corporation in Dayton, provided basic information on projectile shapes and sabot technology for this study. Work conducted by The Ohio State University Aeronautical and Astronautical Research Laboratory under the direction of Dr. John Lee, Laboratory Director, involved aspects of hypersonic aerodynamics, ablation, and projectile design.

Initial presentations by the above named people were made at the Concept Definition Meeting for ESRL concepts at Battelle Columbus Laboratories on August 12 and 13, 1981. Vu-graph material presented at that meeting was documented and distributed to attendees and other interested parties (some of this material is provided in Appendices at the end of this report). Additional analysis was conducted after the meeting by J. Lee and the three railgun experts, namely, R. Marshall, J. Barber, and R. Hawke. These new results, along with the old material, are summarized at the end of this section.

3.1 Battelle Columbus Laboratory Analysis

As mentioned above, Battelle conducted analyses on: (1) identification of concept options; (2) radiation shielding; (3) launch velocity requirements and launch windows, and (4) preliminary conceptualization of projectiles. Results of these analyses are presented in the following subsections.

3.1.1 Identification of Possible ESRL Options

This section discusses possible ESRL options that were identified during the study. The discussion in this section pertains to an overview of the reference concept options that have been selected and why. The various aspects of the overall mission are discussed in the following sections.

3.1.1.1 Possible Applications, Missions and General Requirements

During the early portion of the study, several staff meetings were held to discuss possible applications for an Earth-to-Space rail launching system. Based upon these meetings, basically eight types of mission applications were identified.

(1) Nuclear Waste Disposal in Space
(2) Earth Orbital Applications
(3) Space Probes
(4) Atmospheric Research
(5) Assist Chemical Rocket Launches
(6) Hybrid Propulsion
(7) Lunar Gravity Assist Missions
(8) Toxic Chemical Disposal in Space.

A general discussion of these mission candidates is provided in Section 8.0 and is not discussed further here, except in the context of the two reference missions that have been selected, namely, nuclear waste disposal in space (Mission A) and Earth orbital application missions (Mission B). Figure 3-1 provides an overview of the possible options for these two missions. These are discussed in the next few sections.

3.1.1.1.1 Auxiliary Propulsion

For the nuclear waste disposal in space mission, no auxiliary propulsion is required. This is because solar system escape is the destination and the total velocity impulse would have been supplied at the surface of the Earth. For Earth orbital applications missions, additional propulsion is required to place the payload into an orbit about the Earth. As will be discussed in Section 3.1.5 (entitled Projectile Concepts), Earth storable propellants were selected for the propulsion systems. This selection was based upon the need for simplicity and high density in the propellants. Also, high performance was desirable to maximize the payload that could be carried. Cryogenic propellants, namely, hydrogen, would not be feasible because of its low density. A high acceleration solid propellant propulsion system was considered in the evaluation, but it was believed that a liquid propellant system would provide better performance at lower risk. Structural analysis is required to evaluate the use of solid propellants (likely an end burner) at the expected high-g loadings.

3.1.1.1.2 Payloads

The payload for nuclear waste disposal in space, employing ESRL system, was selected based upon the results of most recent studies by NASA and the Department of Energy (Rice et al, 1982; McCallum et al, 1982; and Reinert et al, 1982). The nuclear waste mix selected for the space disposal payload is discussed in more detail in Section 4.0. The selected nuclear waste mix consists of high-level waste from commercial nuclear power plants in the United States. It is assumed that the bulk of the uranium and plutonium have been processed out of the waste and also that cesium and strontium have been

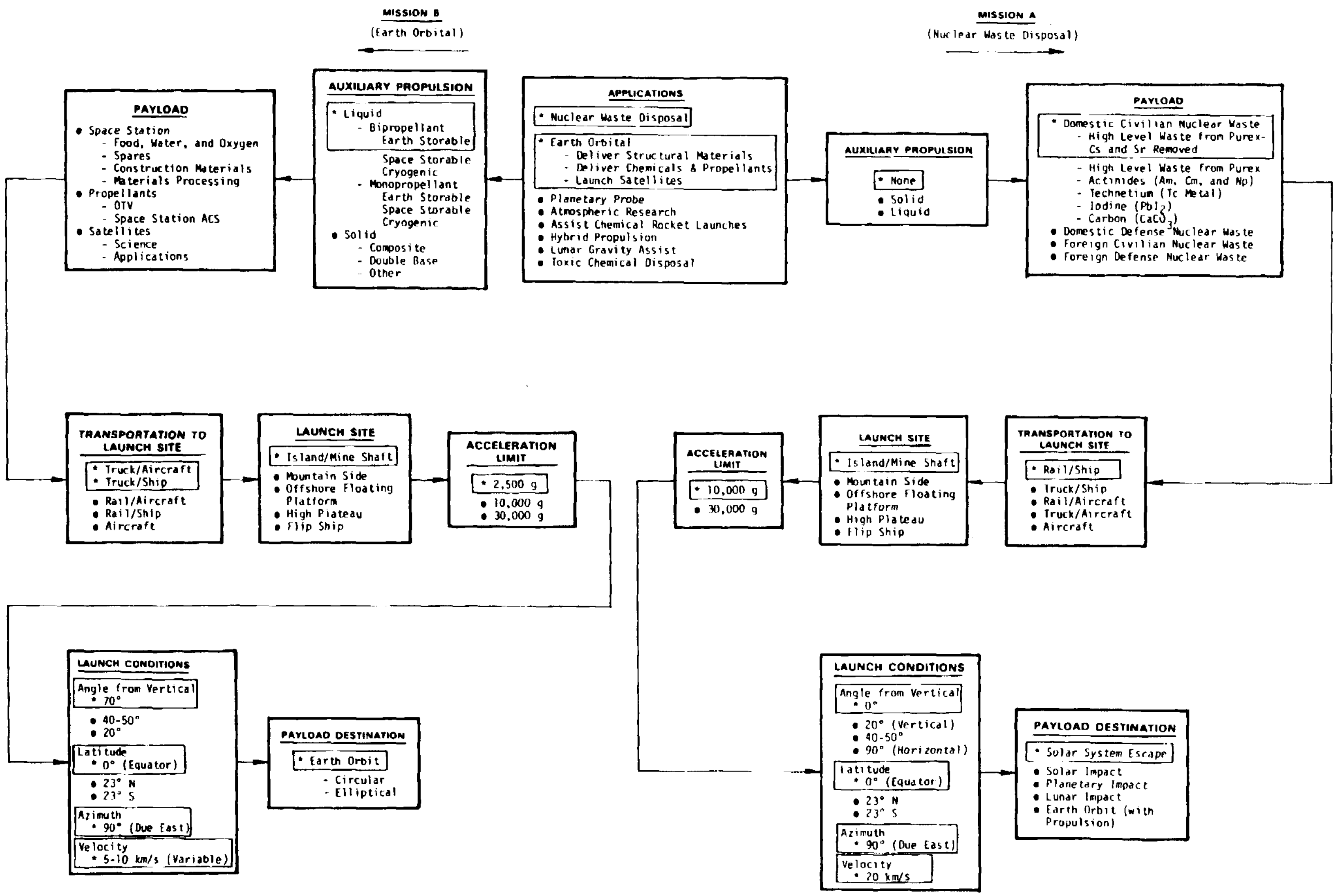


FIGURE 3-1. OVERVIEW OF ESRL REFERENCE CONCEPT OPTIONS

removed and taken to the mined geologic repository. Another waste mix possible is high-level waste from the Purex process, where cesium and strontium are not removed; this would imply larger masses of waste and higher heat loads. Other possibilities include payloads consisting of the actinides, comprised of Am, Cm, and Np, along with the possibility of some of small quantities of Pu and U. Should specific elements or isotopes like iodine, technetium, and carbon prove to be a problem for the mined geologic repository, then disposal of these elements in space could be warranted. The total mass of these specific elements, however, is quite small and would not likely, in and of itself, justify the development of an Earth-to-space rail launcher system. Additional options include the possibility of disposing of defense nuclear waste and foreign (civilian and defense) waste from the international launch site.

Payloads for Earth orbital applications missions include: materials for space manufacturing, space station spares and new construction materials, propellants for support of orbit transfer vehicle operations and life support functions (food, water, and oxygen) on space stations. It might be advantageous to launch water into space by the ESRL system which could later be converted into hydrogen and oxygen components in space, by the use of solar energy. Also, small scientific and applications satellites could be launched to various Earth orbital altitudes on a quick response basis. It also is possible to launch materials directly into geosynchronous orbit.

3.1.1.1.3 Transportation to the Launch Site

Because of a selection of a remote island launch site, surface transportation of nuclear waste for disposal in space is destined to be accomplished by a combination of rail transport on the mainland and ship transport on the open seas to the launch site. It is unlikely that trucks would be used to transport the nuclear waste to the coast. It is also unlikely that aircraft would be used to transport large, heavy nuclear waste shipping casks.

On the other hand, for Earth orbit applications missions, it is likely that certain materials would be shipped to the launch site by aircraft; although for bulky, heavy materials, ships would likely be used.

3.1.1.1.4 Launch Site

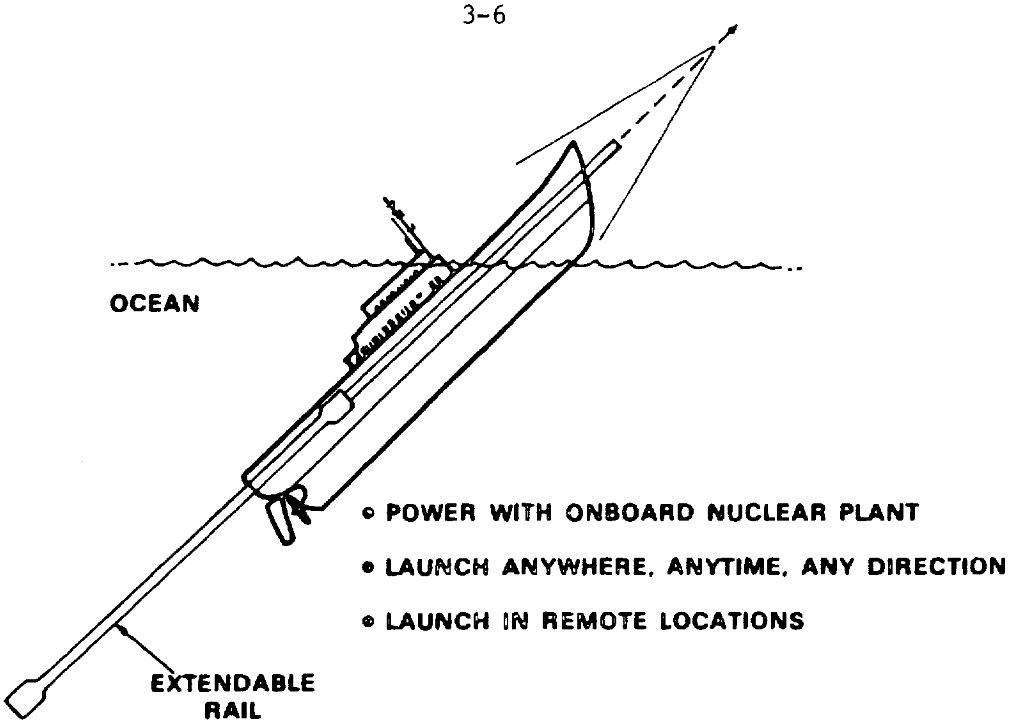
Based upon the Concept Definition Meeting held at Battelle in August, 1981, a consensus was given to select a remote island launch site with a mine shaft for the launcher tube. Other launch site options that were considered included: launching up a mountain side, developing an offshore floating platform, launch along a wall of a high plateau or cliff (a launcher attached along the side), and the possible development of a flipship, where a ship could house the entire railgun facility and be pointed any direction and could provide a launch site at any deep sea area. A concept not listed in Figure 3-1 was a remote island launch site with a hollowed out water section for a variable positioned launcher tube. The flipship concept and the remote island variable launched azimuth/elevation angle concept are shown in Figure 3-2. The driving force behind these two concepts at the time they were conceived was that the Earth-to-space rail launcher system required a variable launch direction to accommodate: (1) the possibility of a multitude of different types of missions; and (2) the changes caused by the Earth's seasons.

3.1.1.1.5 Acceleration Limits

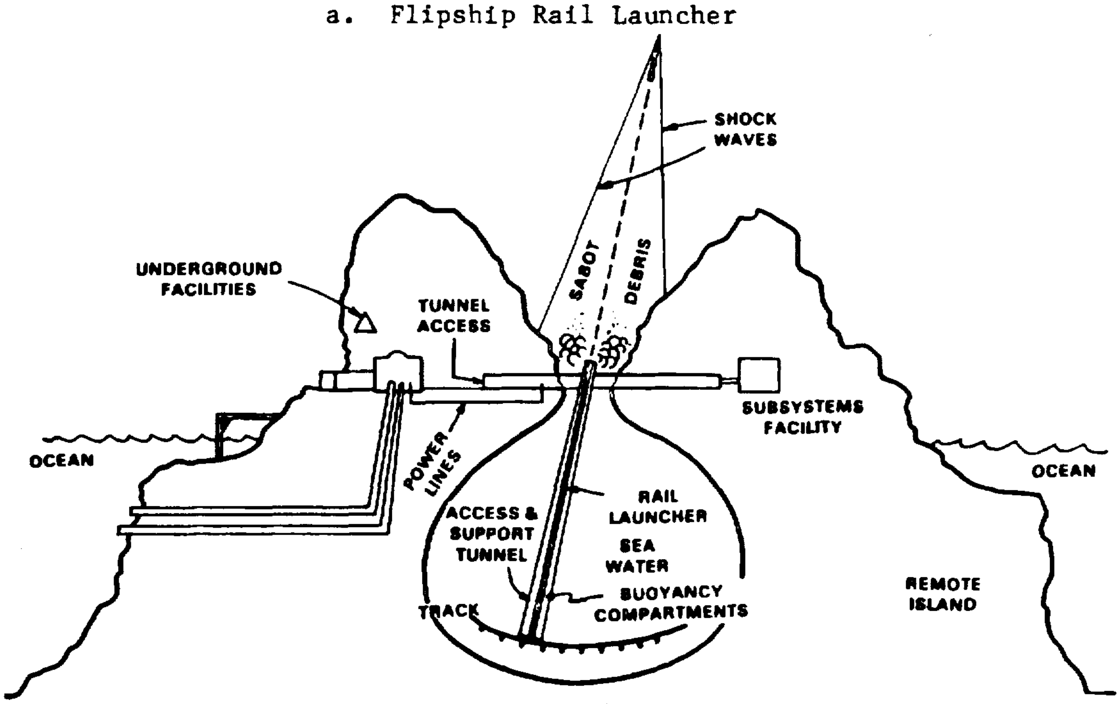
It is desirable to keep the rail launcher as short as possible, and yet keep the acceleration as low as possible, such that the payload will survive the launch environment. Initially, an acceleration limit of 30,000 g's was established as the maximum allowable acceleration for nuclear waste disposal. It was believed that 30,000 g's would allow survival of instrumentation that could be carried onboard the projectile. However, because of the stress created in the projectile structure from such high accelerations, it was decided to back away from the 30,000 g value to 10,000 g's in the launcher tube. For the Earth orbital application mission, it was decided to limit the acceleration to 2500 g's (at 10 km/sec), such that more payloads could be carried to Earth orbit, including satellite systems specially designed to withstand high-g forces. Various gun launched projectiles that contain instrument packages have adequately survived 10,000 g's, and possibly could have survived up to 30,000 g's had they been tested that high (personal communication, Mr. Bill Williams, Martin Marietta, Orlando, Florida).

3.1.1.1.6 Launch Conditions

After an initial decision was made that it did not appear practical to have an all-variable launch azimuth rail launcher tube, fixed launcher tubes at various angles to the vertical were evaluated. Initially, the possibility of using the same launcher tube for space disposal and Earth-orbital applications missions was considered. Although this concept is possible, severe performance penalties occur for the Earth-orbital applications missions, as well as the nuclear waste disposal in space mission. Therefore, it was decided to decouple the two launch requirements and to provide separate launcher tubes for these applications. Because Mission A was the primary driver for the ESRL launching site, an equatorial launch location was desirable. Also, it was determined (see peak curve on Figure 3-13) that roughly a 20 degree inclination from the vertical toward the east was about optimal for nuclear waste disposal in space missions or planetary probes. This would take advantage of the Earth's rotational component and would not change launch window conditions significantly. However, for simplicity in concept, a zero-degree angle from the vertical was selected for the reference case for Mission A. As is discussed in Section 3.1.3, a velocity requirement of 20 km/s was identified as being needed to perform the nuclear waste disposal in space missions having a launch window of about six hours with the first daily launch occurring near the beginning of the launch window (about 4:00 am local time) and the second occuring near the end of the launch window. For the Earth orbital applications, the angle from the vertical of 70 degrees was selected for the reference case. Section 3.1.3 discusses how this value was arrived at. The velocity requirements are also defined in Section 3.1.3.



a. Flipship Rail Launcher



b. Remote Island Variable Azimuth/Elevation Angle Launcher

FIGURE 3-2. POSSIBLE ESRL LAUNCHING CONCEPTS FOR VARIABLE LAUNCH AZIMUTH AND ELEVATION

They range from 5 to 10 kilometers per second, depending on the particular mission application. It is assumed that all Earth orbital missions are launched due East into an equatorial orbit; this eliminates the complex plane change maneuver, allows payload to be maximized, and allows daily multiple launches.

3.1.1.1.7 Payload Destination

Solar system escape was selected as the primary destination for the nuclear waste disposal in space mission. It is the easiest to accomplish of all the candidate destinations listed in Figure 3-1. Solar impact requires more energy and much greater accuracy than the solar system escape destination. Planetary and lunar impact would probably not be politically acceptable by the science community. Earth orbit (via the use of a propulsion system) would probably not be acceptable because of the potential long-term hazards of reentry and on-orbit debris impact.

Payload destinations for Earth orbital missions can be circular or elliptical orbits ranging from several hundred kilometers altitude all the way up to and beyond geosynchronous orbit.

3.1.1.2 ESRL Payloads/Projectile Options

There are a multitude of payloads/projectile options that were identified in this study (see Figure 3-3). This section describes some of the aspects of the payload/projectile by discussing specific topics: (1) payloads, (2) projectile/sabot shape, (3) projectile stabilization, (4) thermal protection system (TPS), and (5) reentry decelerator systems.

3.1.1.2.1 Payload Options

For the nuclear waste disposal in space mission, there are many aspects of the payload which need to be discussed. First of all, the nuclear waste form selected was a cermet waste form, as recommended by the 1981-82 study: "Preliminary Risk Assessment of Nuclear Waste Disposal in Space" (Rice et al, 1982). Other waste forms are possible, but none appeared to be as well suited for space disposal. The cermet waste form is discussed in further detail in Section 4.0. Because of the need for a cylindrical aerodynamic projectile shape, the optimal waste form shape would also be cylindrical. Because the waste form is cylindrical in shape, the waste primary container and the nuclear waste radiation flight shield are also cylindrical. The material selected for the nuclear waste container and shield is high-strength carbon steel. Should the risk of losing a payload in the ocean be considerable, then a high-strength, highly corrosion resistant, Inconel alloy is recommended. This could preclude corrosion of the shield in the sea environment for about 50,000 years. Other metals are also possible, but carbon steel represents the most inexpensive of all.

The payload, as defined for the Earth orbital missions, has been previously discussed (see Section 3.1.1.1.2) and in greater detail in Section

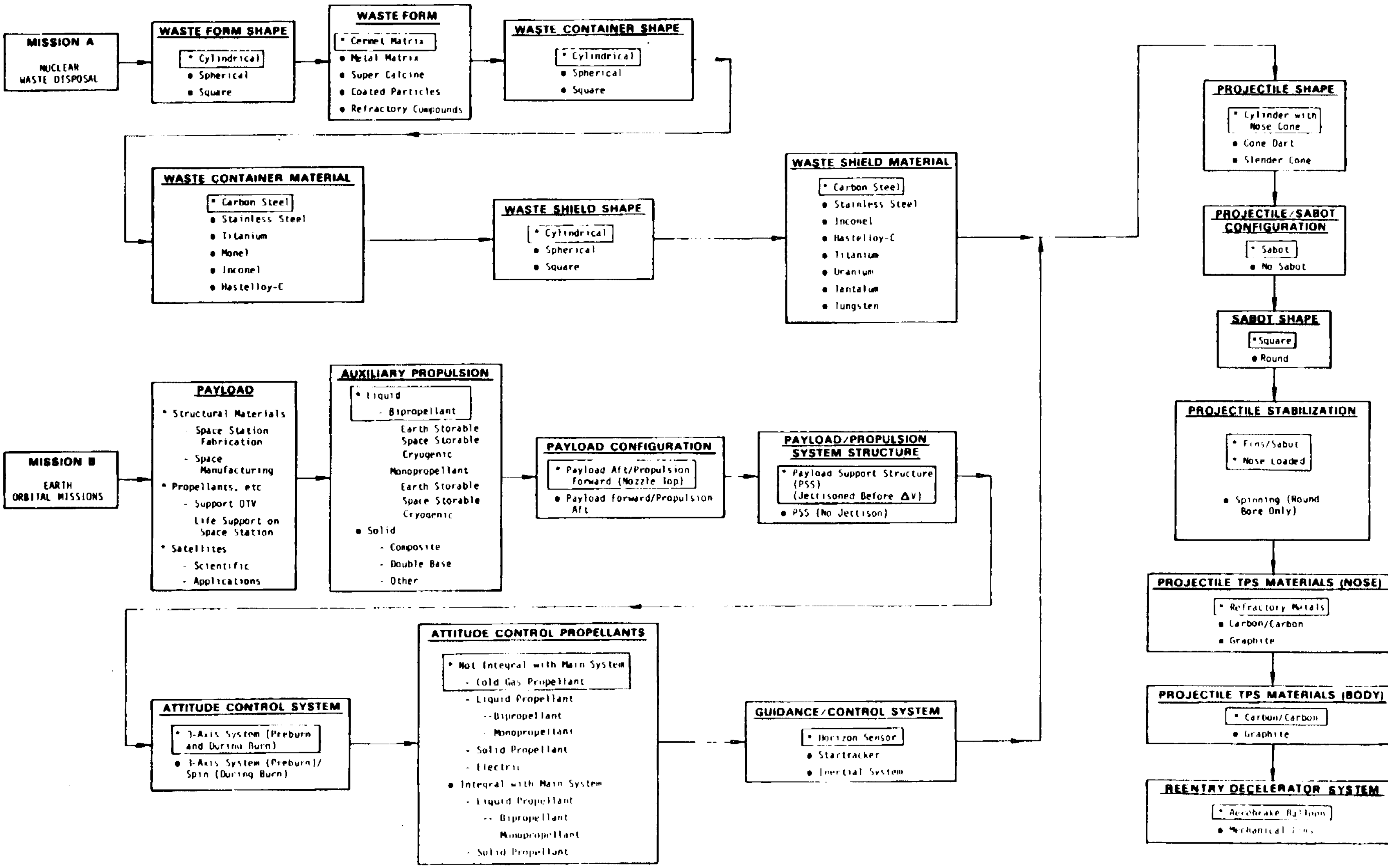


FIGURE 3-3. ESRL PAYLOAD/PROJECTILE OPTIONS

5.3.1.5 and 8.0. Basically, the payload may consist of structural materials for fabricating space stations, materials for supplying space manufacturing activities, chemicals, propellants which support orbit transfer vehicle operations or items which support life support on a space station, namely, food, oxygen, water, etc. Also, payloads could involve small scientific or application satellites. The total overall payload for the Earth orbital missions is not merely comprised of the payload usable in space, but also is comprised of an auxilliary propulsion system, attitude control system, and related guidance and control. These systems are required to place the payload in orbit around the Earth, otherwise, it would reenter before completing the first orbital pass. After a preliminary screening and evaluation, it was decided that liquid-propellant systems appear to be best suited for this application over solid-propellant systems. Earth storable hypergolic propellants are most desirable because they require minimum supporting systems. Also, they are advantageous because of the higher density available to them over cryogenic liquids such as hydrogen and oxygen. The propulsion system assumed is a pressure-fed type using high pressure helium. Regarding the "payload" configuration, the nozzle, propulsion system and the attitude control system (ACS) would be best placed at the nose of the projectile (i.e. nozzle up). This reduces the stress on the propulsion system as impressed on it by mass above it during the launch acceleration phase. To support the payload and the propulsion system during the high-g launch and to carry the loads of the nose cone and other materials, a supporting structure of some type is required. In the reference case, the payload support structure (PSS) would be jettisoned before the propulsion system performs its Δv maneuver. The PSS and nose cone would then reenter and fall into the ocean. The propulsion system could be recovered and returned to Earth via manned vehicles expected to be operational at that time.

It was assumed that the attitude control system (ACS) would be a cold gas (nitrogen) system which would have a three-axis capability during the preburn and burn phase. One option considered was to use the three-axis system to actually spin up the payload prior to the Δv burn. Figure 3-3 shows numerous options for the attitude control propellants. Nitrogen was believed to be the leading candidate, although it would be possible to use the liquid fuel as as a monopropellant (hydrazine).

The guidance and control system would have a computer and a horizon sensor that would be used to determine the proper timing/position for the Δv maneuver.

3.1.1.2.2 Projectile/Sabot Shape

The projectile shape is basically dictated by the aerodynamic behavior of the body. The most reasonable aerodynamic shape for railgun-launched applications would be a cylindrically shaped body with a nose cone and stabilization fins. It may be possible in the future to construct an ESRL launcher tube that has a circular bore to allow the spinup of the projectile (enhances aerodynamic stabilization). Experimental work done at LLNL in 1981-1982 demonstrated the use of round bores. Distinct advantages relating to rail survival appear possible. But, at this time, NASA decided to select the square bore, in keeping with current technology. The shape of the bore determines the shape of the sabot. So for the reference case, we have a sabot which takes a cylindrically shaped projectile and matches it to a square cross-sectional bore. One major advantage of having a square bore and a round projectile is that the fins on the projectile can be accommodated by the four diagonal corners of the square, and they can be easily contained in the sabot mass.

3.1.1.2.3 Projectile Stabilization

After considerable amount of discussion at the ESRL Concept Definition Working Meeting, it was recommended that the projectile have fins and a center of gravity nose forward. This recommendation was most strongly made by Dr. John Lee of The Ohio State University Aeronautical and Astronautical Research Laboratory. He indicated that the fins would not have to be very large, but adequate enough to aid stabilizing the vehicle as it departs the muzzle. The concept of a spinning projectile would be feasible and possible, but only in a round bore launcher. A significant amount of work remains to be accomplished on the aerodynamic stability problem for the projectile. Section 3.7 documents a preliminary assessment of the projectile flight stability problem.

3.1.1.2.4 Projectile Thermal Protection System

Various concepts for thermal protection system (TPS) were presented at the ESRL Concept Definition Meeting in August, 1981. Various recommendations were made and included the use of carbon/carbon materials on the sidebody and refractory materials for the nose tip. Dr. John Lee of The Ohio State University recommended very strongly that the nose of the projectile employ refractory metal that could smoothly melt away as it passes through the atmosphere. He felt this was of utmost importance in keeping the drag as low as possible. Dr. Al Buckingham of LLNL, believed that carbon/carbon would be the appropriate material for the nose tip. The consensus that resulted is that a refractory metal should be used for the nose tip and carbon/carbon material would be used for the sidebody. A considerable amount of analysis still remains to determine the overall characteristics of a thermal protection system for such a high velocity projectile traversing through the dense aatmosphere.

3.1.1.2.5 Projectile Reentry Deceleration Systems

Various reentry decelerator systems were identified and suggested by Battelle. The two basic options included: (1) mechanical blades that would be deployed to give a very large area, high-drag shaped body, and (2) a deployable and inflatable aerobraking ballute, similar to the ones conceptualized for the aerobraked orbit transfer vehicles studied by Boeing Aerospace Company (Boeing, 1981). The aerobrake concept was selected here; no analysis has been conducted.

3.1.1.3 ESRL System Options

Figure 3-4 provides a general overview of all the options considered for the ESRL system. The selected Reference Concept is represented by the asterisk in the boxes inside each major boxed area. The following subsections discuss each major area as indicated in the figure.

3.1.1.3.1 Energy Source

There are numerous and potential energy sources that could be used for the ESRL system, however, many are eliminated from consideration by the fact that a remote island was selected as the launch site. Because of the remoteness of the launch site and the considerable distance away from possible fuel sources, a nuclear power plant was selected for the Reference Concept. There could be measurable environmental impact if coal had to be transported and burned to supply the energy. Hydro would not likely be available on a remote island; solar energy is limited by the fact that on a small island it would be difficult to place solar collectors such that they would not be affected by sonic booms. Wind machines are also possible. Geothermal is a possible power source, but because of the non-specific nature of the island, the potential for geothermal energy is not known. Other advanced concepts are possible, but not considered to be viable at this time or in the time frame of the ESRL. It is believed that nuclear power is more practical than the other options. Utility power is basically not assumed to be available on a remote dedicated island.

3.1.1.3.2 Energy/Power Storage

Based upon the meeting held at Battelle in August, 1981, the consensus was that homopolar generators (HPGs) were ideally suited for this application. Based upon preliminary analysis conducted by Dr. Richard Marshall, the University of Texas, liquid nitrogen cooled inductors were recommended for the reference ESRL system. This was because of a significant reduction in mass required. It was not believed that super conducting would be required for a practical system. Other options, such as chemical and explosive flux generators, capacitors, batteries, and MHD were identified as possibilities, however, no one felt that these would be better than the HPGs for this application.

3.1.1.3.3 Energy Store Distribution

Based on the results of the Concept Definition Meeting in August, 1981, three types of energy stores distribution were identified. The three are: (1) distributed integral; (2) distributed segmented; and (3) single. The distributed energy store (DES) that is integral with the rail was selected for the Reference Concept because of its potential for high efficiency and performance. The distributed segmented rail was well thought of but the concept does not show the performance promise that the DES system does. The single system would have a lower efficiency and, while easy to construct from

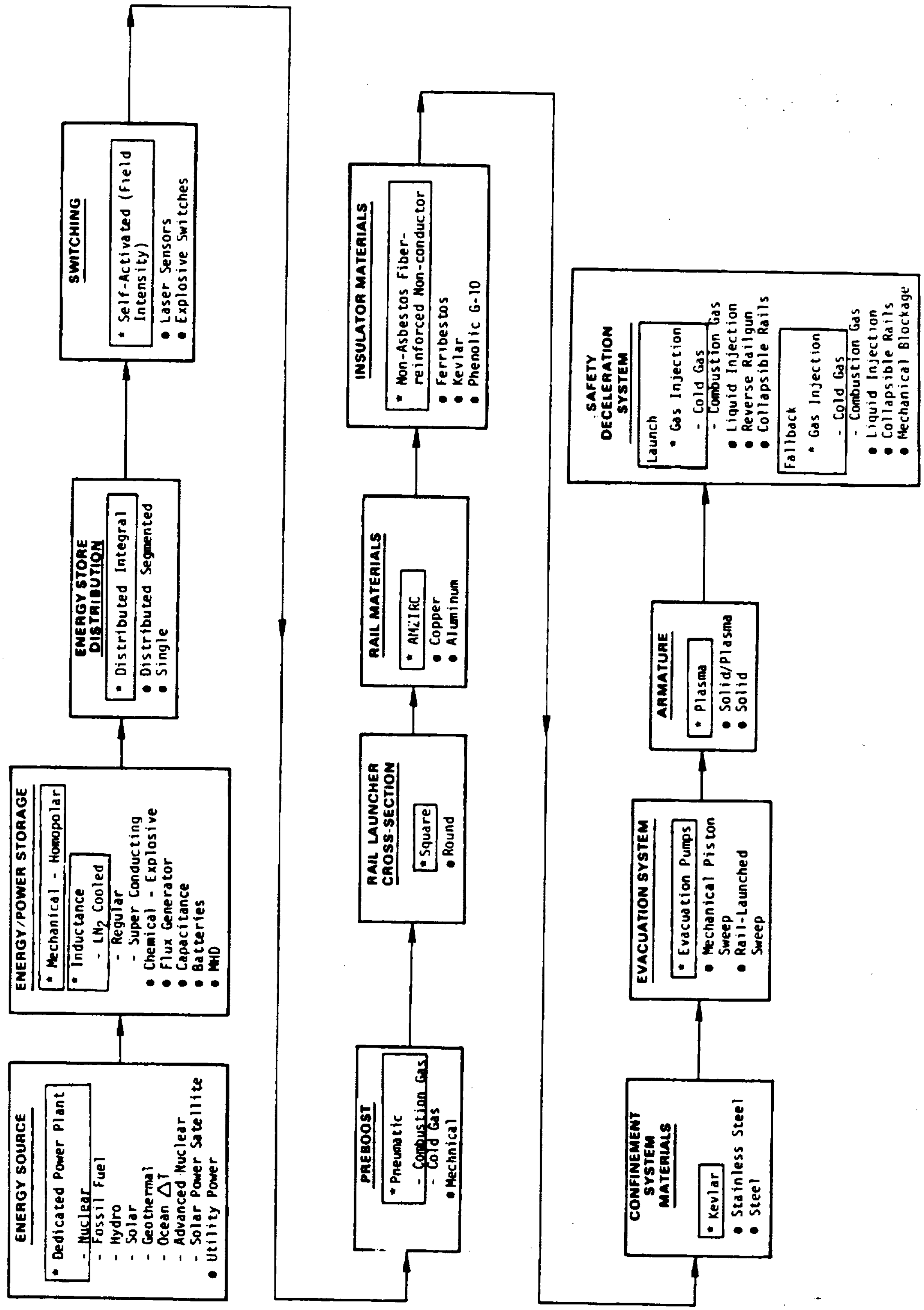


FIGURE 3-4. ESRL SYSTEM OPTIONS

the standpoint of the launcher tube, it would be very difficult to conduct in terms of the energy store. The major subsections that follow this discussion (Sections 3.2, 3.3, and 3.4) describe how distributed integral energy stores, distributed segmented stores, and single stores drive the concept of an Earth-to-space rail launcher system. The three railgun experts were assigned individual analysis for one of these three types of energy distribution concepts.

3.1.1.3.4 Switching

Switching is required for both of the distributed integral and the distributed segmented energy store systems. It is believed that self-activated switching (by using field intensity) is the most effective way to switch in stores of energy into the rails. Other possibilities include lasers/sensors and explosive switches. Switching is vital to the development of an ESRL system which is based upon distributed energy stores. Switching is discussed further in Section 3.2.4.3.

3.1.1.3.5 Preboost

Various concepts for preboost were identified and/or briefly investigated. Possibilities included a mechanical system for accelerating the payload/projectile. Also various pneumatic systems were considered. It was believed that a pneumatic system would be more effective in the preboost phase and provide for longer subsystem lifetime. Preboost is required to boost the payload to a velocity of approximately 1,000 m/s prior to entering into the rail acceleration phase. Dr. Richard Marshall, University of Texas, had recommended that to reduce erosion of the rail surface during the initial acceleration phase, a velocty roughly 1,000 m/s is necessary before rail acceleration occurs. Various concepts for gas acceleration were considered. The primary one suggested for the Reference Concept is based upon a concept of a light gas gun with a piston driver. The concept would involve the high pressure combustion of liquid hydrogen/liquid oxygen to drive a piston, which would in turn drive a helium/nitrogen mixture. This high pressure mixture would force the projectile up and into the railgun section of the launcher tube. If properly designed, the preboost piston would never enter the railgun section. Another possibility would be to use a high pressure cold gas, but this was not considered to be technically viable. One last concept involved the use of a rocket propulsion system onboard the back end of the projectile. The problem with this concept would be that the rocket propulsion system would have enough velocity, that it would trail the projectile out the muzzle and could cause problems with the rail surfaces. Also, contamination by rocket effluent would possibly degrade the rail system. Therefore, the concept selected was the combustion driven light gas gun type preboost. Refer to Section 4.0 for a discussion of the preboost system.

3.1.1.3.6 Rail Launcher Cross-Section

Both round and square rail bore cross-sections for the rail launcher were considered. In the absence of experimental data, the square cross-section was selected for the Reference Concept.

3.1.1.3.7 Rail Launcher Materials

Material options identified for launcher tube construction are identified in this section. The materials choices are shown in Figure 3-4 under "Rail Materials", "Insulator Materials", and "Confinement System Materials". AMZIRC alloy (99.9 to 99.85 percent copper, remainder zirconium) was selected for the rails because of its higher strength (than pure copper) and its good electrical properties. A non-asbestos insulator was believed to be the most advantageous of the insulator materials. Kevlar was chosen for the confinement system material.

3.1.1.3.8 Evacuation System

Various concepts for evacuating the rail launcher tube were identified and are shown in Figure 3-4. The most effective way for keeping air out of the system is believed to be an evacuation pump-type system. Other novel ideas, including a mechanical piston which should be pulled through the launch tube to evacuate the air, were considered, as well as a rail-launched sweeper device which would sweep the air out, but this would likely create a problem in the coordination of the launch of the payload. A laser-breakable diaphram at the muzzle would be replaced after each launch.

3.1.1.3.9 Armature

Three possible armatures were identified for an Earth-to-Space Rail Launcher concept. They are: (1) plasma; (2) solid/plasma; and (3) solid. The plasma armature has been selected as a reference for the ESRL system concept. A solid/plasma armature may be, however, the best armature for this application (see discussion in Section 3.2.4.1.1). Significant additional technology work would have to be performed to verify the performance of the solid/plasma armature. Significant work has already been conducted on the plasma and solid armatures for railguns.

3.1.1.3.10 Safety Deceleration System

To protect the rail launcher system from destruction that could occur during a misfire, a cold gas injection system was identified as a system which, when properly employed, would prevent the projectile fallback and destruction in the launcher tube. The cold gas system is believed to be the most safe and effective method for decelerating a payload which is falling back on its breech after a misfire. Proper design is necessary to assure that as the pressure would buildup in the base of the rail launcher breech, it would not lead to a pressure rupture of the system. A safety deceleration system for a launch of a payload when a misfire occurs is desirable to slow down the projectile in the rail launcher tube such that it will not leave the tube. There is a possibility that this could not be accomplished if the misfire would occur beyond the critical point in the launch. It is believed that the cold gas injection system is the most practical, probably employing nitrogen gas. A liquid injection system could damage the rails, as well as the projectile. Also, there is a possibility of collapsing the rails near the end of the muzzle to mechanically decelerate the payload. This would most likely result in payload breech and, for nuclear waste missions, this is not desirable.

3.1.1.3.11 Rail Maintenance Options

Sections 4.0 and 6.0 of this report discusses in more detail the service tunnels, service elevators, and items of this type that would be needed to support the maintenance of a rail laucher tube. One novel idea for maintaining rail tolerances in the launch tube would be to have a mechanical milling device which could be pulled up through the launcher bore, to actually mill the AMZIRC copper rail material away to the desired tolerance. This could be conducted when necessary. The rail thickness would have to be designed to accommodate the desired lifetime of the system.

3.1.2 Radiation Shielding Analysis

Early in the study, nuclear waste shielding calculations were performed using the following assumptions: (1) commercial high-level waste, as defined in the 1980 study by Rice, et al (Modified PW-4b waste mix in cermet form, having 90 percent of the Cs and Sr removed--waste assumed to be 10 years out of the reactor); (2) QAD Shielding Analysis Code was employed; (3) radiation was to be limited to 10 rem/hr at 1 meter (sideways) from the cylindrically-shaped waste form; and (4) low-cost steel was assumed for the shield. The radiation limit of 10 rem/hr at 1 meter was recommended based upon the following logic.

- "Standard" space disposal radiation shielding limit is 1 rem/hr at 1 meter

- Other limits are also given as 1 rem/hr or 1000 mrem/hr at 1 meter (see Section 2.0).

- Because of the remoteness of the launch activity, the vertical launch, and the requirement for a high degree of strength in the shield (in the event of accidents, etc.) it was believed that perhaps 10 rem/hr might be acceptable to the international community.

The nuclear waste shielding calculation utilized the source term representing 10-year-old PW-4b waste with 90 percent of the cesium and strontium removed. An ORIGEN calculation for this composition gave the source term below:

Source Term*

Photon Energy (Mev)	Photons/s-MTHM
0.30	3.19×10^{13}
0.63	5.42×10^{14}
1.10	1.28×10^{14}
1.55	1.08×10^{12}
1.99	3.50×10^{9}
2.38	4.89×10^{7}
2.75	2.28×10^{7}
3.25	1.42×10^{7}
3.70	9.09×10^{6}
4.22	5.74×10^{6}
4.70	2.71×10^{6}
5.25	1.71×10^{6}

*Note: Spontaneous fission neutrons and alpha-n neutrons per MTHM = 4.30×10^{8}/s.

The geometry for the shielding calculation is as shown below. For each waste form diameter used, a number of QAD computer calculations was performed for various ℓ/d ratios and for various shielding thicknesses. For each calculation, the dose rate was calculated at a number of detector points (see figure below) in the shield and in the air outside the shield. The results of these calculations were plotted as rem/hr at 1 meter versus shielding thickness. From these plots the desired shielding thicknesses were taken.

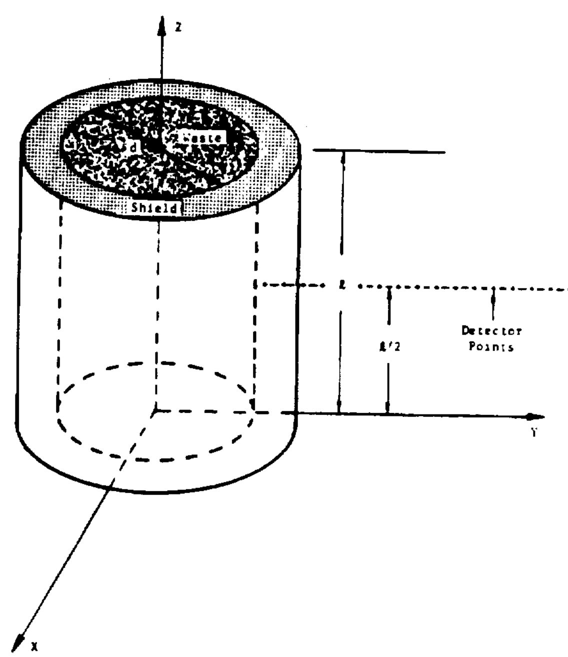


Figure 3-5 presents the results of the shielding calculations and Table 3-1 provides some overall parametric nuclear waste payload size and mass characteristics. It should be noted in the table that the larger the waste form diameter, the more efficient is the payload mass delivered per mission. During the course of the study it was suggested--"why not increase the allowable radiation dose to 100 rem/hr at 1 meter and significantly improve the payload mass." The bottom half of Table 3-1 shows the results of this calculation. Basically, the allowable waste form payload increases by about a factor of two (2) for a 100 rem/hr at 1 meter.

As this study progressed, and results were in on a parallel study effort (see Rice et al, 1982), it was decided to assume the Reference waste mix for standard space disposal. This mix was similar to the Modified PW-4b mix (Rice et al, 1980), but exhibited much lower thermal and radiation outputs. This "new" waste mix for space disposal assumed 95 percent Cs and Sr removal and a much longer storage time, of the order of 30 to 50 years out of the reactor. For this mix and for a 25 cm diameter waste form with a length to diameter ratio of about 5, a steel radiation shield thickness of about 12 cm for 10 rem/hr at 1 meter was deduced from the working level dose curves. This value was used for the reference ESRL case (Mission A). Actually a 11.5 cm shield coupled with an 0.5 cm primary steel container was assumed for the Reference Concept.

3.1.3 Launch Velocity Requirements

This section discusses the launch velocity requirements for both the solar system escape mission and the Earth orbital missions.

3.1.3.1 Solar System Escape Mission

Discussion in this section pertains to the development of the velocity requirements for solar system escape missions (e.g., nuclear waste disposal in space).

For motion under the influence of a single attracting body, a simple relationship exists between the speed of the projectile and the radial distance to the center of attraction. This equation, an energy conservation relationship, may be written as follows for escape trajectories:

$$v^2 - \frac{2\mu}{r} = v_\infty^2 \qquad (1)$$

In this equation, v is the velocity magnitude of the projectile at any radial distance, r; μ is the gravitational constant for the attracting body; and v_∞ is the hyperbolic excess velocity which occurs as r increases without limit.

Note that the minimum velocity needed to escape can be computed by setting the hyperbolic excess velocity equal to zero. Then,

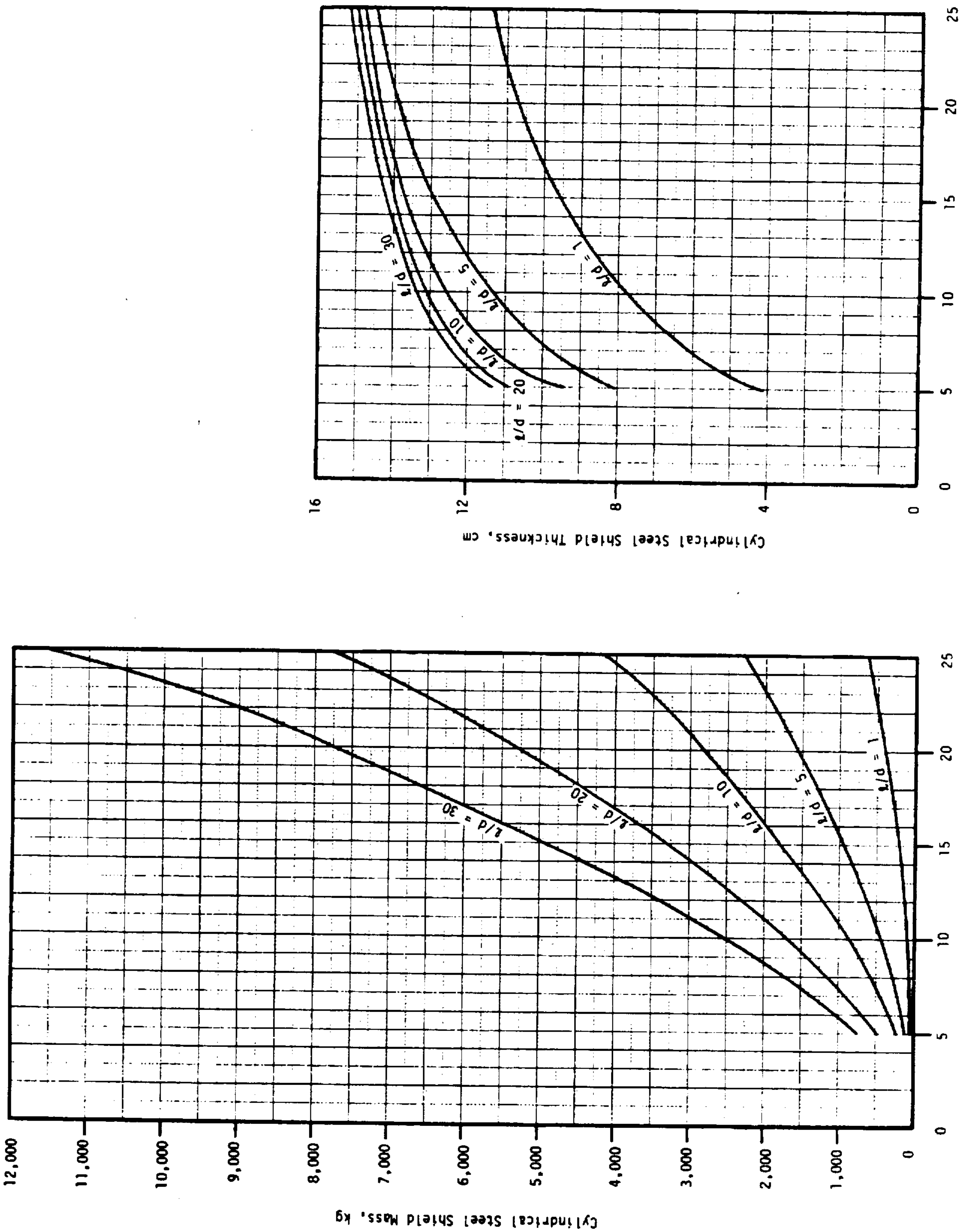


FIGURE 3-5. RESULTS OF PARAMETRIC SHIELDING ANALYSIS

TABLE 3-1. SHIELDED PAYLOAD CHARACTERISTICS FOR 90 PERCENT CS AND SR REMOVAL--10 YEAR OLD HL WASTE

Waste Form				Shield		Payload			
Diameter (cm)	Length (cm)	L/D	Mass (kg)	Thickness (cm)	Mass (kg)	Diameter (cm)	Length (cm)	Mass (kg)	Payload/ Shield Mass Ratio
10 Rem/Hr at 1 Meter									
5	25	5	3.2	8.2	112	21.4	41.4	115	0.029
5	50	10	6.4	9.5	236	24.0	69.0	242	0.027
5	100	20	12.8	10.9	521	26.8	121.8	534	0.025
5	150	30	19.1	11.4	795	27.8	172.8	814	0.024
10	50	5	25.5	11.4	449	32.8	72.8	474	0.057
10	100	10	51.0	12.6	889	35.2	125.2	940	0.058
10	200	20	102.0	13.2	1,715	36.4	226.4	1,817	0.059
10	300	30	153.0	13.4	2,527	36.8	326.8	2,680	0.060
20	100	5	205.0	13.9	1,544	47.8	127.8	1,749	0.133
20	200	10	408.0	14.4	2,848	48.8	228.8	3,256	0.143
20	400	20	817.0	14.5	5,330	49.0	429.0	6,147	0.153
20	600	30	1,225.0	14.6	7,860	49.2	629.2	9,085	0.156
25	125	5	399.0	14.5	2,272	54.0	154.0	2,671	0.177
25	250	10	798.0	14.7	4,108	54.4	279.4	4,706	0.194
25	500	20	1,595.0	14.9	7,832	54.8	529.8	9,427	0.204
25	750	30	2,393.0	15.1	11,692	55.2	780.2	14,085	0.205
100 Rem/Hr at 1 Meter									
10	50	5	25.5	5.8	144	21.6	61.6	169	0.177
10	100	10	51.0	7.5	377	25.0	115.0	428	0.135
20	100	5	205.0	8.7	756	37.4	117.4	961	0.271
20	200	10	408.0	9.2	1,473	38.4	218.4	1,881	0.277
25	125	5	399.0	9.0	1,141	43.0	143.0	1,540	0.350
25	250	10	798.0	9.4	2,202	43.8	268.8	3,000	0.362

$$v = \sqrt{\frac{2\mu}{r}} = v_{escape} \qquad (2)$$

Substituting Equation (2) into Equation (1),

$$v = \sqrt{v^2_{escape} + v^2_\infty} \qquad (3)$$

Equation 3 can be used to compute the launch velocity required to escape the Earth (neglecting atmospheric drag) and retain any given value of v as the distance from Earth approaches infinity. For this application, the escape velocity at the Earth's surface, 11.19 km/s, can be used.

Similarly, Equation 2 can be used to compute the velocity needed to escape from the Solar System, assuming that the Sun is the only significant attracting body. Using the radial distance from the Sun equal to the nominal radius of the Earth's orbit, and the appropriate gravitational constant, the heliocentric escape speed is about 42.14 km/s.

Since the speed of the Earth in orbit around the Sun is about 29.8 km/s, the value of v after Earth escape [from Equation (1)] must equal the difference between 42.14 and 29.8 km/s, assuming the v vector is aligned perfectly with the Earth's orbital motion. Consequently, the minimum v is 12.34 km/s, and the corresponding ideal velocity at the Earth's surface (from Equation 3) is 16.66 km/s. It should be noted, however, that the ideal launch velocity is slightly less because some benefit is obtained from the rotational speed of the Earth about its axis.

The vis-viva energy equation, written with respect to the Sun is,

$$v_s^{\,2} - \underbrace{2\,(29.80)^2}_{} = v_{\infty_s}^{\,2} \qquad (4)$$

Heliocentric speed after escaping Earth

Square of solar escape speed at a distance from Sun equal to radius of Earth's orbit (29.80 km/s is Earth's orbital speed)

Speed after leaving Solar system

Now, if the hyperbolic excess velocity after escaping Earth is aligned with the velocity of the Earth around the Sun,

$$v_s = 29.80 + v_{\infty_e} \qquad (5)$$

or

$$v_{\infty_e} = v_s - 29.80 \qquad (6)$$

Substituting (4) into (6),

$$v_{\infty_e} = \sqrt{v_{\infty_s}^{\,2} + 2\,(29.80)^2} - 29.80 \qquad (7)$$

But, from the vis-viva equation written with respect to Earth,

$$v_{\infty_e} = \sqrt{v^2 - (11.19)^2} \tag{8}$$

Total velocity at Earth surface

Earth escape velocity at Earth surface (vector addition of launch velocity and rotational velocity)

So, equating (7) and (8),

$$\sqrt{\bar{v}^2 - (11.19)^2} = \sqrt{v_{\infty_s}^{\,2} + 2\,(29.80)^2} - 29.80 \tag{9}$$

Solving for v,

$$v = \sqrt{\left[\sqrt{v_{\infty_s}^{\,2} + 2\,(29.80)^2} - 29.80\right]^2 + (11.19)^2} \tag{10}$$

Excess speed after Solar system

Earth's orbital speed

Earth's escape speed at Earth's surface

For velocities less than that needed to escape the solar system, the relationship between orbital period and the launch velocity was developed from basic orbital mechanics relationships. Figures 3-6 and 3-7 provide information on time to encounter, either the Earth, if less than ideal minimum, or nearest star, if more than minimum.

3.1.3.2 Earth Orbit Applications Missions

The following discussion presents the development of the launch velocity requirements for ESRL Earth orbital missions. Because the trajectory resulting from the ESRL launch is ballistic, it is necessary to carry additional propulsion to give the projectile a velocity increment, Δv, necessary to place it in an Earth orbit. The diagram shown below is a schematic indicating the velocities and angles of interest here. Definitions of the symbols are given below:

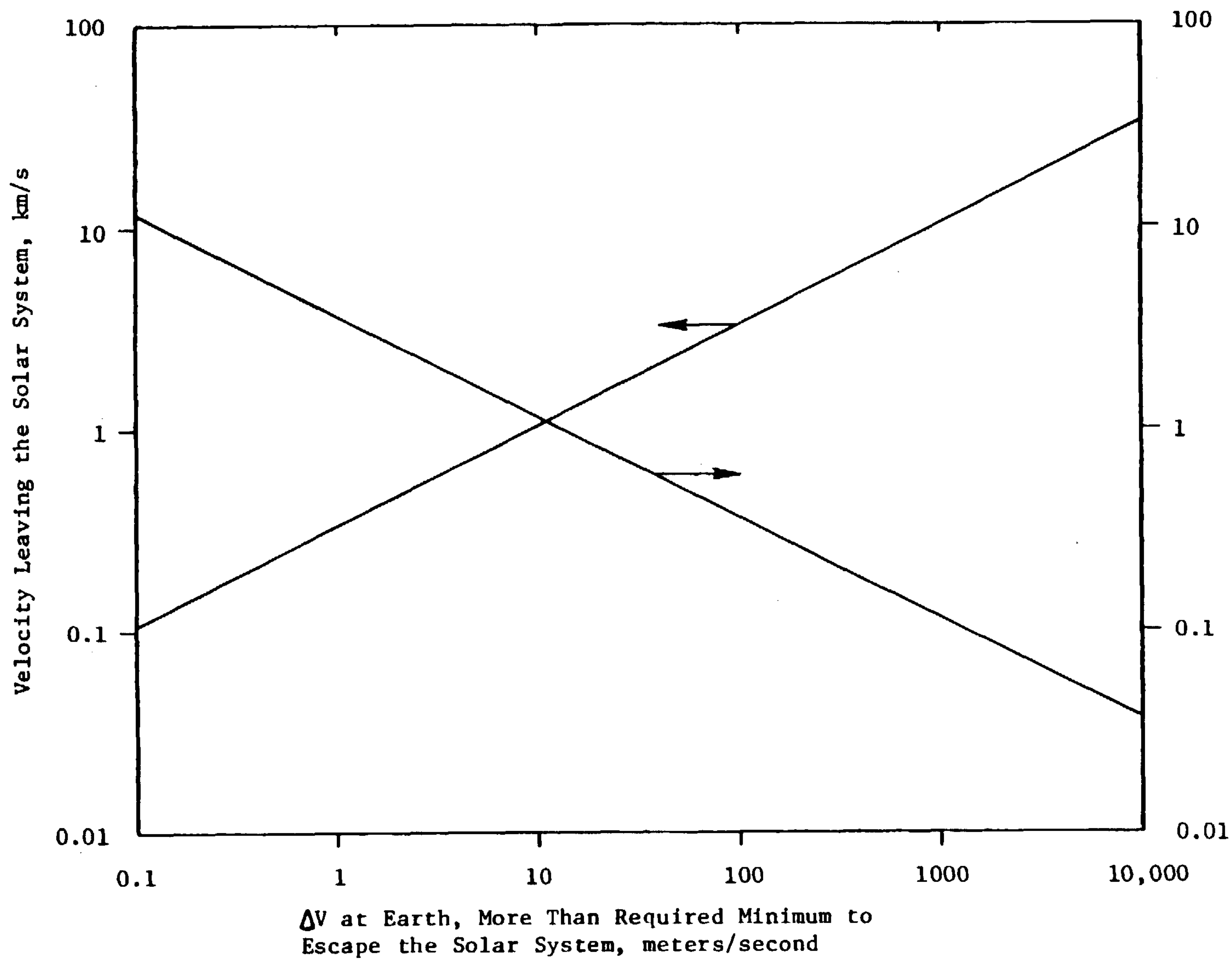


FIGURE 3-6. SOLAR SYSTEM DEPARTURE SPEED AND TIME TO NEAREST STAR AS A FUNCTION OF IDEAL EXCESS EARTH SURFACE VELOCITY

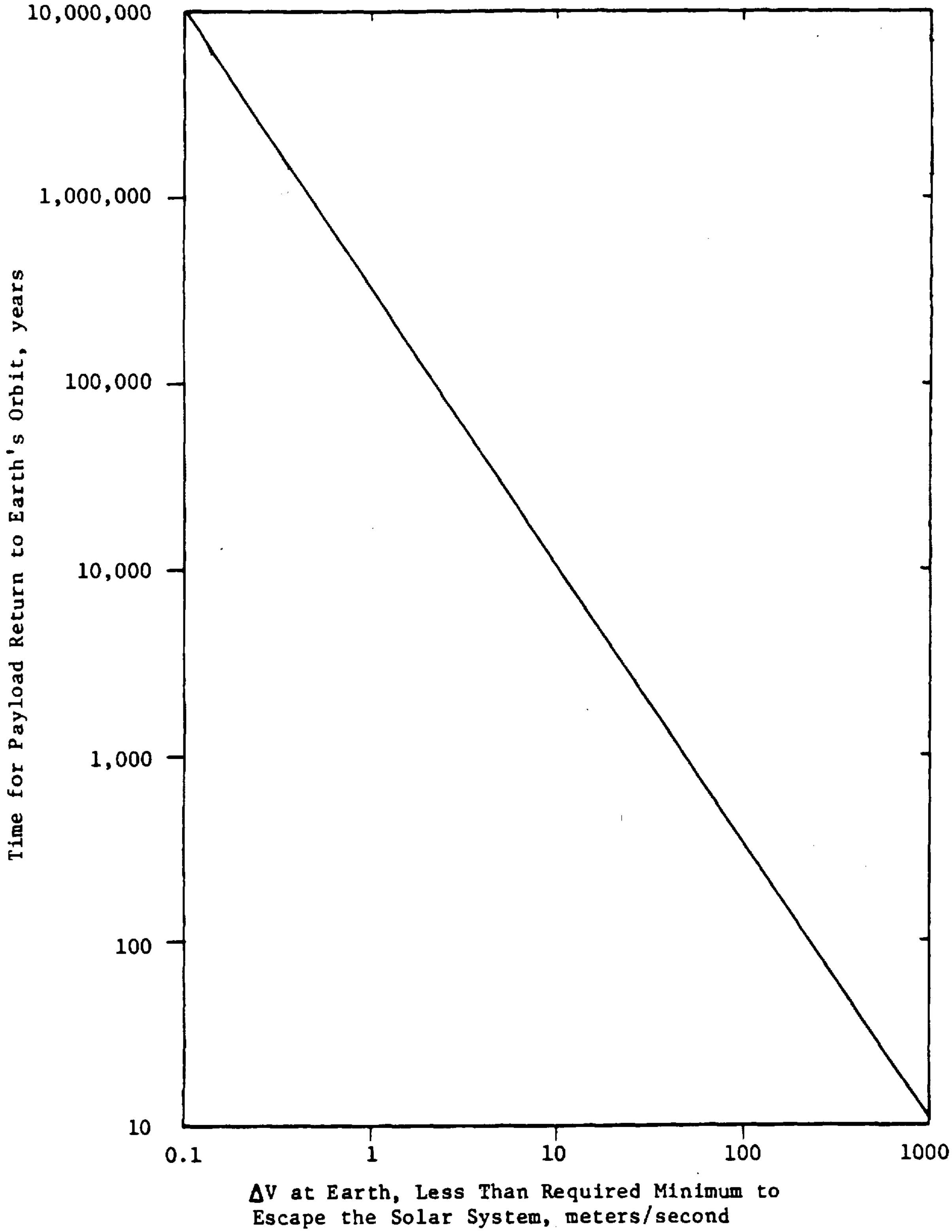


FIGURE 3-7. **TIME TO EARTH ORBIT REENCOUNTER AS A FUNCTION OF IDEAL EARTH SURFACE VELOCITY (LESS THAN MINIMUM)**

v_L = launch velocity

v_o = orbital velocity

v = velocity of ballistic velocity at orbital altitude

Δv = velocity increment necessary to place the projectile in orbit

θ_L = launch angle, measured from the horizontal

θ = angle of trajectory at orbital altitudes measured from the local horizontal

The energy of a given orbit is constant such that the energy at launch is equal to the energy at a particular altitude of the same orbit. The law of conservation of energy (neglecting drag) is written as follows:

$$E = \frac{v^2}{2} - \frac{\mu}{r} \tag{11}$$

where E is the energy of the orbit, v is the velocity, μ is the gravitational constant for Earth, and r is the radial distance measured from the center of the Earth. For a given velocity at a particular altitude, the corresponding launch velocity at the surface can be determined from Equation 11.

Angular momentum must also be conserved and is expressed as,

$$H = rv\cos\theta \tag{12}$$

In this equation, H is the angular momentum of the orbit and θ is the angle of trajectory measured from the local horizontal. Using the values of velocity at a given altitude and at the surface from Equation 11, the launch angle, θ_L, can be determined from Equation 12 for different θ's at altitude.

The previously obtained launch parameters do not account for the contribution to the velocity of a rotating Earth or for the effects of drag, and they must be corrected.

For an eastward launch, the launch velocity required is actually less than the total velocity calculated above. The rotational velocity of Earth (0.465 km/s at the equator) must then be subtracted from the previously-calculated horizontal component of launch velocity. This subtraction results in lower launch velocities, but increases the angle of launch.

The final correction to the velocity is to compensate for velocity losses due to atmospheric drag. From Section 3.1.4.3, the ratio of initial launch velocity to final velocity upon leaving the atmosphere for the Earth orbital missions is:

$$\frac{v_L}{v_f} = e^{\frac{0.0547}{\sin\theta_L}} \qquad (13)$$

From the same section, the exponent of the exponential function ($0.0547/\sin\theta_L$) is determined for the reference Earth orbital projectile ($C_D = 0.1$) and assumes a sea-level launch.

Figure 3-8 shows the launch velocities, v_L, and angles, θ_L, calculated as a function of different trajectory angles, θ, at an altitude of 500 km above Earth. For the design configuration, having a 20 degrees launch angle, the corresponding launch velocity is about 6.85 km/s for a trajectory angle of 0 degrees at 500 km.

The additional propulsion system required for orbit insertion is sized by the velocity increment necessary to match the desired orbital velocity at the given altitude ($v_o = 7.61$ km/s for a 500 km circular orbit).

The law of cosines defines the necessary Δv:

$$(\Delta v)^2 = v^2 + v_o^2 - 2vv_o\cos\theta \qquad (14)$$

Figure 3-9 illustrates the above relationship between trajectory velocity, v, angle of flight, θ, and the required velocity increment, Δv, for a circular 500 km orbit ($v_o = 7.61$ km/s).

From Figure 3-8, the launch velocity was found to be 6.85 km/s for a launch angle of 20 degrees and a trajectory angle of 0 degrees at the orbital altitude (500 km). Following the described procedure, the velocity of the projectile trajectory at a given altitude may be found. The launch conditions correspond to a velocity of 5.51 km/s at 500 km altitude. By substituting this value into Figure 3-9, the velocity increment, Δv, necessary to circularize into a 500 km orbit is 2.1 km/s for the design configuration (launch angle fixed at 20 degrees from the horizontal).

3.1.4 Launch Window Analysis for Space Disposal

3.1.4.1 Computational Approach

The necessary and sufficient condition for a projectile to escape the solar system on an unpowered trajectory can be simply stated. At a distance from the Sun approximately equal to the radius of the Earth's orbit, and at a distance from the Earth great enough that the Earth's gravitational attraction is negligible compared to the Sun, the projectile must have a speed of about 42.12 km/s with respect to the Sun (heliocentric speed).

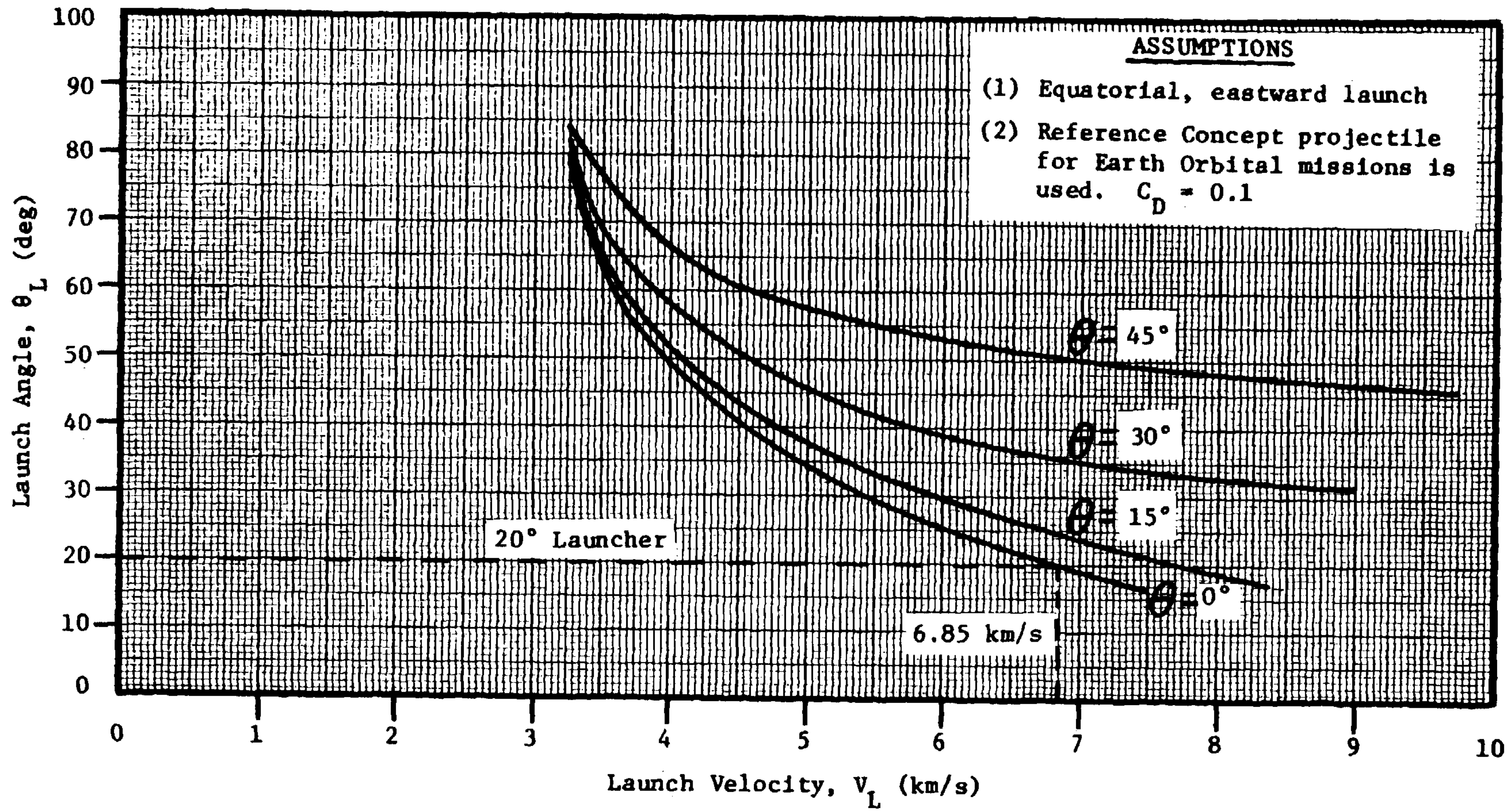


FIGURE 3-8. LAUNCH VELOCITY AND ANGLE REQUIREMENTS FOR 500 KM CIRCULAR ORBIT

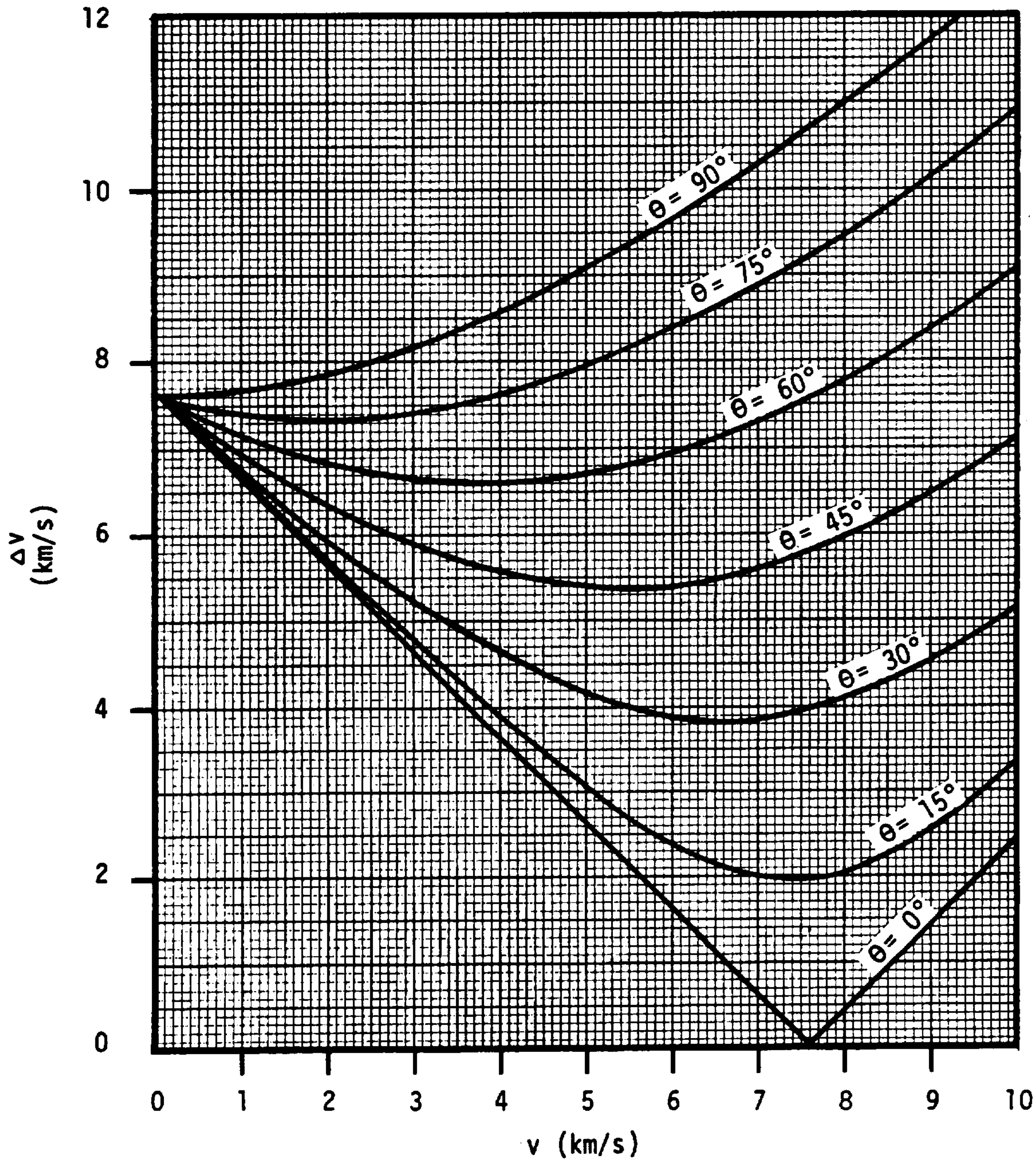


FIGURE 3-9. ΔV REQUIREMENTS FOR 500 KM CIRCULAR ORBIT

Conceptually, the escape can be considered in two steps (the patched-conic technique). First, the projectile is launched from the Earth's surface with sufficient velocity to follow a hyperbolic Earth-escape trajectory. During this phase, the Earth and the projectile, since they are in relatively close proximity, are being accelerated nearly equally toward the Sun by the Sun's gravitational field. Consequently, the Sun's effect on the projectiles trajectory, relative to Earth, is insignificant.

As the projectile approaches the asymptote of the Earth-escape hyperbolic, its speed, relative to Earth, approaches a particular value (the hyperbolic excess velocity) which is a function of the initial launch velocity velocity at the surface. The hyperbolic excess velocity vector can be added vectorially to the velocity of the Earth around the Sun to compute the scalar heliocentric speed, which may then be compared to the 42.12 km/s solar escape requirement. If the 42.12 km/s criterion is exceeded, the projectile will then follow a Sun-centered hyperbolic path out of the solar system; if not, the projectile will enter a closed elliptic orbit about the Sun.

The patched-conic concept, just described, was used for all ESRL launch window computations described herein, obviating the need for detailed and time-consuming integration of the projectile equations of motion. The bulk of the computational effort is then reduced to solving the complex geometrical relationships between launch site latitude, time of day, time of year, launch velocity, and launch direction.

3.1.4.2 Vertical Launches

A parametric study of vertical impulsive launch requirements for solar system escape was completed under the following assumptions:

(1) Launch occurs vertically and impulsively at sea-level
(2) No atmospheric drag loss is considered
(3) The Earth is round and rotating
(4) The Earth's orbit around the Sun is circular.

Using the patched-conic technique, the hyperbolic escape trajectory relative to Earth was computed, including the eastward velocity component caused by Earth rotation. From this computation, the hyperbolic excess velocity, at a great distance from the Earth, was determined, as well as the direction of the velocity vector. The hyperbolic excess velocity was then added vectorially to the heliocentric velocity of the Earth and the total heliocentric velocity of the projectile was then compared to the heliocentric escape velocity (42.12 km/s at a solar distance equal to the radius of the Earth's orbit). The parametric effects of launch latitude, time of day, and time of year were then examined.

In the time period near 6 a.m., the radial vector outward from the center of the Earth is aligned most closely to the direction of the Earth's motion around the Sun. Furthermore, for high-energy launches, the escape trajectory relative to Earth is curved eastward (because of Earth rotation) only slightly. As a consequence, the optimum launch time for any latitude or launch date is about 6 a.m.

To illustrate, Figure 3-10 shows the final heliocentric velocity as a function of time for equatorial launches. Two times of year are shown in the figure (corresponding to the worst and best launch dates, as will be discussed below). For an optimal launch date, the absolute minimum launch velocity is about 16.63 km/s at about 6 a.m. If higher launch velocities are available, a period of several hours would be suitable for launches. For instance, for an 18 km/s launch capability, the launch window would vary from about 4.7 hours on the optimum launch date down to about 3.6 hours on a worst launch date.

Figure 3-11 illustrates the effects of launch latitude and date for 6 a.m. launches at a fixed launch velocity of 18 km/s. It is interesting to note that for launch latitude less than 23.45 degrees (the inclination of the equatorial plane to the ecliptic plane) there are two optimum launch periods each year. For equatorial launches, the optima occur at about 90 days and 270 days after vernal equinox. As the launch latitude progresses northward from the equator, the two optima approach each other and coalesce into one optimum (at the autumnal equinox) for a north latitude of 23.45 degrees. For launch latitudes greater than 23.45 degrees, the optimum time of year is the autumnal equinox. It may also be observed in Figure 3-11 that the the maximum final heliocentric velocity is independent of latitude in the range from zero to 23.45 degrees.

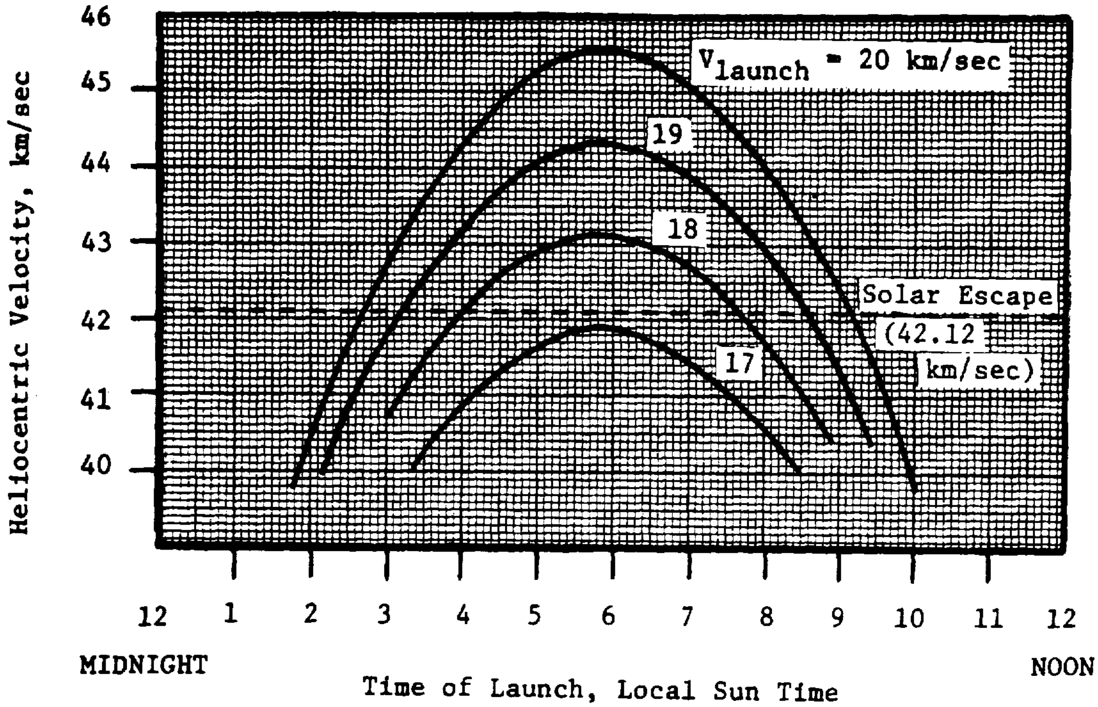
The minimum launch velocity for solar system escape is shown in Figure 3-12 as a function of launch latitude and date of launch. All launches were assumed to occur at 6 a.m.

Figures 3-11 and 3-12 illustrate that the advantage of an equatorial launch site is not that it reduces the minimum launch velocity, but that it reduces the penalty that must be paid for launching at non-optimal times of the year.

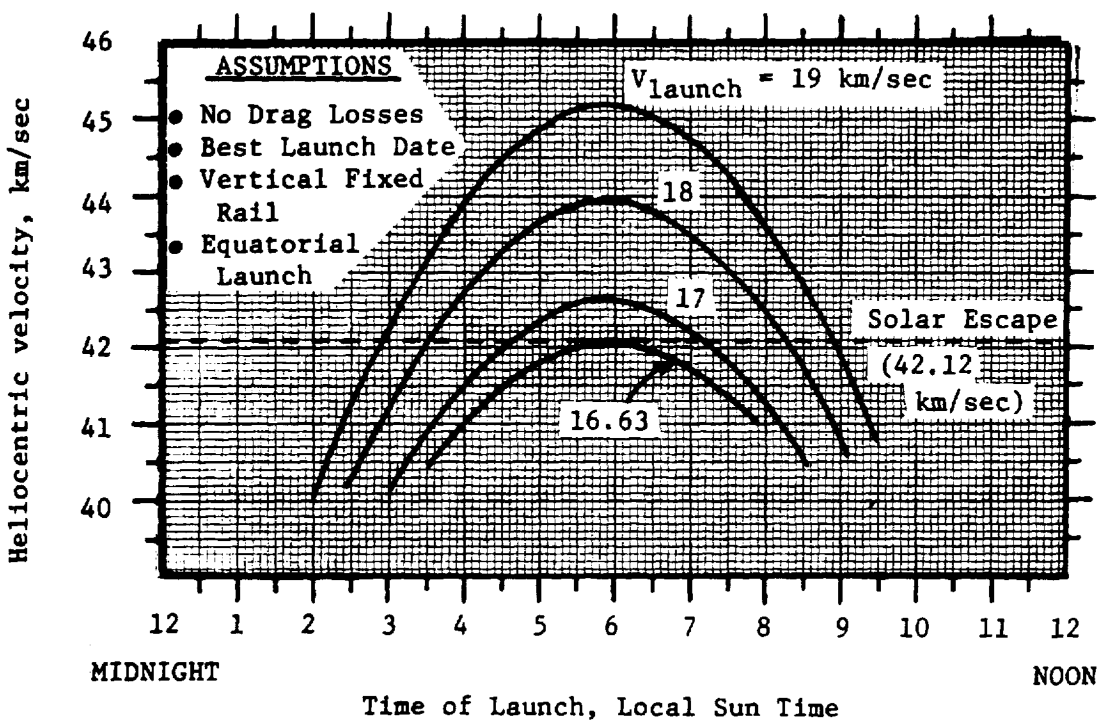
3.1.4.3 Effects of Non-Vertical Launches and Atmospheric Drag

Atmospheric drag losses were computed on the assumptions that the path is a straight line during atmospheric passage and that the atmospheric density is an exponential function of altitude. Under these conditions, the ratio of initial velocity to final velocity becomes:

$$\frac{V_o}{V_f} = \exp\left[\frac{g}{2\cos\alpha}\left(\frac{C_D A}{W}\right)\frac{\rho_o}{\beta}\right]$$



a. Equatorial Launch at Vernal Equinox (a Worst Date)



b. Equatorial Launch 91 Days After Vernal Equinox (a Best Date)

FIGURE 3-10. LAUNCH WINDOWS FOR ESRL SOLAR SYSTEM ESCAPE USING VERTICAL, FIXED RAIL WITH NO DRAG

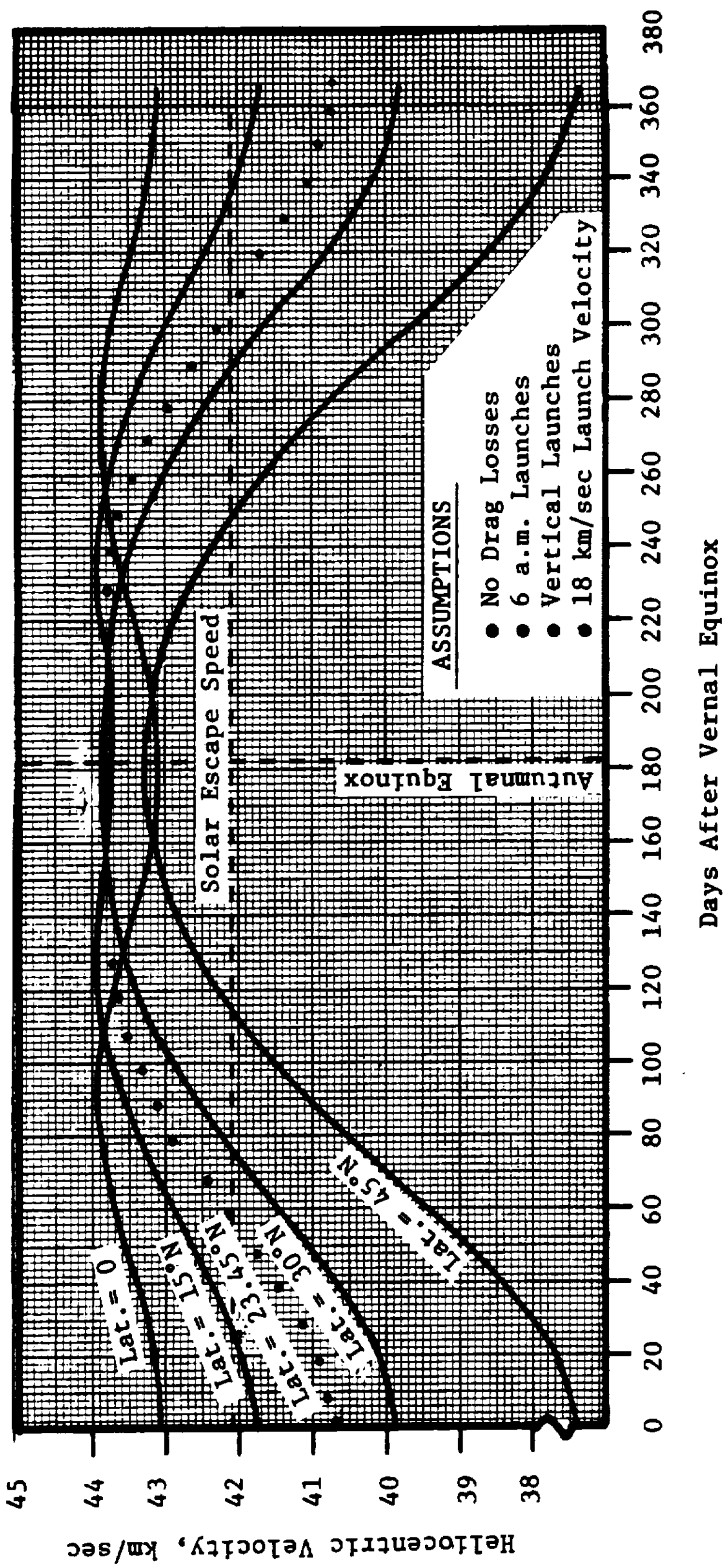


FIGURE 3-11. ATTAINABLE HELIOCENTRIC VELOCITIES AS A FUNCTION OF LATITUDE AND TIME OF YEAR FOR ESRL SOLAR SYSTEM ESCAPE USING VERTICAL, FIXED RAIL WITH NO DRAG

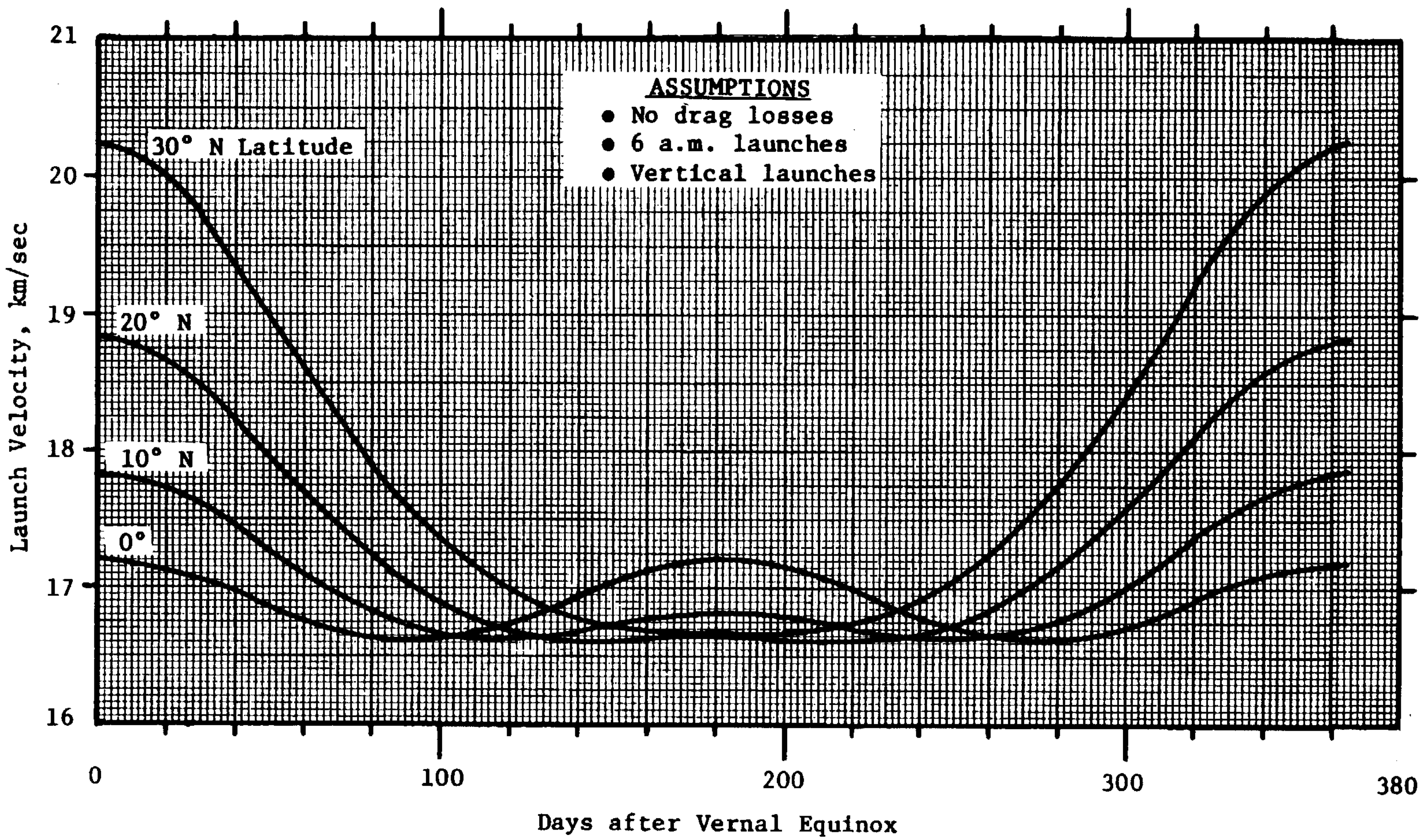


FIGURE 3-12. MINIMUM LAUNCH VELOCITY REQUIRED FOR SOLAR SYSTEM ESCAPE AS A FUNCTION OF LATITUDE AND TIME OF YEAR USING VERTICAL, FIXED RAIL WITH NO DRAG

where,

V_o = initial launch velocity
V_f = final velocity after leaving atmosphere
g = gravity constant
α = angle of launch, from vertical
C_D = drag coefficient
A = reference area of projectile
W = weight of projectile
ρ_o = atmospheric density at sea-level
β = exponential constant of density ($\approx$ .00003346 per foot).

The drag loss is seen to be a function of the flight path angle, measured from the vertical direction, and the ballistic coefficient which is the reciprocal of the quantity in parentheses.

For candidate projectile designs, the ballistic coefficient is quite large (of the order of 93,000 kg/m^2) which indicates a large ratio of inertia force to drag force. For this reason, the drag loss is only about 6 percent for vertical launches, increasing to about 10 percent for launches 50 degrees from the vertical.

Figure 3-13 shows the final heliocentric velocity as a function of time of day and flight path angle from the vertical on an optimum launch date and for a launch velocity of 19 km/s. Notice that a launch direction tilted 20 degrees to the east of vertical yields the highest heliocentric velocity. At this angle, the benefit of an eastward launch (to take advantage of the Earth's rotation rate) outweighs the increased drag loss. At greater deviations from the vertical, the drag loss penalty becomes more and more dominant.

On a best launch date, as represented in Figure 3-13, there is no advantage in launching in directions other than east or west to expand the daily launch window. It is seen that a window of 11 hours or so would be possible on the best date if the launch velocity were 19 km/s.

For other times of the year, the optimum launch direction each time of the day is a combination of the correct azimuth of launch (the angular displacement from north of the ground track), and the angle from the vertical. Figure 3-14 is similar to Figure 3-13 except that a worst-day launch is considered and the curve represents the envelope of optional combinations of azimuth and flight path angle. For example, on this launch date, the best direction of launch is 30 degrees from the vertical in a direction of 130 degrees from north, or roughly southeast.

It is apparent, from Figure 3-14, that the launch window on the worst date is approximately the same as that of the best date, if the launch rail can be pointed in the required directions.

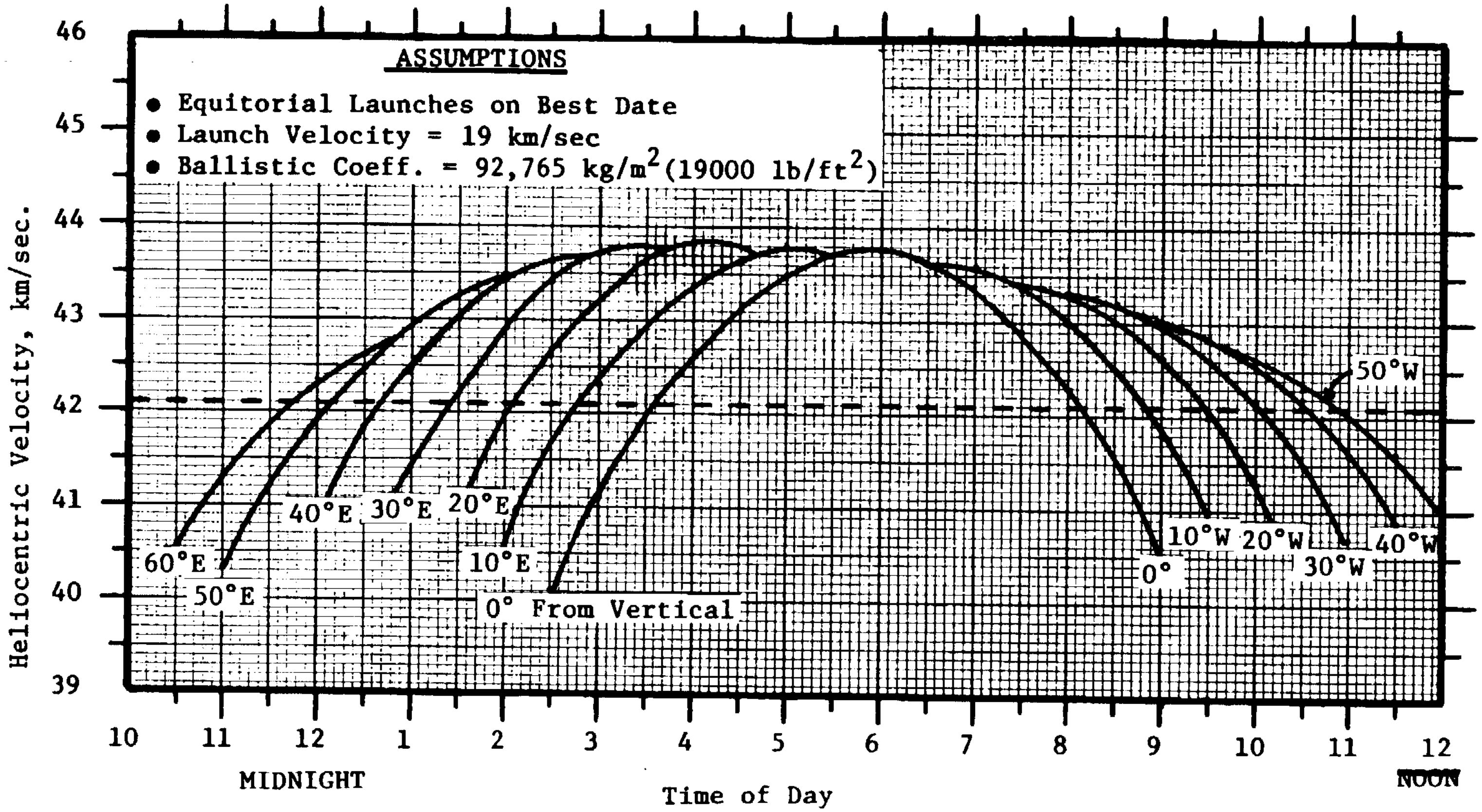


FIGURE 3-13. ESCAPE LAUNCH WINDOW VARIATION AS A FUNCTION OF ANGULAR DEVIATION FROM THE VERTICAL WITH NO ATMOSPHERIC DRAG

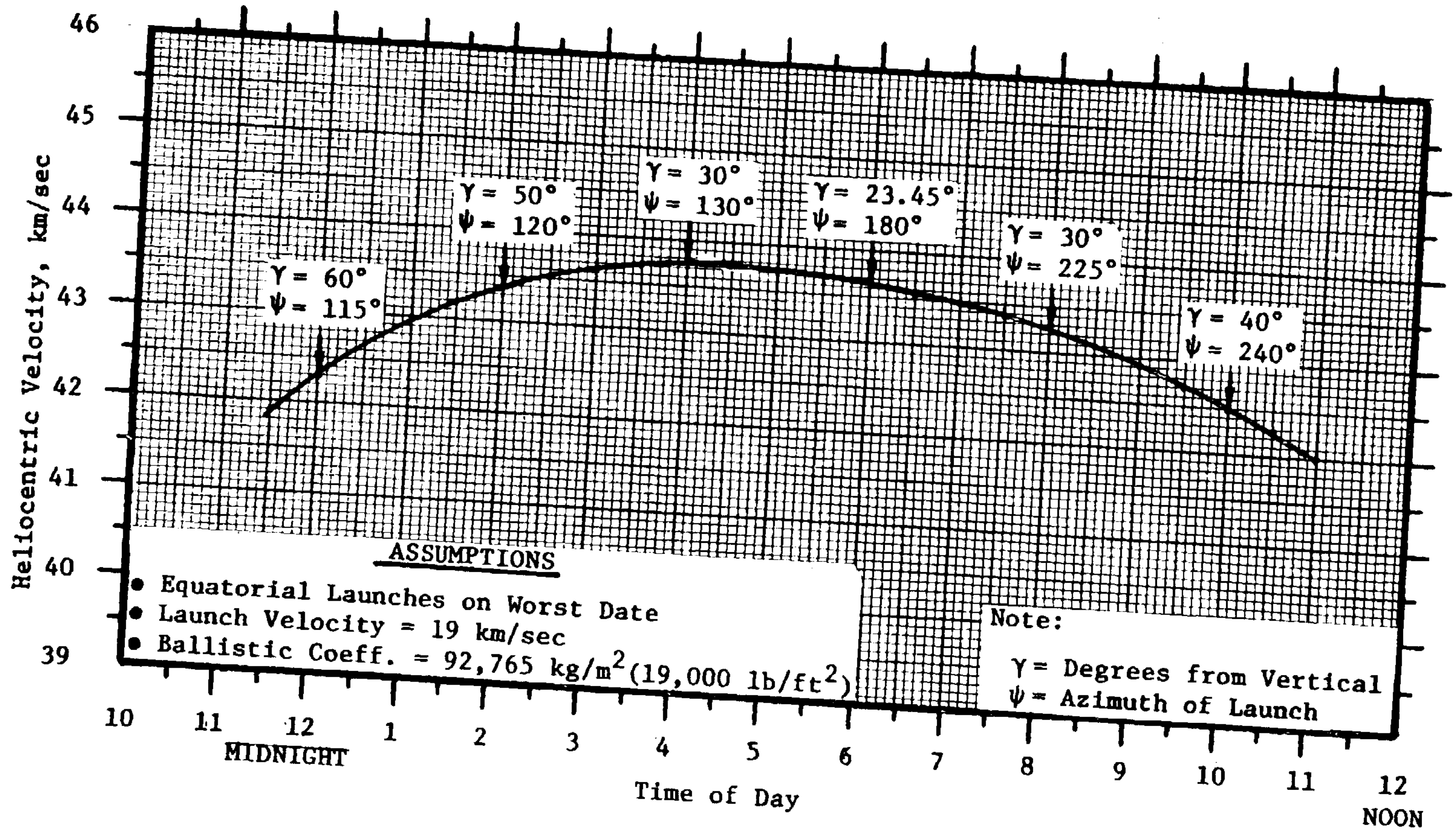


FIGURE 3-14. ESCAPE LAUNCH WINDOW ENVELOPE AS A FUNCTION OF ANGULAR DEVIATION FROM VERTICAL WITH ATMOSPHERIC DRAG (ON WORST DAY)

3.1.5 Projectile Concepts

During this study it was only possible to develop preliminary projectile concepts for the ESRL system. A considerable amount of analytical work remains in the areas of structural analysis, thermal analysis, and most importantly, aerodynamic analysis. Suggestions provided by: Dr. John Lee, Ohio State University; Dr. Al Buckingham, LLNL; and Mr. Hal Swift, of PAI Corporation, were used as a basis for the concepts discussed here. The projectiles for Missions A and B are discussed below.

3.1.5.1 Nuclear Waste Disposal in Space (Mission A) Projectile

The projectile for Mission A (see Section 4.3.1) was conceptualized to conform with the requirement that 0.50 MT of high-level waste (HLW) be disposed per day via solar system escape. For a reasonable sized launcher, with "achievable" rail stresses, one projectile could carry a waste form mass of about 250 kg (0.25 MT). Thus, there is a need for two waste launches every day. The basic requirements and desirable characteristics for the projectile were that:

(1) The waste form payload mass be 250 kg of HLW cermet

(2) The projectile diameter must be within the 67 cm bore limit

(3) The radiation shield surrounding the payload limit the radiation dose to 10 rem/hr at 1 meter. (Increasing the limit to 100 rem/hr at 1 meter would allow the payload to double)

(4) A launch sabot be used that is jettisoned in the atmosphere immediately after the projectile leaves the muzzler

(5) Fins for aerodynamic stabilization

(6) A high melting point, high heat of fusion nose metal be used

(7) The projectile be able to survive 10,000 g's during the rail launch phase

(8) The projectile expected to survive atmospheric flight and inadvertent reentry.

The reference waste form for space disposal is the Oak Ridge National Laboratory iron/nickel based cermet (Rice et al, 1982). A cermet is a dispersion of ceramic particles in a continuous metallic phase. The reference cermet is formed by a process involving dissolution and precipitation from molten urea followed by calcination and hydrogen reduction to produce a continuous metallic phase (Rice et al, 1980). Non-hydrogen reducible oxides would form the ceramic portion of the ceramic/metal matrix waste form. This waste form has been shown to have superior properties as compared to other potential waste forms for space disposal (Rice et al, 1980). The iron/nickel-based cermet has high waste loading (67.4 percent), a thermal conductivity 9.5 Watts/m-C), a high density (6.5 g/cc), and an excellent structural integrity characteristics (Rice et al, 1982). The waste form would be made in the form of a cylinder/cone 25 cm in diameter and 95 cm in length (see Figure 3-15). The form would have a mass of approximately 250 kg. During the formation process, the waste form would be pressed and formed in a 0.5 cm thick steel container with an enclosed end. After formation, an end cap would be electronic beam welded to the main container rim. This activity would be conducted in a hot cell.

The primary containment for the radioactive waste will be a 30 kg stainless steel cylindrical container, 0.5 cm thick. This container provides primary containment for the waste form during the various defined mechanical and thermal loads to which the total payload is subjected in anticipated normal and accident conditions. These loads would be mitigated in varying degrees by the waste form itself, by the cylindrical flight radiation shield (also the auxiliary radiation shield during storage or surface transport and ground handling), and by the shipping cask which provides additional protection for surface transportation. To protect structural integrity, the primary steel container should not exceed a temperature of 416 C during normal conditions (Rice, 1981).

The container would be housed in a steel flight radiation shield. The shield is intended to limit radiation to no more than 10 rem per hour at 1 meter from the shielding surface under normal conditions. The shield would be approximately 11.5 cm thick, conform to the container shape, and have a mass of about 1100 kg. Auxiliary shielding would be designed such that radiation exposure limits for ground personnel are not exceeded during operations (this would be 1 rem/hr at 1 meter). For normal conditions, the temperature limit for flight radiation shield is 416 C (Rice, 1981). During accident conditions, the shield should not exceed 1280 C (Rice, 1981).

The nose tip of the projectile would be slightly blunted and would be constructed of tungsten (see Figure 3-15). As the projectile traverses the atmosphere, the tungsten metal is expected to begin melting cleanly, leaving an eroded, but smooth nose surface. The body of the projectile is the radiation shield covered with about 1 cm of carbon/carbon material applied in such a way to provide strength and thermal protection. No detailed analysis has yet been conducted to verify survivability.

For stabilization during flight, four small stabilization fins would be attached to the rear of the projectile (see Figure 3-15). Also, at the rear of the projectile, a jettisonable, high-strength, ceramic non-conducting sabot would be used to: (1) protect the projectile and fins from excessive heating from contact with the driving plasma armature, and (2) proper positioning in the rail launcher tube. No aerodynamic analysis has yet been conducted to verify projectile stabilization. Dr. Lee has done a preliminary investigation to determine the stability (see Section 3-7).

A radio transmitter beacon would be located in the instrument package under the nose cone, along with an aerobraking decelerator system to be deployed automatically after the projectile leaves the atmosphere. This would

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OF POOR QUALITY

DIMENSIONS, cm		ESTIMATED MASS CHARACTERISTICS, kg	
PROJECTILE LENGTH	170	WASTE FORM	250
WASTE FORM LENGTH	95	SHIELD/CONTAINER	1140
WASTE FORM DIAMETER	25	NOSE CONE	440
SHIELD/CONTAINER THICKNESS	12	AFT SABOT	40
PROJECTILE DIAMETER	51	FORWARD SABOT	100
SABOT THICKNESS	22 - 8	TPS	25
OVERALL DIAMETER	67	INSTRUMENTS	50
		FINS	10
		TOTAL	2055

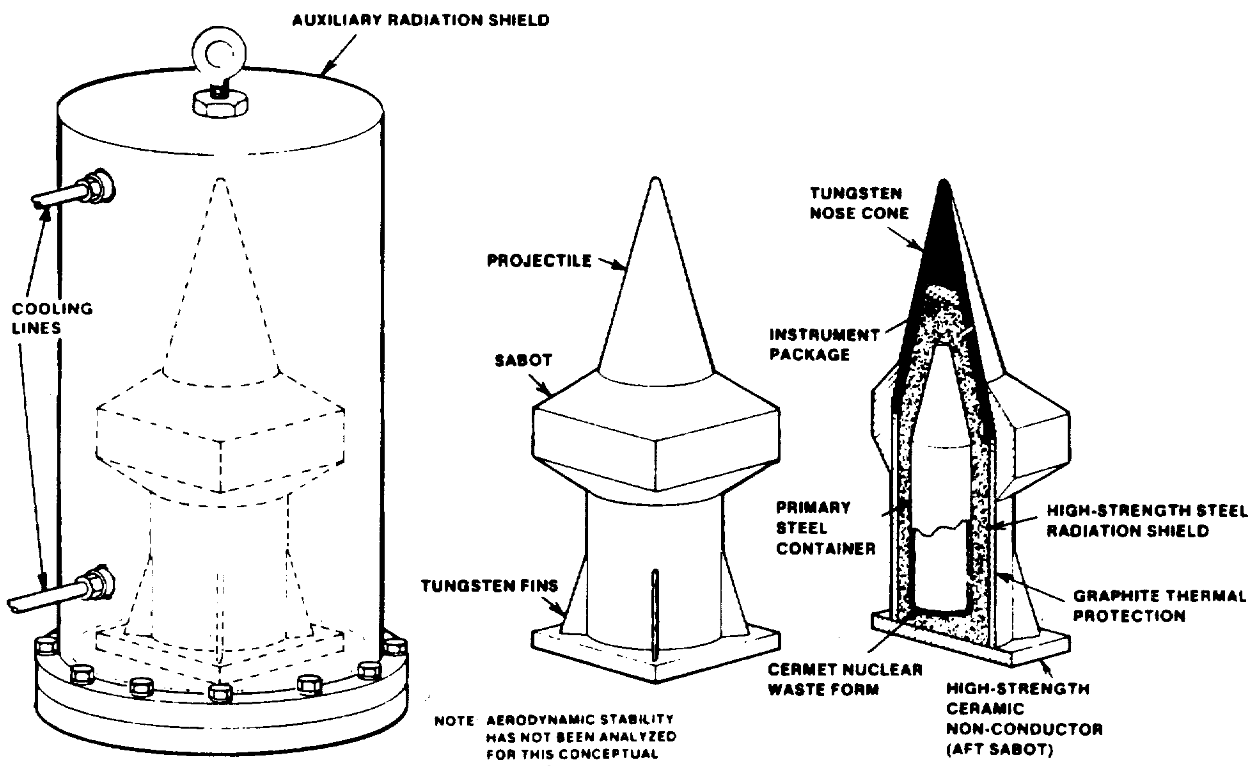


FIGURE 3-15. ESRL PROJECTILE CONCEPT FOR NUCLEAR WASTE DISPOSAL IN SPACE

allow a low velocity reentry if a misfire occurs, otherwise the payload will continue along its escape trajectory.

The assembled projectile, with fins, would be supported by a small sabot (forward and aft) for the acceleration portion of the launch. After the projectile leaves the ESRL, the sabot components would automatically be separated away in the initial contact with the atmosphere, leaving the projectile body and the exposed fins.

The total mass of the projectile, with its payload, is estimated to be about 2 MT.

3.1.5.2 Earth Orbital Applications (Mission B) Projectile

Early in the study it was determined that for a reasonably sized launcher bore and a large projectile with an acceleration of 2500 g's, a mass of about 6.5 MT would be appropriate. Without conducting thermal, stability, aerodynamic analysis, preliminary mass and material characteristics were estimated. Masses for the tungsten nose cone, steel payload support structure (PSS), carbon/carbon thermal protection system and sabots were calculated based upon expected volumes and densities of materials (see Figure 3-16). For the 6500 kg Earth orbital applications projectile, a certain portion of the mass must be allocated to projectile's propulsion system and payload. The payload must be large enough to be practical. Preliminary mass estimates for the projectile indicate that approximately 2300 kg may be available for the projectile's payload and propulsion system.

The useful payload mass is a fraction of this value (2300 kg). The following relationships were used to estimate the useful payload mass:

$$m_t = m_{ps} + m_p + m_{pl} \qquad (1)$$

$$f = \frac{m_p}{m_{ps} + m_p} \qquad (2)$$

$$\Delta v = I \ln\left(\frac{m_t}{m_{ps} + m_{pl}}\right) \qquad (3)$$

where,

m_{ps} = propulsion system dry mass

m_p = main propellant mass

m_{pl} = useful payload mass

f = mass fraction of the propulsion system

Δv = the velocity impulse requirement at altitude in m/sec

I = specific impulse in m/s.

Solving these equations we arrive at:

DIMENSIONS, cm

PROJECTILE LENGTH	360
PROJECTILE DIAMETER	90
SABOT THICKNESS	26 - 5
OVERALL DIAMETER	100

ESTIMATED MASS CHARACTERISTICS, kg

INSTRUMENTS	30
MAXIMUM PAYLOAD	650
ASTRIONICS	25
ACS	50
PROPULSION SYSTEM (DRY)	425
PROPELLANT	1150
NOSE CONE	1020
FORWARD SABOT	200
AFT SABOT	100
PSS	2730
TPS	100
FINS	20
TOTAL	6500

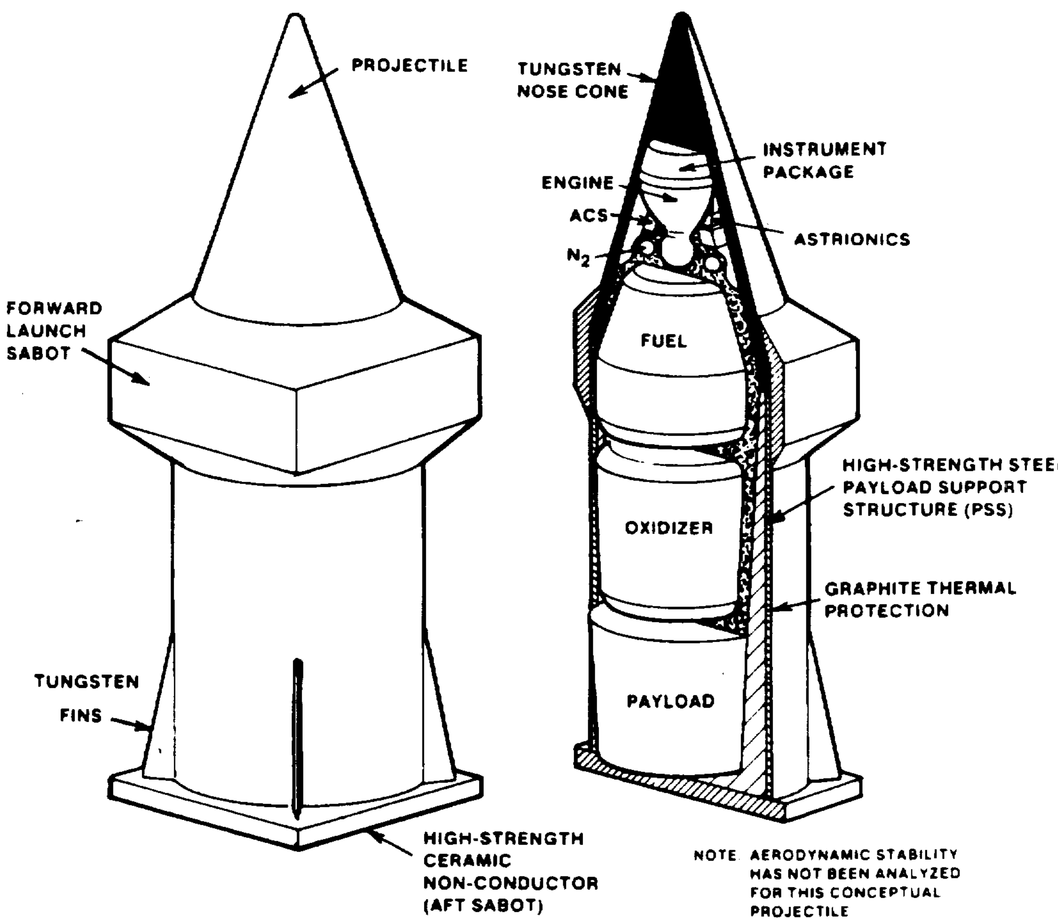


FIGURE 3-16. PROJECTILE CONCEPT FOR EARTH ORBITAL APPLICATIONS

$$\frac{m_{pl}}{m_t} = \frac{1}{\{1 + \gamma + (f/1-f)\gamma\}} \tag{4}$$

where,

$$\gamma = \left\{ \frac{1-e^{\Delta v/I}}{e^{\Delta v/I}-(1/1-f)} \right\}$$

$m_{ps} = \gamma m_{pl}$
$m_p = (f/1-f)m_{ps}$.

The results are plotted in Figure 3-17. Section 3.1.3.2 provides background information on the Δv's required to attain certain orbits. Figure 3-17 provides parametric data of various $\Delta v/I$ values, where I is specific impulse delivered by the propulsion system. For various values of $\Delta v/I$, and propulsion system mass fractions, f, the ratio of the payload mass to the total mass available for the propulsion system (wet) and payload is given.

Preliminary evaluation indicated the need for a simple hypergolic and high density propellant propulsion system with a high specific impulse. Many different propellant combinations were considered: RP-1/LOX, MMH/NTO, A-50/NTO; N_2H_4/ClF_3, etc. The most favored propellant system was N_2H_4/ClF_3, based upon specific impulse (I_{sp}), propellant density, stability, and ignitability (hypergolic). For the reference mission of a 500 km circular orbit, a Δv of approximately 2100 m/sec is needed (see Section 3.1.3.2). For an attainable value of specific impulse of 3000 m/s, the value of $\Delta v/I$ is 0.7. This (see Figure 3-17) means that the mass of useful payload, m_{pl} is 2300 kg x 0.28 = 644 kg (rounded to 650 kg). delivered to a 500 x 500 km orbit.

The propulsion system with its payload would be configured with its nozzle up and surrounded by an attitude control system and astrionics (see Figure 3-16). The current propulsion concept has a ClF_3/N_2H_4 pressure fed propulsion system with torroidal propellant tanks. The system would be designed to withstand the high g-loading--expected to be about 1100 g's. The propulsion system would have an oxidizer to fuel (O/F) ratio of 2.8, A_e/A_t = 14.0 for a chamber pressure of 100 N/cm^2 (150 psi). An I_{sp} of about 3000 m/s is predicted for these conditions with these propellants (Rowe, 1974). Roughly 1150 kg of propellant (850 kg ClF_3 and 300 kg of N_2H_4) would be needed. A mass fraction of 0.7 was assumed, giving the total propulsion system (including ACS and astrionics) mass of 500 kg. It was assumed that a 1000 s duration burn would accomplish the Δv burn at a 500 km altitude at a thrust level of about 110,000 N (25,000 lb_f).

It is estimated that about 240,000 cc of volume is possible for payload. A payload density of 2.7 g/cc would allow full use of the 650 kg payload mass potential. For payload densities less than 2.7 g/cc, and no increase in projectile mass (above 6500 kg), the payload mass is expected to vary with density. If the projectile were allowed to grow in length (larger PSS) for low density payloads, keeping the total projectile mass constant (at 6500 kg) additional payload volume (more than 240,000 cc) would be possible.

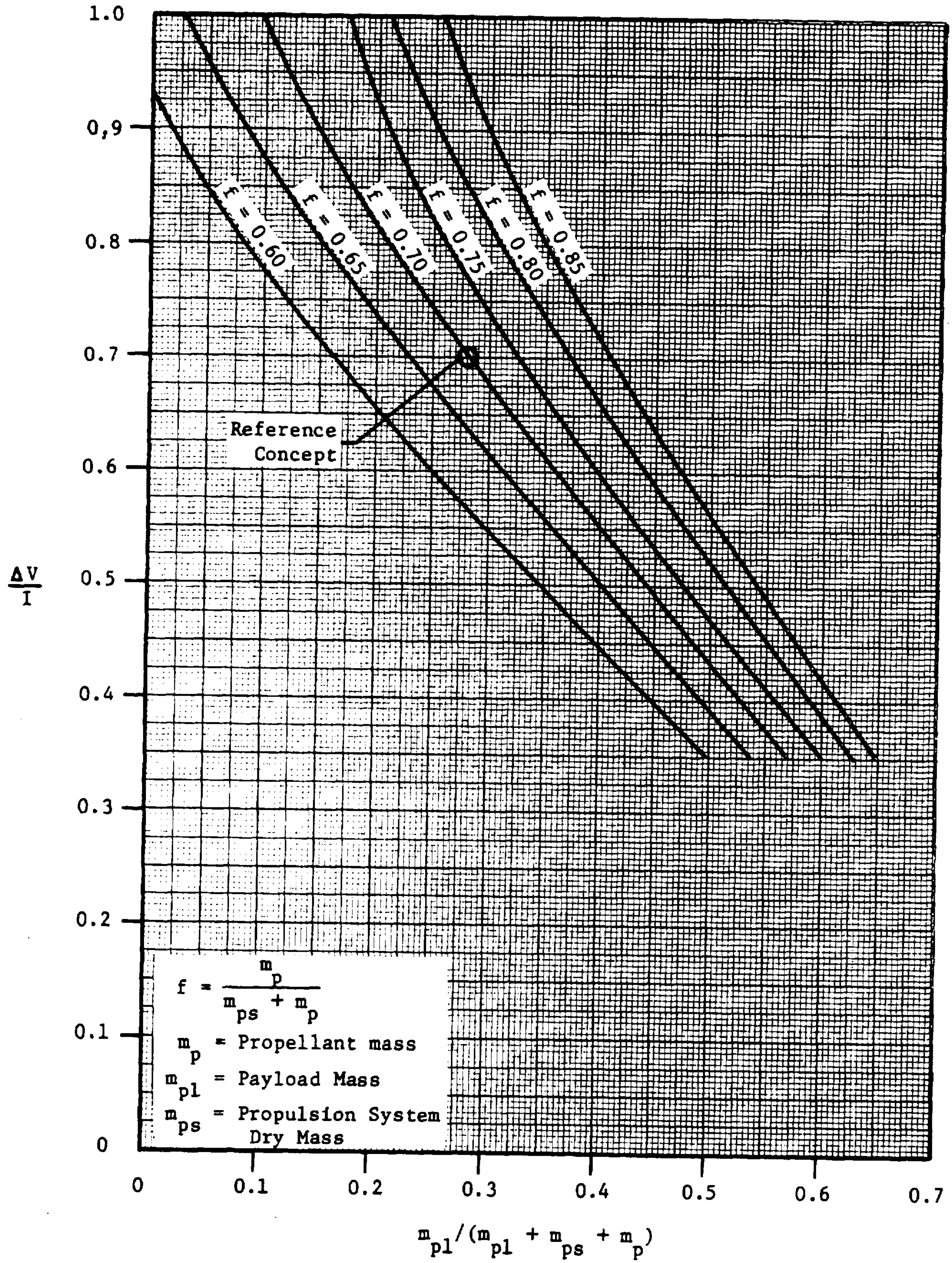


FIGURE 3-17. BASIS FOR THE PROPULSION SYSTEM AND PAYLOAD MASS ESTIMATES

Table 3-2 provides a summary of payload sizes and masses possible for a 6500 kg projectile launched into a 500 km circular orbit to support a space station type activity. Figure 3-18 provides a plot of the relationship between payload mass and payload bulk density. If the projectile were allowed to grow somewhat in length and mass, there would be no significant impact on the launcher; more energy would be required to be stored in the HPG's--the system is currently slightly over specified.

TABLE 3-2. PAYLOADS POSSIBLE WITH CURRENT CONCEPT

Payload Type	Density, g/cc	Payload Length, cm	Payload* Mass, kg
LN_2	0.81	80.0	264
Water (H_2O)	1.00	77.3	316
LO_2	1.14	75.4	352
Argon	1.40	72.1	415
Aluminum	2.70	58.0	650
Titanium	4.51	35.0	650
Iron	7.86	20.0	650

*Assumes zero mass for accommodating payload material within the payload volume.

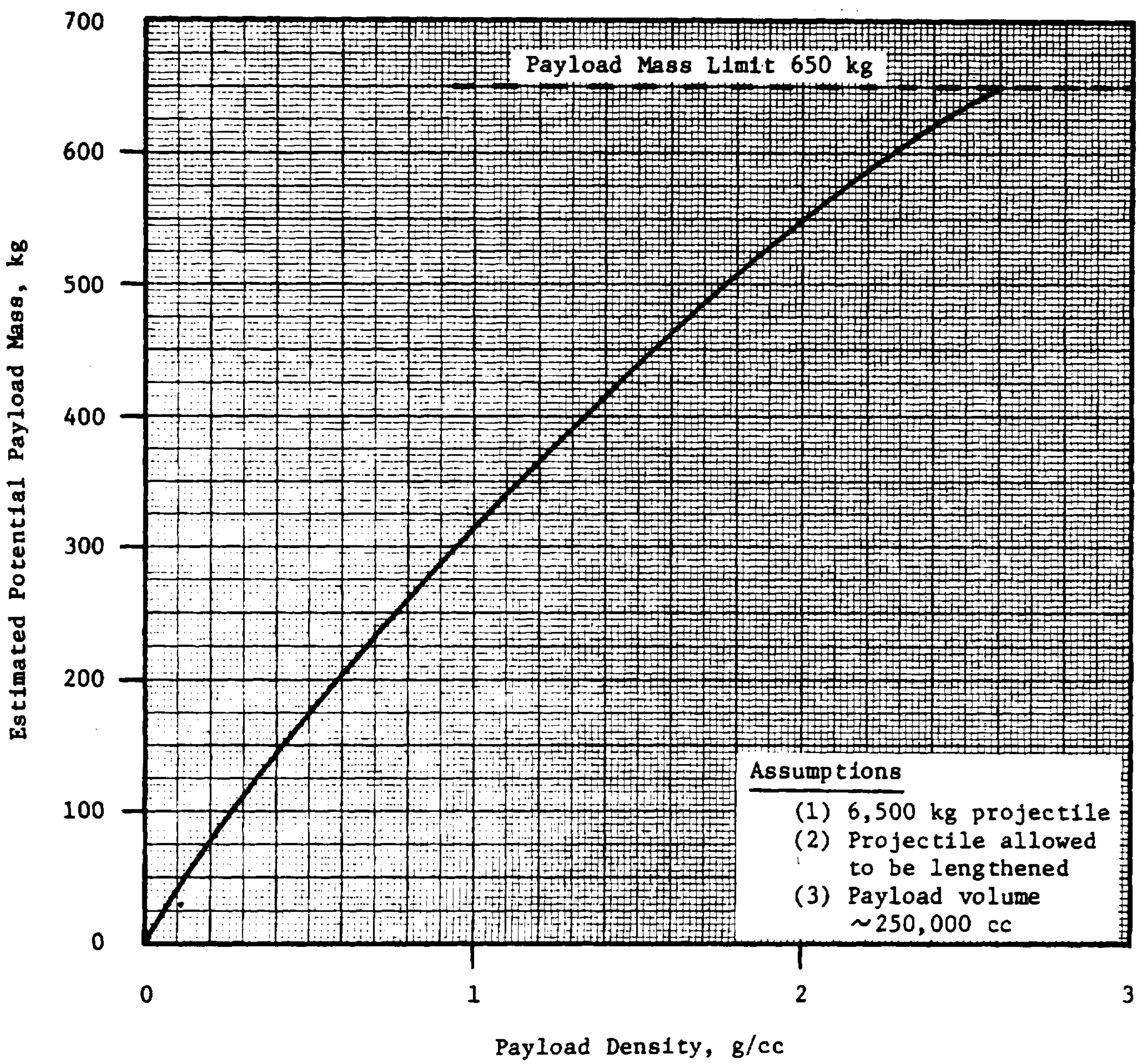


FIGURE 3-18. ESTIMATED POTENTIAL PAYLOAD MASS AS A FUNCTION OF PAYLOAD DENSITY

3.2 Distributed Energy Store Railguns (University of Texas)

This section of the final report presents an updated version of the presentation made on August 12-13, 1981, at the ESRL Concept Definition meeting by Dr. Richard A. Marshall. The major update in information is that the allowable maximum acceleration of the projectile was reduced from 30,000 g to 10,000 g. The launcher system addressed here is the distributed energy store (DES) system in which energy sources required to power the gun are distributed along the length of the gun (Marshall and Weldon, 1980; Marshall, 1979; Holland, 1981).

The modern ideas about railguns arose in the period 1968 to 1978 with the macroparticle acceleration project at the Australian National University (ANU). It began with John Barber's doctoral program in the Department of Engineering Physics (Barber, 1972) and ended with the attainment of a velocity of 5.9 km/s of a three-gram mass using a plasma armature in the Department's railgun (Rashleigh and Marshall, 1978). It was demonstrated that railguns do indeed work, and a clear understanding was obtained of what factors are involved in the design of railguns and railgun systems. Using this information it is possible to produce realistic railgun designs for a wide variety of applications, such as very high velocity research tools, hypervelocity weapons, and space-launchers.

In this section of the report, the basic conceptual design of a Distributed Energy Store (DES) Earth-to-space Rail-Launcher (ESRL) is presented together with analyses of many of the considerations involved in the conceptual design.

3.2.1 Background Information on Railgun Research

In the past two to three decades, much has been learned about the technologies that will be required to design, build, and operate a large railgun launcher. Historically, the first major demonstration was the construction of the homopolar generator (HPG) at the Australian National University (ANU) in Canberra. This machine showed that it was possible to make very large electromechanical energy stores in the 1 GJ range. It stores energy as rotational kinetic energy of two 40-ton rotors. This energy can be extracted electrically into a suitable circuit in about 1.5 seconds.

The other important factor demonstrated by the Canberra HPG is that solid brushes can be used to carry the very large currents involved for the second or so that is necessary (Marshall, 1966). The machine was originally designed to use sheet NaK jets to transfer current to and from the rotors. This was inconvenient, costly, and dangerous (Hibbard, 1962). The use of solid brushes on the machine has made its operation both convenient and safe and it has now been in regular use in ANU's Research School of Physical Sciences since about 1965.

The success of the Canberra HPG led other groups to apply the techniques learned to their own machines. One such group was the Center for Electromechanics at the University of Texas at Austin (CEM-UT) which has been in the HPG business since 1974 when a 5 MJ machine was built (Weldon et al, 1974). This machine was upgraded in 1981 to 10 MJ (Bullion, 1981) as a part of National Science Foundation program ("Rail and Seam Welding with the HRP Welding Process"--NSF Grant ISP/8005198).

Another lesson learned from the Canberra HPG is that from a material usage point of view it is inefficient. The energy storing elements, namely the rotors, have a mass of 73 MT compared with the yoke structure, which has a mass of 1,270 MT. Thus, less than 6 percent of the mass of the machine is useful for energy storage. An analysis of this situation (Marshall, 1981) led to an HPG concept in which, in principle at least, all of the magnetic circuit can be used as rotational energy store, the so-called all-iron-rotating (AIR) concept. The construction and testing of a 6.25 MJ AIR machine (Gully, et al, 1981) at CEM-UT will be accomplished by early 1982. Figure 3-19 shows a cross-sectioned view of the machine showing the rotor, stator, excitation coils, and electrical circuit. Figure 3-20 is a drawing of the complete AIR HPG.

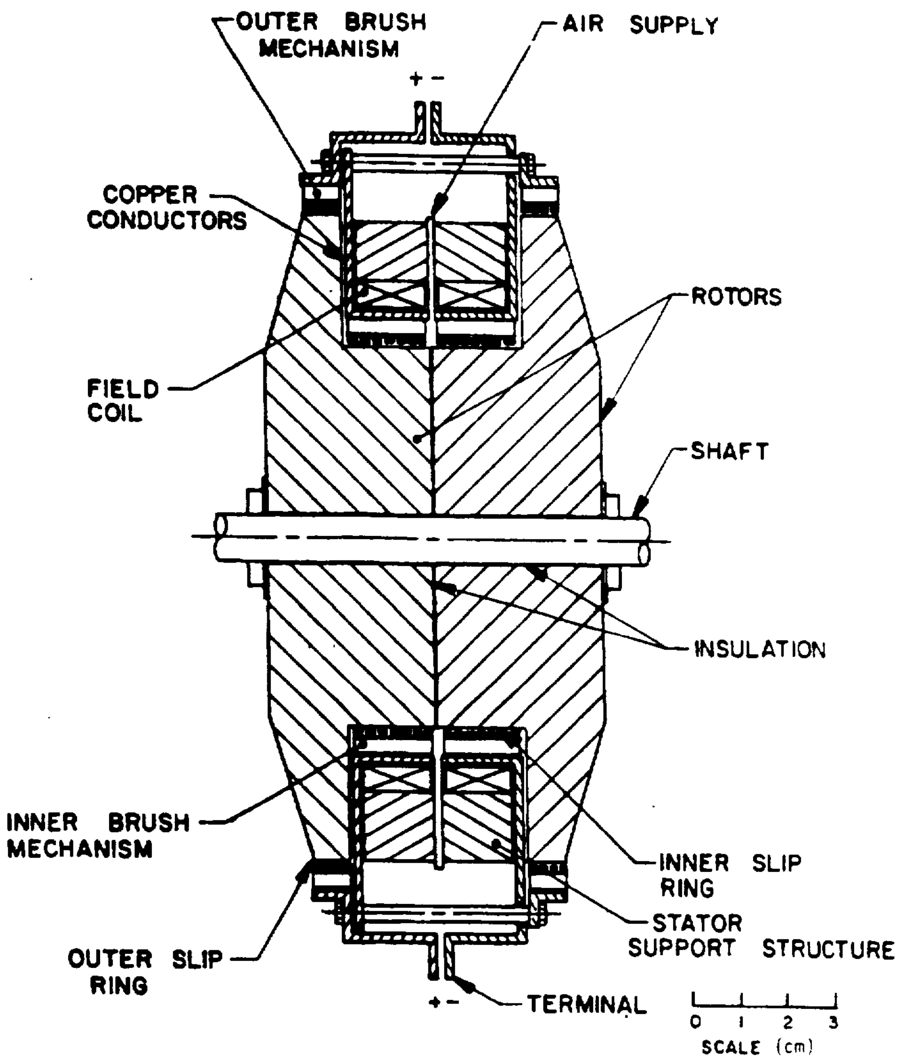


FIGURE 3-19. CROSS-SECTION VIEW OF THE 6.25 MJ AIR HPG

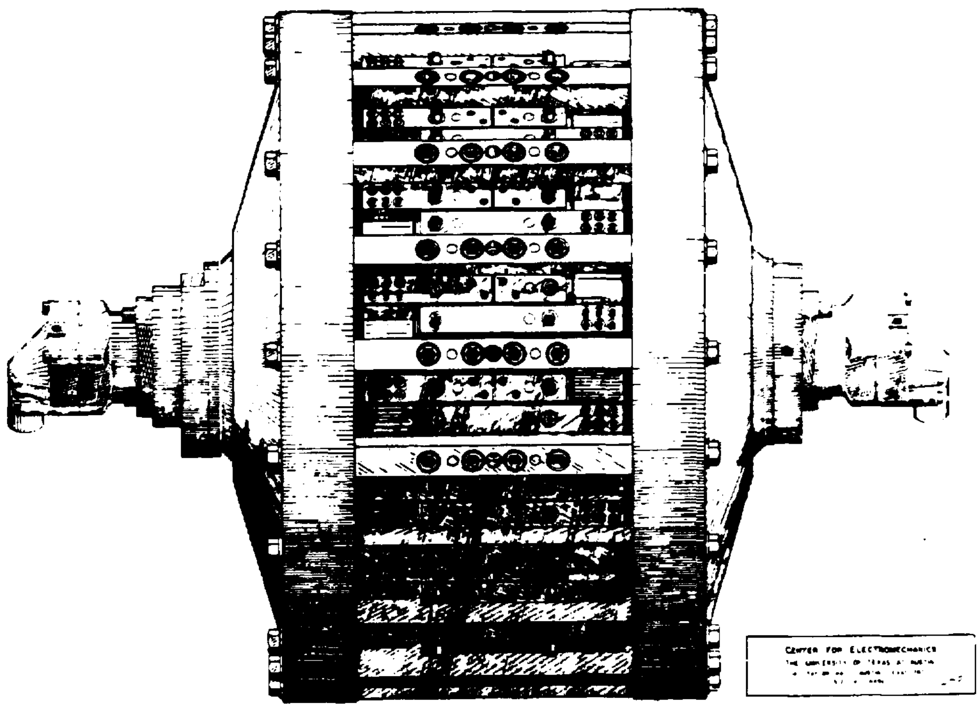


FIGURE 3-20. A COMPLETE VIEW OF THE 6.25 MJ AIR HPG

The principle on which a railgun operates is illustrated in Figure 3-21. An electric current is made to flow along one rail, across an armature to the other rail back down which it flows to the energy source. The current flowing in the rails produces a magnetic field between them in a direction normal to the plane which contains them. The armature experiences a force parallel to the rails as a result of the interaction of the current in it with the magnatic field between the rails. In any particular rail armature system, this force may be computed by integrating the down-gun components of J x B forces on all elements of the armature where J is the current density and B is the magnetic field due to the current in the rails and in the armature. There are two practical difficulties with this procedure however. The process is tedious and it also assumes that the current density is known at all points in the rails and armature; information not simple to find.

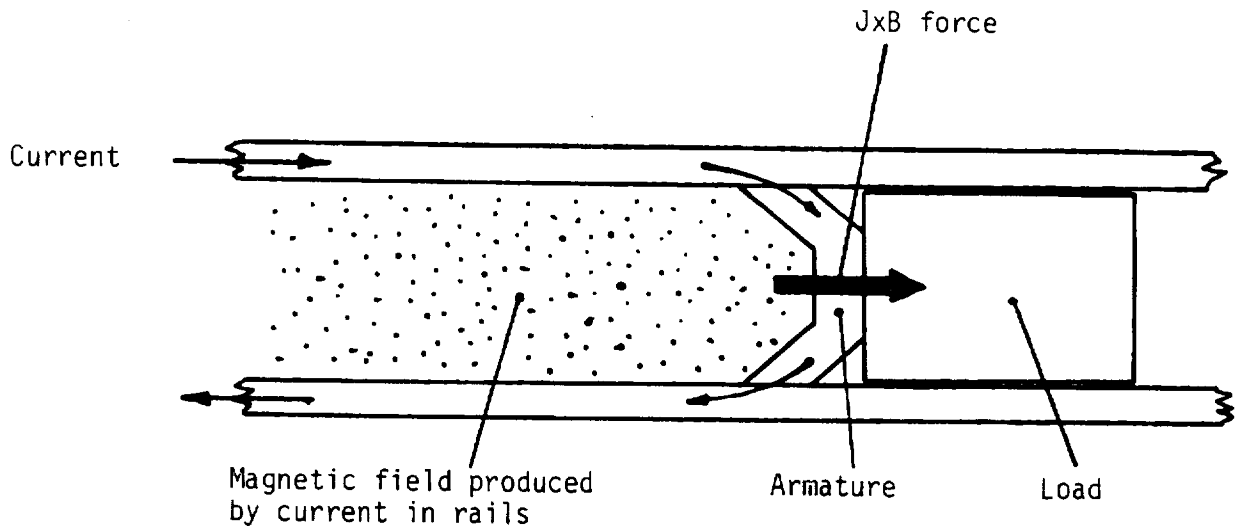


FIGURE 3-21. THE BASIC RAILGUN CONCEPT

However, there is a simple way to find the force. The formula for the force is given as:

$$F = 0.5\, L' I^2$$

where I is the railgun current, L' is inductance per unit length of the rails, and F is the force generated. The only uncertainty is the value of L'. In a well designed railgun it varies over only a small range. It depends on how far the current has penetrated into the rails near the projectile but calculations of L' show the effect to be small (Grover, 1962). As stated above, the computed value for the Canberra railgun was between 0.5 and 0.6 μH/m, depending on what assumptions were made. The value determined experimentally from the gun's performance was 0.42 μH/m (Rashleigh and Marshall, 1978). This was a small railgun, a bore of half-inch square, and the effect of mechanical friction between projectile and barrel would probably not be negligible. With larger guns, such as rail launchers, the effect of friction would diminish and L' would rise closer to its theoretical value.

The Canberra railgun system has shown that an HPG-inductor combination will provide the correct kind of current control to make a railgun work. Two other power supply systems have also been demonstrated. The railgun groups at LLNL and LASNL have shown that explosively driven flux compressors can also be successfully used (Fowler, 1980). A description of this method is given in Figure 3-22. The second system involves the use capacitors with pulse-shaping circuits to give a desired current wave form. Historically, one of the most ambitious railgun programs attempted was conducted by General Electric shortly after World War II. This program used capacitor banks having a total energy of 10 MJ plus pulse-shaping net works (Brate, 1957). These apparently worked although the gun itself was not a marked success.

More recently a capacitor-inductor system has been successfully used to power a railgun at the CEM-UT (Marshall and Stump, 1981). This gun has also demonstrated that copper rails may be used many times, about 70 shots having been made on the one pair of rails. It used two energy stores in tandem and is the forerunner of a more ambitious DES system.

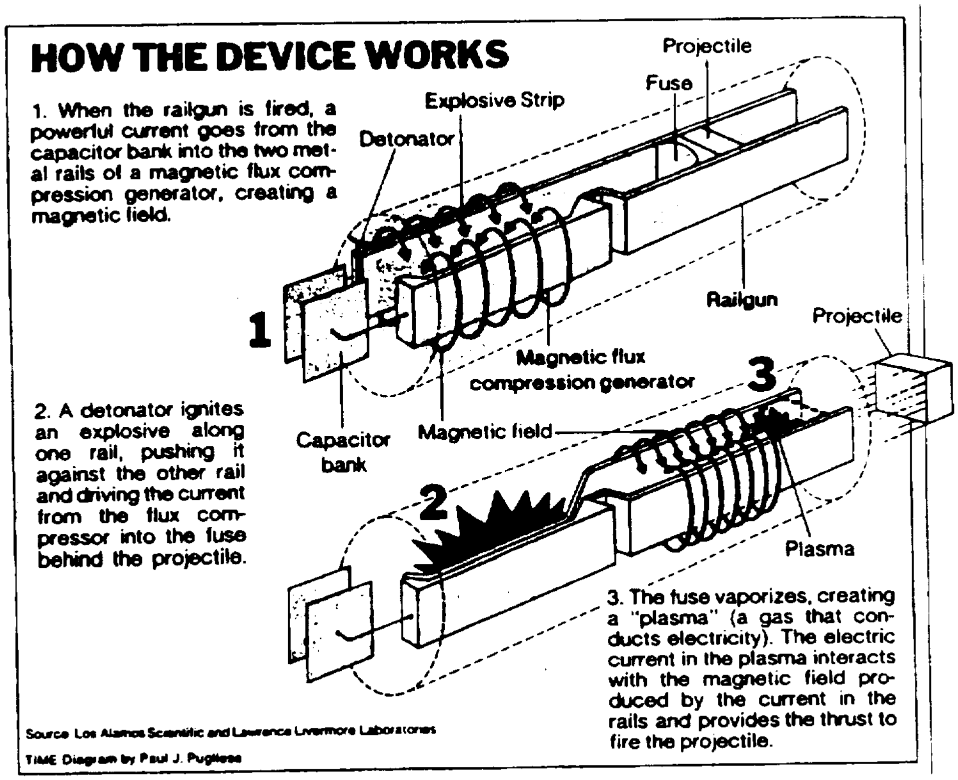


FIGURE 3-22. THE FLUX COMPRESSOR DRIVEN RAILGUN

3.2.2 ESRL Requirements

Table 3-3 lists the requirements for seven different candidate launchers, as selected by Battelle for parametric analysis. Projectile mass and diameter were selected on the basis of the desired payload and the stresses in the launch projectile during launch. The acceleration of 10,000 g's (1 g = 9.81 m/s^2) was chosen as being a reasonable compromise between projectile stresses and launcher length. Lower accelerations reduce the stress but increase the launcher length. The launcher exit velocity of 20 km/s was chosen to enable nuclear waste projectiles to be propelled from the surface of the Earth with sufficient velocity to penetrate the atmosphere and to have sufficient remaining velocity to escape from the solar system (see Cases A-1 through A-6). Case No. B-1 was included for the purpose of assessing the possibility of injecting payloads (with a ΔV capability) into Earth orbit.

TABLE 3-3. ESRL REQUIREMENTS FOR PARAMETRIC ANALYSIS

Parameter/Case No.	A-1	A-2	A-3	A-4	A-5	A-6	B-1
Projectile mass, kg	6,500	2,055	2,055	650	650	206	6,500
Projectile diameter, cm	55	55	17.7	17.7	9.9	9.9	55
Acceleration, g	10,000	10,000	10,000	10,000	10,000	10,000	2,500
Exit velocity, km/s	20	20	20	20	20	20	10

3.2.3 ESRL Analysis Summary

Table 3-4 lists the calculated parameters of the rail-launcher system based upon the requirements given in Table 3-3. Launcher length is calculated from exit velocity and acceleration ($v^2 = 2$ as). The force required to accelerate the projectile is determined from the projectile mass and acceleration (F = Ma).

TABLE 3-4. SUMMARY OF ESRL CALCULATED PARAMETERS (UT)

Parameter/Case No.	A-1	A-2	A-3	A-4	A-5	A-6	B-1
Launcher length, m	2,039	2,039	2,039	2,039	2,039	2,039	2,039
Acceleration time, ms	204	204	204	204	204	204	408
Force, N	638	202	202	63.8	63.8	20.2	159
Delivered energy density, MJ/m	638	202	202	63.8	63.8	20.2	159
Current, MA	50.5	28.4	28.4	16.0	16.0	9.0	25.2
Rail height, cm	119.4	67.1	67.1	37.8	37.8	21.3	59.6

From a system point of view, perhaps the most important parameter is the energy that must be delivered to the projectile per unit length of the launcher. The work done on the projectile as it is accelerated is equal to the accelerating force multiplied by the distance through which the point of application of the force moves. Thus, the delivered energy density is numerically equal to the accelerating force, and is constant along the launcher, when the acceleration is constant. It is this parameter which determines what a launcher will look like physically, as will be seen in the next section.

The current required to accelerate the projectile is obtained directly from the force from the expression, $F = 0.5\ L'\ I^2$, where L' is the inductance per unit length of the launcher rails. The value obtained experimentally (Rashleigh and Marshall, 1978) for L' in the Canberra railgun was 0.42 μH/m. For larger railguns such as the ESRL, L' will be larger (better) and closer to the theoretical value for a square-bore launcher with thin rails of 0.6 μH/m. The conservative value for L' of 0.5 μH/m has been taken in this work.

The final line in Table 3-4 gives the rail height. As is discussed below, the maximum pressure on the rails in the launcher is the same as the pressure on the projectile's sabot, i.e., the accelerating force divided by the launcher bore area. Assuming that the allowable normal stress on the face of the launcher rails is 65,000 psi (44,800 N/cm^2) then it is a straightforward matter to calculate the rail height.

With direction from Battelle, Case A-2 and Case B-1 were selected as the best candidates for the ESRL. These are summarized below.

In Case A-2, as can be seen from Table 3-3, it is assumed that the total projectile mass is 2,055 kg, that the average acceleration of the projectile in the launcher is 10,000 g's, and that the required exit velocity from the launcher is 20 km/s. From these requirements it follows that the launcher length is 2,039 m, that the accelerating force required is 202 MN, and that the total kinetic energy of the projectile at the moment of launch is 411 GJ. The acceleration time is simply exit velocity divided by acceleration (204 ms). From the accelerating force, the required launcher current is computed to be 28.4 MA, with the reasonable assumption that the inductance per unit length of the rails is 0.5 μH/m.

The bore of the launcher is now defined by the allowable pressure that the armature plasma plus the electromagnetic repulsion exerts on the rails. This is the same pressure as seen by the base of the projectile's sabot. If an allowable pressure of 65,000 psi (44,800 N/cm^2) is assumed (based on AMZIRC rails) then the bore is 67.1 cm square. (Note that the force exerted by the armature on the projectile is independent of launcher bore size.) The choice of bore size depends mainly on three things. Larger size and lower pressure will make it easier to hold the rails flat shot after shot. Smaller size increases the magnetic field between the rails and this will help energy store switch-on. The third factor is projectile and sabot design. This may be the most important of the three.

The launcher layout is dominated by the energy stores. The thrust of 202 MN means that 202 MJ must be delivered to the projectile per meter of gun length. Assuming an efficiency of transfer of energy from the inductors to the gun of 85 percent, and that the transfer of energy from homopolar (HPG) to inductor is also 85 percent, then the overall efficiency is 72 percent. Thus the HPG energy density required along the launcher is 280 MJ/m. If the energy stores are spaced at five per meter, then each HPG will require have an energy of 56 MJ, it being a machine of about 1.8 m diameter by 1.5 m long and weighing about 10 MT.

The inductors will store 48 MJ of energy at a current of 4 MA. To charge them with the assumed efficiency of 85 percent, they must have a resistance of no greater than 2.7 $\mu\Omega$ (assuming an HPG voltage of 110 V).

A preliminary optimization for inductors of the coaxial type (chosen because they produce no external magnetic field) indicates that if liquid nitrogen cooled aluminum is used, each inductor will have a mass of between 1.0 and 1.5 MT. (Note that a room temperature inductor of aluminum will have a mass of 23 MT. It will also be twice the size, i.e., eight times the volume.) Inductor mass is quite insensitive to the number of turns N. Inductor dimensions depend more strongly on N, being smaller for larger N. For an N of four the inductor has a diameter of 1.8 m and a length of 1.5 m. This matches the size of the HPG's nicely. To enable the HPG-inductor assemblies, as shown in Figure 3-23, to be fitted in along the length of the launcher at the required density, they may have to be arranged around the gun bore (at 30° angular increments) as shown in Figure 3-24. The total number of HPG inductor energy store assemblies required is about 5 x 2,040 = 10,200. For cost estimation purposes, it is reasonable to assume that 10,000 assemblies are required.

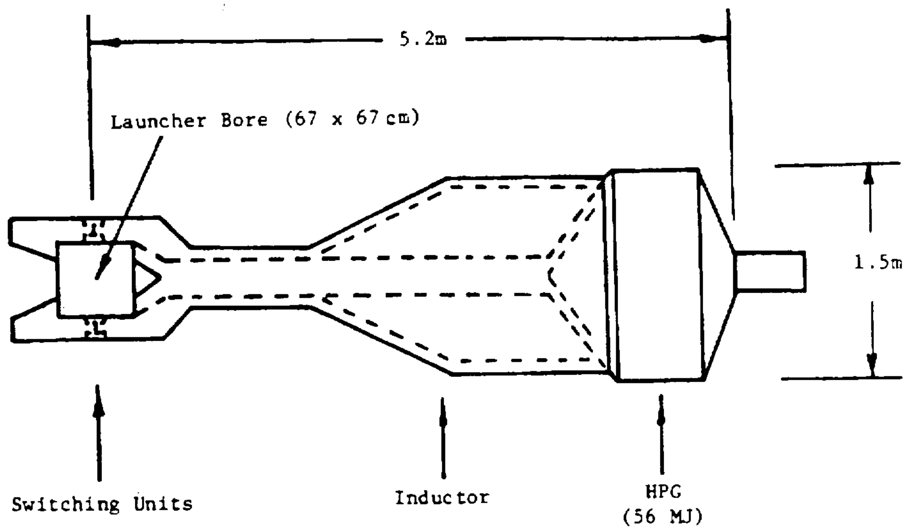


FIGURE 3-23. HPG/INDUCTOR ENERGY STORE UNIT

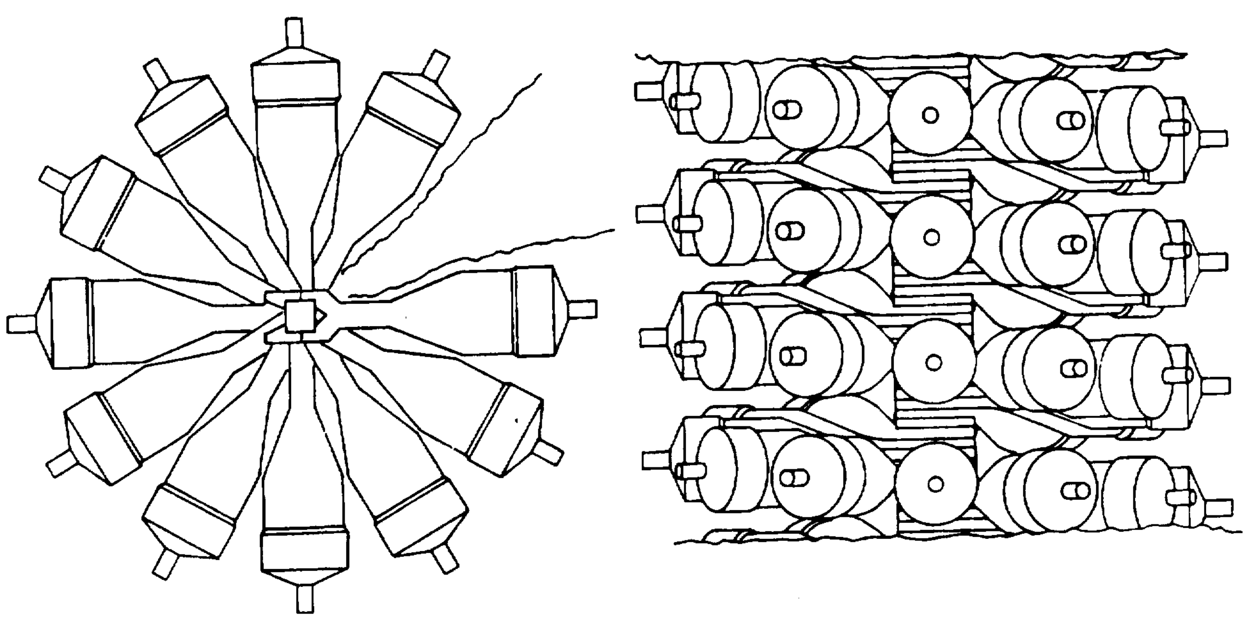


FIGURE 3-24. SPIRAL ARRANGEMENT OF ENERGY STORES
AROUND CENTER LINE OF LAUNCHER TUBE

Examination of the costs associated with the manufacture of the all-iron-rotating HPG lead us to believe that the likely cost of producing a run of 10,000 HPG's would be $1,000 to $1,500 per MJ. Thus each 56 MJ HPG would cost around $70,000, the cost of 10,000 machines being $700 million. By comparison the inductors are simple devices and their cost will be closely related to the bulk cost of high conductivity aluminum bar stock. Less is known about just what the detailed design of the switching units will be, but we think that their cost, together with the cost of the 0.2 m length of launcher to which each switch and energy store assembly is connected, will be low compared with the cost of the HPG's; perhaps as low as 20 percent.

In Case B-1, the overall "appearance" of this case is very similar to that of Case A-2. The increased projectile mass of 6,500 kg and reduced acceleration of 2,500 g's combine to give a maximum acceleration force (MN) and energy (MJ/m) delivery requirement for unit length of launcher of about the same, namely 159 compared with 202, i.e., 79 percent. Because the required launch velocity is reduced to one-half, while acceleration is reduced to one quarter, the maximum launcher length remains unchanged at 2,039 m. The maximum armature current is slightly smaller (25.2 MA) as is the launcher bore (59.6 cm square). The maximum kinetic energy at launch is 325 GJ which is 79 percent of that for Case A-2. Note that if the launch velocity in Case B-1 is reduced, then the kinetic energy at launch is reduced as the square root of velocity.

3.2.4 ESRL Detailed Analysis

The following section presents the results of ESRL analysis performed by Dr. Richard Marshall, CEM-UT. Topics include:

- Specific ERSL Subsystem Analysis
  - Armature
  - Rails
  - Energy stores
  - Projectile
- ESRL Simulations
- Switching Issues

3.2.4.1 Specific ESRL Subsystem Analysis

This section presents a technical discussion of analyses and issues pertaining to specific ESRL subsystems. Discussion includes the armature, rails, energy stores, and projectile.

3.2.4.1.1 Armature

To simulate the ESRL systems there are a number of important parameters that must be known. These are discussed below in terms of a plasma and solid aluminum armature.

Plasma Armature. To simulate the performance of ESRL systems, it is desirable to know what voltage drop can be expected across the armature. The observed rail-to-rail voltage drop across the plasma armature (measured at the muzzle of the gun) in the Canberra railgun (half-inch bore) was 150 V and roughly constant for all current from 100 kA to 300 kA. It is estimated (Powell and Batteh, 1980; and McNab, 1980) that 1/3 of the volt drop occurred at each rail, leaving 50 V for the resistive drop in the plasma. Thus, the resistive drop is about 39.4 V/cm (100 V/inch).

It is to be expected that the plasma conditions in the two ESRL's being considered will be similar to those in the Canberra railgun. In Case A-2, the launcher bore is 67.1 cm square. Therefore, the expected plasma resistive volt drop would be 2,650 V, to which 100 V should be added for rail drop, giving a total armature volt drop of 2,750 V.

In Case B-1, the calculated launcher bore is 59.6 cm square, giving a total armature volt drop of 2,450 V.

Solid Aluminum Armature. It is instructive to examine the possibility of using metal armatures in case for some reason it turns out to be undesirable to use a plasma armature. Because of gouging, it is not likely that a metal armature sliding on the rails will be satisfactory above speeds of one kilometer per second (Barber, Marshall, and Muttick, 1974). It might, however, be desirable to have a metallic armature structure to carry current most of the distance from rail to rail with a small plasma gap at each end of it to complete the circuit.

The use of such metallic armatures is possible for ESRL applications, where it is not attractive in small-bore railguns. The reason is that the resistive temperature rise of an armature for a given armature velocity in a railgun (when the armature alone is being accelerated) is directly proportional to the thickness of the armature in the direction of motion, and is independent of the bore of the gun (Marshall, 1979). For practical reasons, armature thickness is limited to some fraction of the bore size, thus a large bore favors a large armature thickness.

To calculate the parameters for an aluminum armature for Case A-2, first note that the current density squared times time required to raise aluminum from liquid nitrogen temperature to its melting point of 660 C is 0.58×10^9 $(A/cm^2)^2s$. This is known as the "action constant" and it takes into account the change of electrical resistivity with temperature. In Case A-2, a current of 28.4 MA has to be carried for a time of 204 ms. Therefore conduction area required is 533 cm^2, i.e., an armature 7.9 cm thick in the 67.1 cm bore gun--quite a reasonable proposition. The mass of this armature would be 98.5 kg which is less than 5 percent of the total projectile mass. The volt drop in this armature (at room temperature) is a negligible 11 V. The armature thickness is about the same as that required for a plasma armature (see below), so the volt drop for the total armature including the drop due to the plasma end conduction would be only a few hundred volts.

Similarly, in Case B-1, armature thickness required is 12.6 cm giving an armature mass of 124 kg, being less than 2 percent of the total projectile mass. The total armature volt drop will again be about 300 V. An armature like this will also be quite workable.

3.2.4.1.2 Rails

The following section discusses rail resistance and rail pressure.

Rail Resistance. For ESRL simulation it is desirable to have a simple expression for rail resistance. The depth d to which current will penetrate a conducting rail in time t is given by:

$$d = \sqrt{\rho t/2\mu}$$

where ρ is the resistivity of the rail material (2×10^{-8} ohm·m, for copper) and μ is the permeability of free space ($4\pi \times 10^{-7}$).

At ESRL speeds, current is carried for 5 to 10 m in the rails. Thus, choice of a characteristic length of 1 m will give a conservative, i.e., high, resistance. For this length,

$$t = 1/v$$

giving

$$d = 0.09/\sqrt{v}$$

Thus, rail resistance per unit length, R', is given by

$$R' = 0.4 \times 10^{-6}\sqrt{v} \quad \text{(ohm/m)}$$

This is a reasonable expression to use for both Case A-2 and Case B-1 (rail height is close to 0.6 m for both).

It is occasionally and incorrectly said that a barrier to achieving high velocities in a railgun is that rail resistance will get to unacceptably high values. The following simple argument shows why this is not so in the case of the DES railgun systems.

As noted above, skin depth d is proportional to $\sqrt{\rho t}$ where ρ is the resistivity of the rail material. Because in a DES railgun the pattern of current with respect to distance backwards from the armature has a nearly constant shape, the time t in the above expression is proportional to reciprocal velocity, giving d proportional to $\sqrt{\rho}/v$. Thus, R' is proportional to $\rho/\sqrt{\rho}/v$

$$= \sqrt{\rho v}$$

The I^2R loss per unit length of rail is proportional to I^2R' times the time taken for the current wave to pass, i.e.,

$$\sqrt{\rho v} \times 1/v$$
$$= \sqrt{\rho / v}$$

This expression shows that the lower rail resistivity gives lower losses even though it also gives smaller skin depths. There is therefore no advantage in using higher resistivity rail materials to increase skin depth. It also shows that resistive losses decrease as velocity increases.

Rail Pressure. The general construction of square bore railgun launchers would be generally like that shown in Figure 3-25, but many other specific construction methods are possible. A pair of electrically conducting rails are held at a constant distance apart with spacers near each edge, the whole assembly being contained within a housing which performs the main functions of keeping the rails and spacers accurately located, and can withstand the forces generated when the railgun is fired.

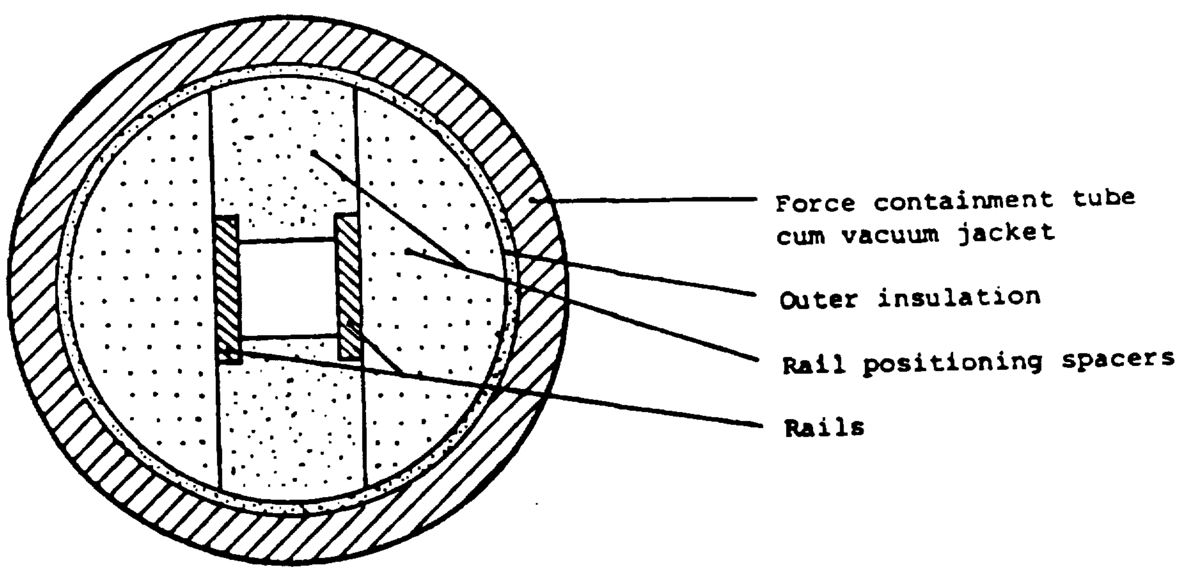


FIGURE 3-25. CROSS-SECTION OF THE CANBERRA RAILGUN

In a rail-launcher in which a plasma armature is used, the projectile is in fact propelled by the pressure of the plasma on its rear-most face. The projectile must therefore fit the launcher bore in a gas tight manner to prevent the loss of plasma past it. In that sense it is like a regular gas pressure gun. The difference is that in the latter, the pressure is carried all the way back to the gun breech. In a railgun the reaction pressure is provided by the interaction of the armature current with the field produced by the current in the rails, i.e., by the railgun effect.

The pressures developed in a railgun are shown in Figure 3-26. The pressure on the back of the projectile is simply the 0.5 $L'I^2$ force divided by the bore area. Observations made in the Canberra railgun and subsequent theoretical work (Powell and Batteh, 1980; and McNab, 1980) indicate that the plasma armature is typically 10 cm thick. Thus, the pressure in the plasma falls off with distance rearwards as shown. If the plasma is in static equilibrium across the whole back face of the projectile, then the pressure on the rails and spacers will fall as the plasma pressure falls. The electromagnetic pressure on the rail is readily found at any point by computing the magnetic field due to the current in the opposite rail and in the armature and multiplying it by the current per unit width at that point. The e.m. pressure rises from zero at the back face of the projectile to a maximum at the back of the plasma, and then falls to about three quarters of the maximum plasma pressure a few launcher diameters back where the field produced by the current in the armature has become small. It is interesting to note that in the plasma region, the sum of the e.m. pressure and plasma pressure gives a smooth curve.

In the case of launchers where the launcher bore is fairly large compared with the 10 cm thickness of plasma, it will probably be possible to prevent the plasma from coming in contact with the spacer by shaping the back of the projectile as shown in the lower sketch in Figure 3-26. If this were done, then the only functions the spacers would perform would be to hold the rails apart and to act as guides for the projectile. A gas-tight seal between projectile and spacer would not be required, which may be a valuable point in easing possible problems in the design of energy store switches.

3.2.4.1.3 Energy Store

The remaining numerical value needed to simulate launcher performance is the inductance, L, of the inductor of each energy store.

For Case A-2, as stated above, 202 MJ must be delivered to the projectile per meter of launcher length. At five energy stores per meter, then each store must deliver 40 MJ to the projectile. For 85 percent energy transfer, then each inductor should contain 48 MJ when fully charged. Assuming also that peak current in each inductor is 4 MA then L may be found from the energy expression,

$$\text{Energy} = 0.5\, L\, I^2$$

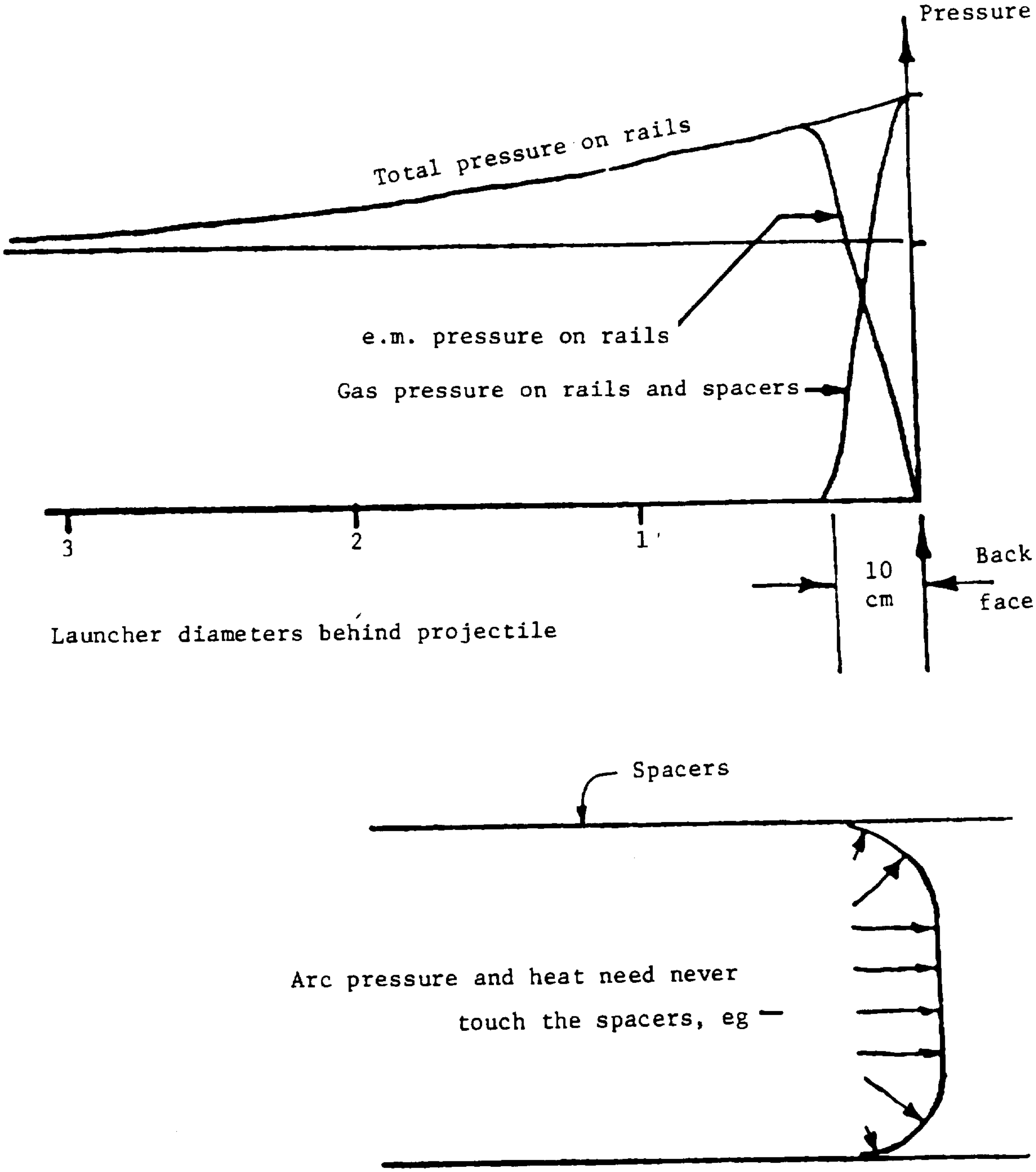


FIGURE 3-26. PRESSURES ON RAILS AND SPACERS

giving

$$L = 6\ \mu H$$

For Case B-1, 159 MJ must be delivered to the projectile per meter of launcher length. Again, taking five stores per meter with 85 percent efficiency of energy transfer, then each inductor should hold 37.4 MJ, when charged. Assuming a current of 3.2 MA peak in each inductor, then the inductance required is found to be 7.3 μH. Note that the choice of a smaller value of peak current gives a larger value of inductance--it is energy per store that is fixed--and the effect on launcher behavior will be to have smaller current peaks as each store switches on, and more stores in action at any one time. The reverse will happen when larger currents are chosen. The current peaks will be larger and fewer stores will be in action at a time. The efficiency of energy transfer is affected to only a small degree by the choice.

3.2.4.1.4 Projectile

It is well known that launching a long slender projectile at high accelerations produces high stresses in it. The fact is illustrated by considering a one-inch cube of steel. A pressure on its base of 0.28 psi will cause it to accelerate at one g (0.28 lb/in^3 is the density of steel). To accelerate the cube at 10,000 g, the base stress required is 2,800 psi. A 20-inch-long cylinder would require a base stress of 56,000 psi.

These data indicate that to propel long projectiles, special techniques may be necessary to keep the stresses acceptably low. One possibility is shown in Figure 3-27. It might be possible to fasten a series of "sails" (for want of a better name) along the projectile. If gas pressure can be maintained between the sails such that the pressure difference across each sail is the same, then the propelling force would be divided equally between the sails' attachment points. This would reduce the maximum stress in the projectile by a factor equal to the number of sails. The pressure distribution between sails might be maintained by causing a continuous flow of gas to pass forward from space to space through some kind of pressure relief valves. It might be possible to generate this gas by having the armature plasma ablate it from the rear face of the rear-most sail.

3.2.4.2 ESRL Simulations

The following section discusses the railgun simulations that were conducted in support of the ESRL assessment.

3.2.4.2.1 Launcher System

To simulate the performance of the DES launcher, a parametric model must be constructed, as shown in Figure 3-28 (Marshall and Weldon, 1981; Marshall, 1976). Each store is represented by the inductor L with its associated resistance R. Each inductor is delivering current to the launcher

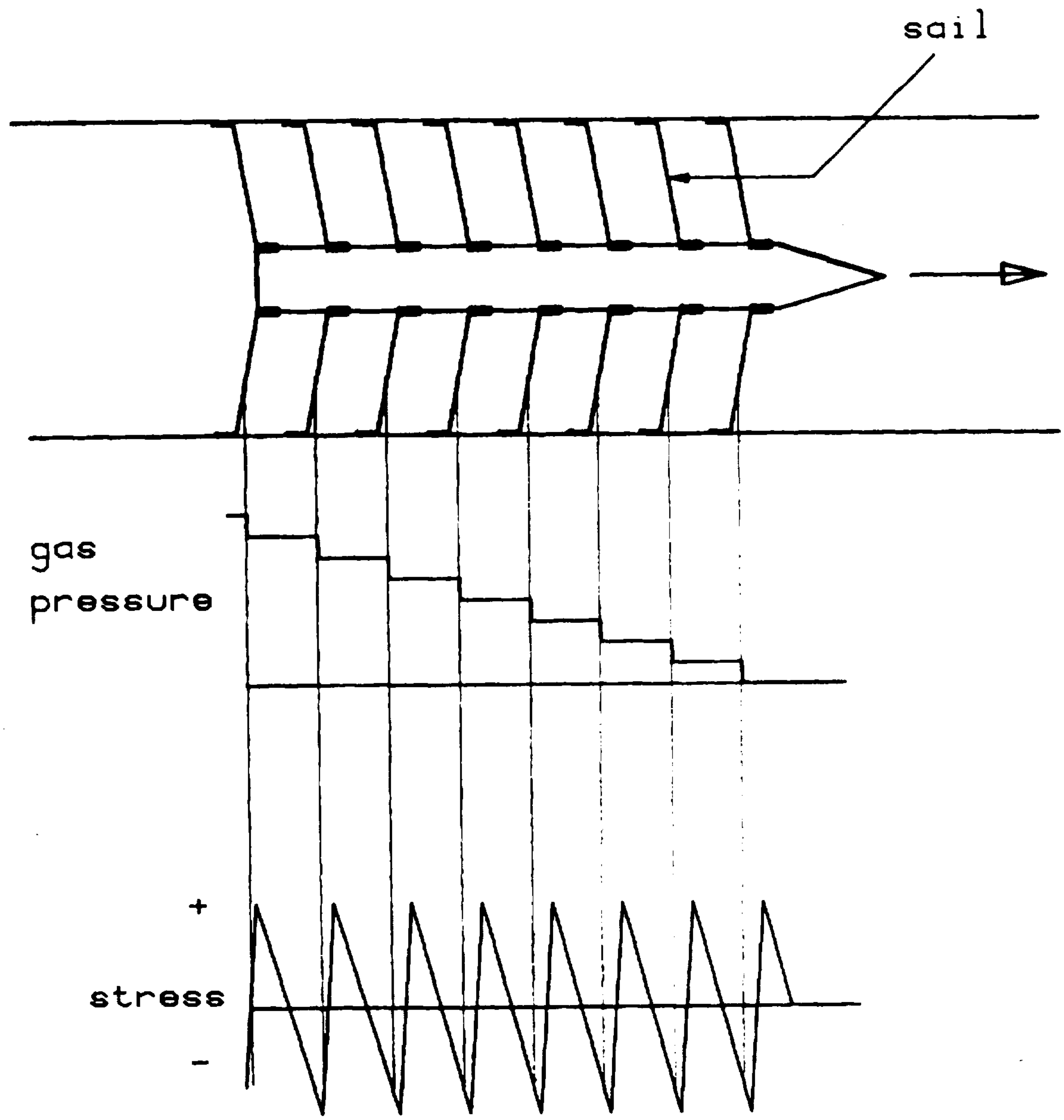
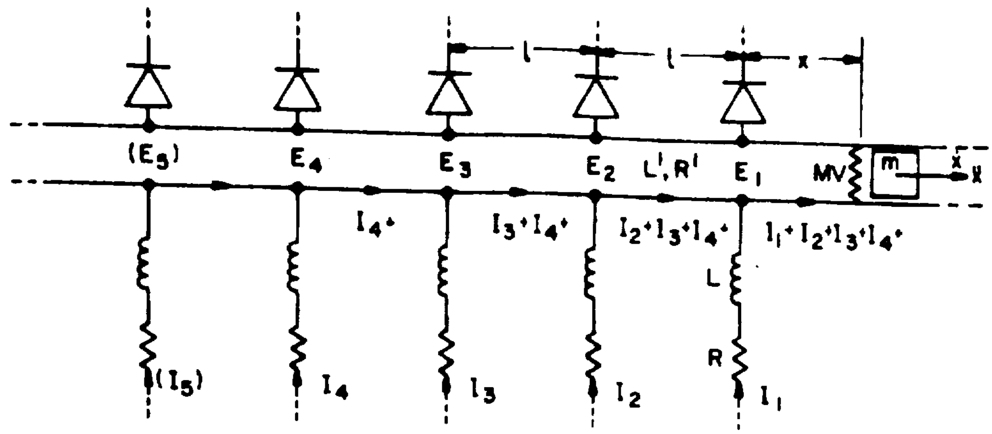


FIGURE 3-27. CONCEPT OF DISTRIBUTING PROPULSION FORCES ALONG A LONG PROJECTILE



Circuit diagram showing parameters and variables used in the analysis.

$$
\begin{array}{llll}
-E = L\dot{I}_1 + RI_1 & E_1 = & L'x(\dot{I}_1+\dot{I}_2+\dot{I}_3+\dot{I}_4+) + & (L'\dot{x} + R'x)(I_1+I_2+I_3+I_4+) + MV \\
-E_2 = L\dot{I}_2 + RI_2 & E_2-E_1 = & L'\ell(\dot{I}_2+\dot{I}_3+\dot{I}_4+) + & R'\ell(I_2+I_3+I_4+) \\
-E_3 = L\dot{I}_3 + RI_3 & E_3-E_2 = & L'\ell(\dot{I}_3+\dot{I}_4+) + & R'\ell(I_3+I_4+) \\
-E_4 = L\dot{I}_4 + RI_4 & E_4-E_3 = & L'\ell(\dot{I}_4+) + & R'\ell(I_4+) \\
\cdots & \cdots & \cdots & \cdots
\end{array}
$$

Eliminating the voltages E_n gives

$$
\begin{array}{lll}
L(\dot{I}_1) + L'x(\dot{I}_1+\dot{I}_2+\dot{I}_3+\dot{I}_4+) = & R(-I_1) & - (L'\dot{x} + R'x)(I_1+I_2+I_3+I_4+) - MV \\
L(-\dot{I}_1+\dot{I}_2) + L'\ell(\dot{I}_2+\dot{I}_3+\dot{I}_4+) = & R(I_1-I_2) & -R'\ell(I_2+I_3+I_4+) \\
L(-\dot{I}_2+\dot{I}_3) + L'\ell(\dot{I}_3+\dot{I}_4+) = & R(I_2-I_3) & -R'\ell(I_3+I_4+) \\
L(-\dot{I}_3+\dot{I}_4) + L'\ell(\dot{I}_4+) = & R(I_3-I_4) & -R'\ell(I_4+) \\
\cdots & \cdots & \cdots
\end{array}
$$

giving

$$
\begin{bmatrix}
(L+L'x) & L'x & L'x & L'x & \cdots \\
-L & (L+L'\ell) & L'\ell & L'\ell & \cdots \\
0 & -L & (L+L'\ell) & L'\ell & \cdots \\
0 & 0 & -L & (L+L'\ell) & \cdots \\
\cdots & \cdots & \cdots & \cdots & \cdots
\end{bmatrix}
\begin{bmatrix}
\dot{I}_1 \\ \dot{I}_2 \\ \dot{I}_3 \\ \dot{I}_4 \\ \cdots
\end{bmatrix}
=
\begin{bmatrix}
R(-I_1) - (L'\dot{x} + R'x)(I_1+I_2+I_3+I_4+) - MV \\
R(I_1-I_2) \quad -R'\ell(I_2+I_3+I_4+) \\
R(I_2-I_3) \quad -R'\ell(I_3+I_4+) \\
R(I_3-I_4) \quad -R'\ell(I_4+) \\
\cdots
\end{bmatrix}
$$

Defining the matrix equation as

$$[A][B] = [C]$$

then values of the rates of change of the currents, $\dot{I}_n$, are obtained from the equation $[B] = [A^{-1}][C]$.

FIGURE 3-28. THE DES LAUNCHER SYSTEM AND EQUATIONS as shown, the current flowing in each launcher stage being also as shown. Each stage length is l and the armature has moved a distance x into stage number one. In the simulation, it is assumed that at the moment the projectile's armature passes an energy store input point, the current from that store begins to flow at full initial value. The diode symbol indicates that as each (rearmost) energy store current falls to zero that store is removed from the computation in progress.

3.2.4.2.2 System Equations

The first step in simulating the performance of a DES launcher is to derive an expression for computing the currents flowing in the stages. The method of doing this is shown in Figure 3-28. The first set of equations consists of equation pairs giving expressions for E the voltage at each energy input point. The first is in terms of the energy store parameters; the second is in terms of the launcher parameters where L' is its inductance per unit length and R' is its resistance per unit length. The equation pair for the first stage contains terms x, and the projectile velocity x-dot. The term MV is the volt drop from rail to rail across the armature (called MV for "muzzle volts" because it is the voltage as measured across a railgun's muzzle).

Elimination of E from the equation pairs gives the second set of equations, which can be solved for all I-dot in the form of the matrix equation shown.

The performance of the system may now be simulated instant by instant from any given starting point (such as the first energy store operating only, with full current), by solving first for rates of change of currents, I-dot, enabling the updated current values to be obtained. From the total current in the first stage the projectiles acceleration is obtained (Marshall, 1978). The updated velocity is obtained by adding the velocity increment, acceleration multiplied by the time step. Likewise x is updated by adding the increment, velocity times the time step.

3.2.4.2.3 Simulation of the Launcher, Case A-2

The simulation of Case A-2 gives the curves of current versus projectile travel shown in Figure 3-29. The first part of the curve, up to a distance of four meters along the launcher, shows how the driving current builds up stage by stage. A projectile velocity of 1 km/s is given at Z = 0. This is the assumed velocity of injection into the launcher. The other curve fragments are also total current versus projectile position, but at different velocities. In obtaining these fragments, the appropriate launcher resistance (listed below) was used. The program in each case was run a sufficient number of steps to allow the currents being delivered by the "nth" store at each switch time to remain steady. This took about 500 iterations. About 20 energy stores were in action at one time, most of the time.

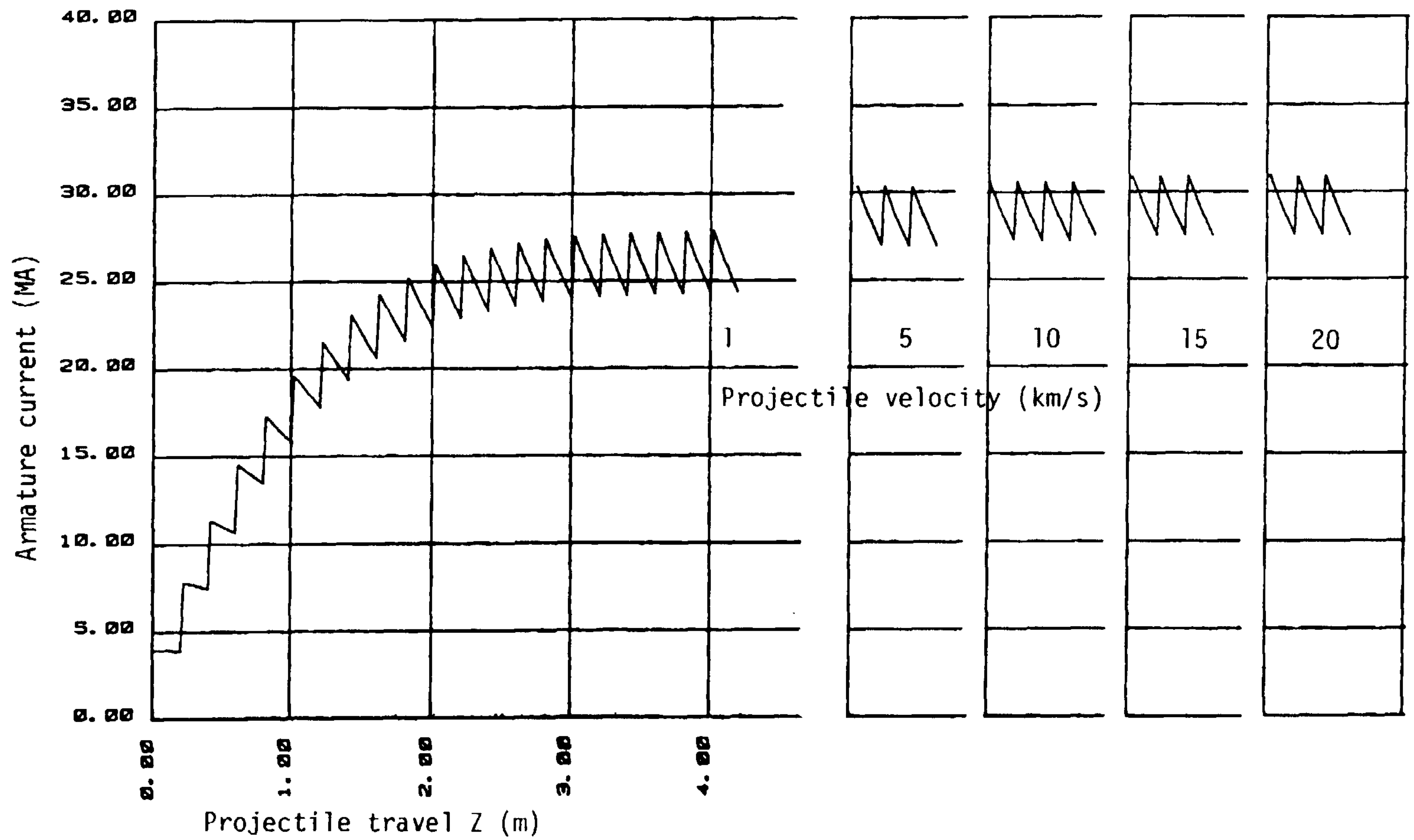


FIGURE 3-29. ARMATURE CURRENT VS. PROJECTILE TRAVEL, AND ARMATURE CURRENT AT DIFFERENT VELOCITIES, FOR LAUNCHER CASE A-2

The parameters used in Case A-2 were:

Coil inductance	6 μH	
Initial current	4 MA	
Initial inductor energy	48 MJ	
Inductor resistance	2.7 μΩ	
Launcher inductance	0.5 μH/m	
Launcher resistance	13 μΩ/m @	1 km/s
	28	5
	40	10
	49	15
	57	20
Projectile mass	2,055 kg	
Armature volt drop	2,750 V	
Stage length	0.2 m	
Initial velocity	1,000 m/s	

The energy delivered to the projectile as it passes through one stage is simply its increase in kinetic energy. The extra energy that has become available for projectile acceleration in this same pass is the energy of one store. Thus the ratio of these is the efficiency of transfer of energy from inductor to projectile. These are

at 1.2 km/s, transfer efficiency is 72.9 percent

5	87.5
10	89.0
15	90.6
20	90.8

indicating that the figure of 85 percent assumed in the summary above is realistic.

The computed average efficiency from Z = 0 to Z = 4 is about 50 percent where a considerable portion of the energy delivered has gone to "loading" the launcher with magnetic field. Once this "wave" is charged it moves on down the launcher without requiring any further charging energy. Since about 20 energy stores are in action at any one time, the "wave" energy is equal to about that contained in 10 stores. This is a very small part of the total energy involved.

3.2.4.2.4 Simulation of the Launcher, Case B-1

As for the previous case, Figure 3-30 shows driving current versus projectile travel for the first four meters of travel and at three different projectile velocities.

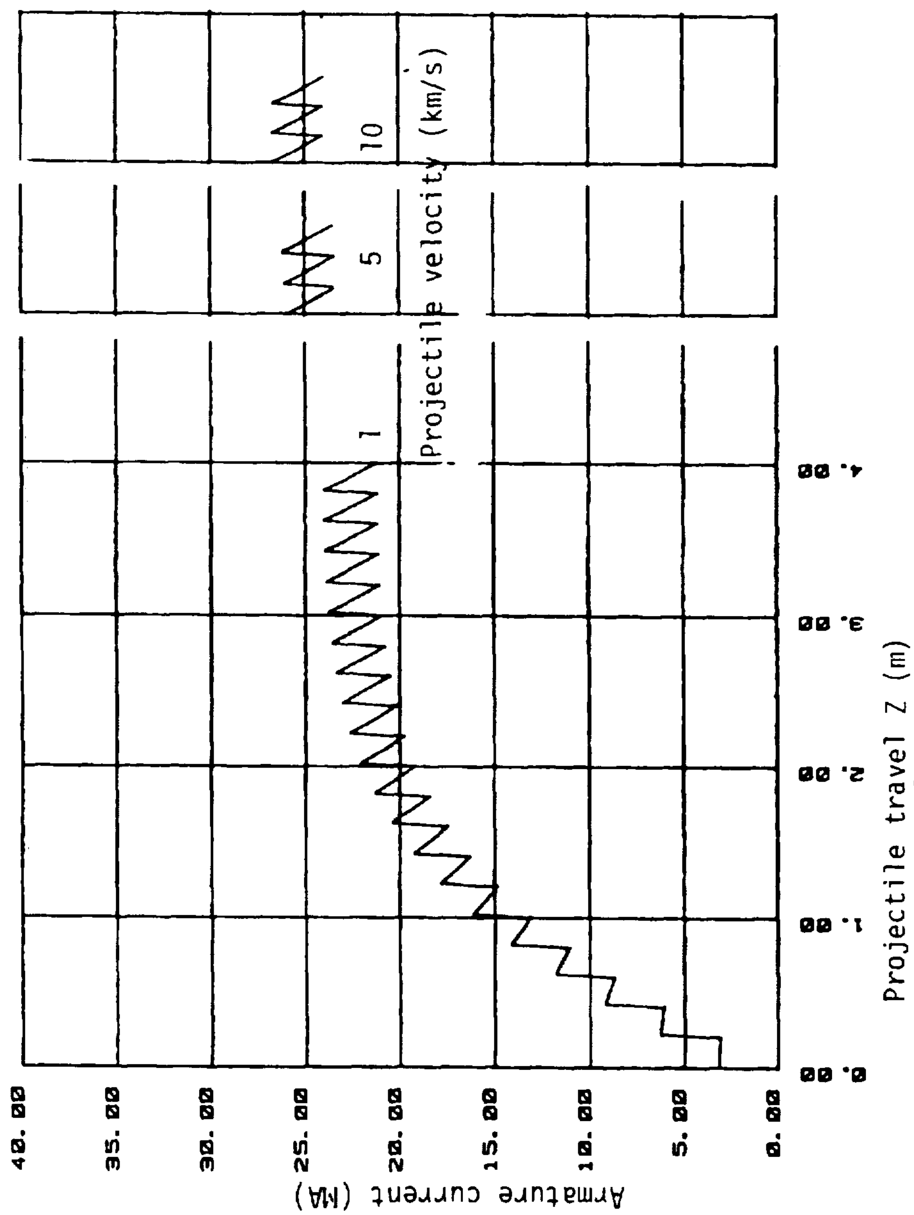


FIGURE 3-30. ARMATURE CURRENT VS. PROJECTILE TRAVEL, AND ARMATURE CURRENT AT DIFFERENT VELOCITIES, FOR LAUNCHER CASE B-1

The parameters used in Case B-1 were

Coil inductance	7.3 μH
Initial current	3.2 MA
Initial inductor energy	37.4 MJ
Inductor resistance	3 μΩ
Launcher inductance	0.5 μH/m
Launcher resistance	13 μΩ/m @ 1 km/s
	28 5
	40 10
Projectile mass	6,500 kg
Armature volt drop	2,450 V
Stage length	0.2 m
Initial velocity	1,000 m/s

The efficiency of energy transfer, inductor to projectile, is computed to be

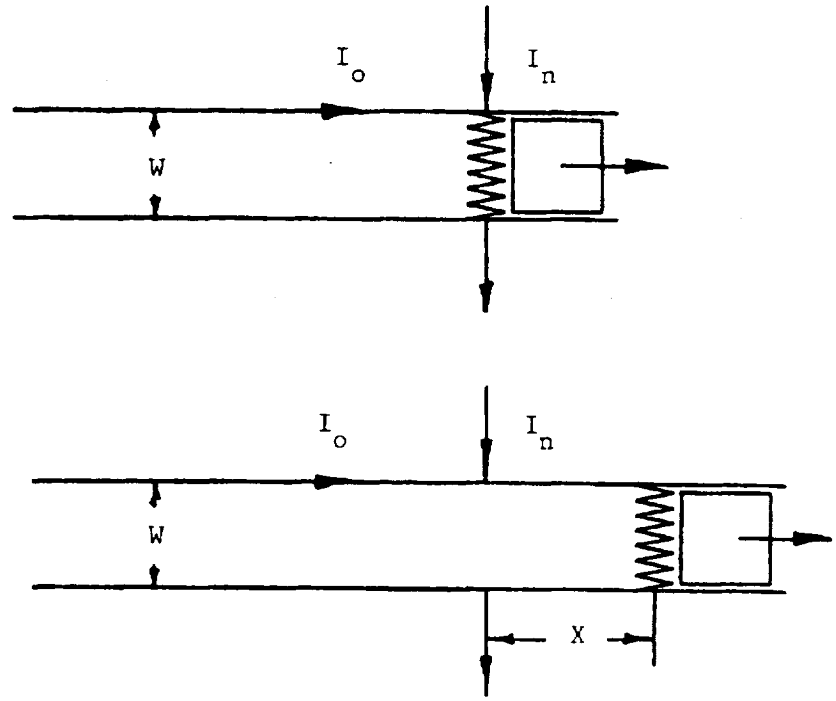
at 1.05 km/s,	70 percent
5	84.5
10	88.8

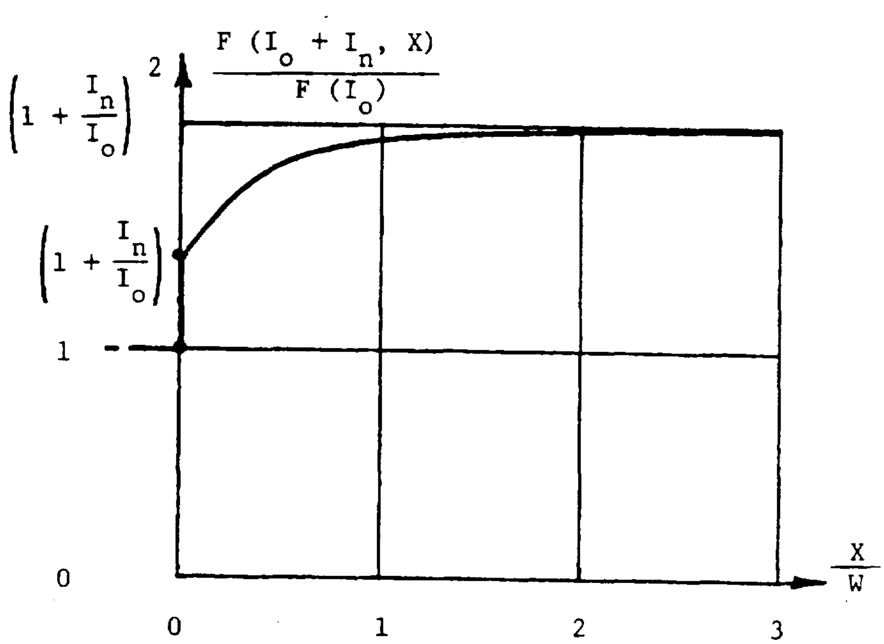
Again, the efficiency assumptions made are shown to be reasonable.

The efficiencies in the two cases are comparable at the same velocities. The efficiencies in Case B-1 are slightly lower than those in Case A-2 because armature volt drop does not vary with current, and because the current is lower. If armature volt drop behaved as a true resistance, i.e., was proportional to current, then the efficiencies in the two cases would be more nearly equal.

3.2.4.2.5 Discussion of Simulations

In the simulations, it is assumed that the current from each energy store into the gun rises instantly from zero to full value as the armature enters the stage in question, as indicated in the top diagram of Figure 3-31. It was also assumed that the accelerating force on the projectile rises to the full value instantly. In fact, what would happen is as follows. The J x B force on the armature is just that. At the moment of switch-on, the current is increased by the amount being supplied by the newly connected energy store, but the magnetic field with which the armature current interacts is not changed. Thus the force increases by a first power law, not a square law, as indicated in the bottom diagram of Figure 3-31. This diagram also shows that after the projectile has travelled one rail width the driving force has risen to just about full I squared value. The effect of this will be to mitigate to some extent the spikey nature of the driving force that would be expected from current wave form given by the simulation. It is likely that other "lagging" effects during switching would further smooth the driving force. It may be possible to use some kind of cushioning of the driving force as indicated in Figure 3-32.





Ratio of propulsive force as a function of projectile travel for a step increase in gun current

FIGURE 3-31. FORCE PICK-UP MECHANISM

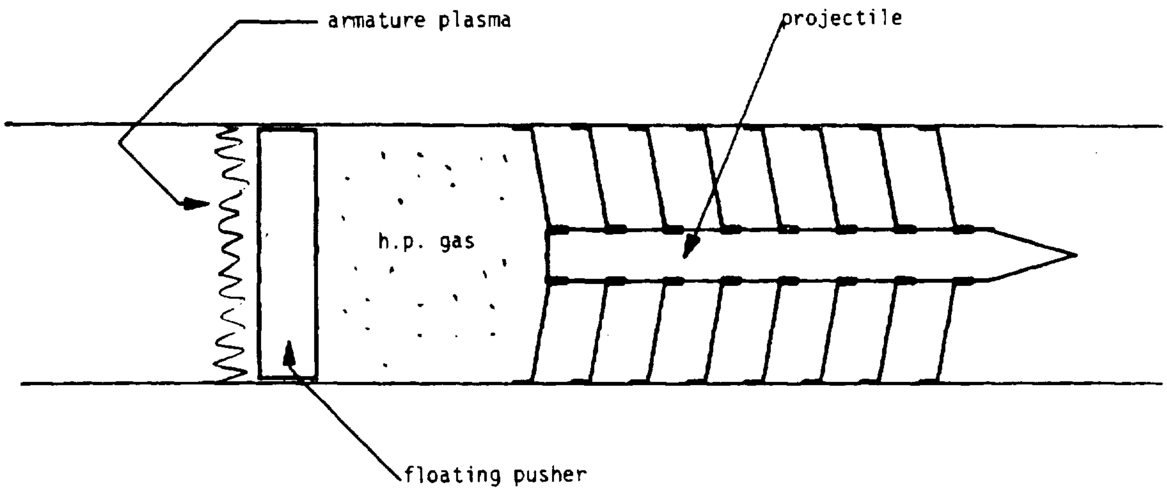


FIGURE 3-32. DRIVING FORCE CUSHION CONCEPT

The most important aspect of the simulations is obtaining of figures for the efficiency at which energy can be transferred from the inductors to the launcher. It is believed that the values obtained are realistic. The nature of the force pick-up mechanism has little effect on the simulations. The small lag in the force pick-up will also in real life cause a similar lag in reaching full back voltage, so that the energy transfer will in fact be little affected.

3.2.4.3 Switching Issues

It is of paramount importance in any accelerator in which a sequence of energy input devices are used that the energy stores be brought into action, be turned on, at just the right moment. The obvious way to do this is to detect the arrival of the projectile at appropriate points along the launcher and to have this cause a switching sequence to occur. With the ESRL launcher a more direct method is possible. In a sense the projectile rides along the launcher on a magnetic wave, and it may well be feasible to have the front of this wave actually do the switching (Marshall, 1976). In this way, automatic synchronization of switching with projectile position would be achieved. The principle is illustrated in Figure 3-33. The idea is that an arc is drawn above and below the launcher rails in spaces connected to the launcher bore. The timing of the drawing of this arc is not critical. It just has to be done some short time before the armature passes. It may be necessary to take positive steps to ensure that the arcs remain stationary before the armature arrives, by having conductors (shown as "X") nearby carrying current in the reverse direction. Such reverse current carrying conductors are required in any event as is shown below.

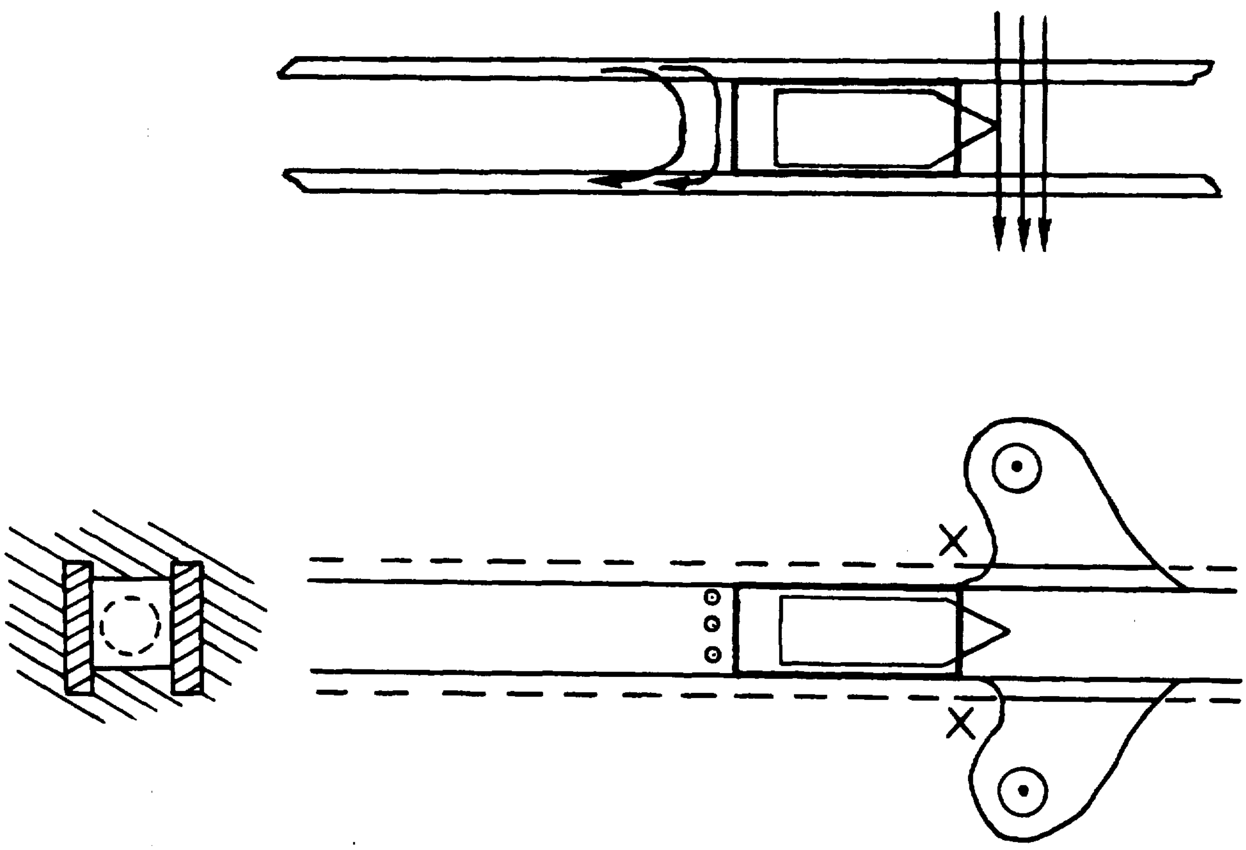
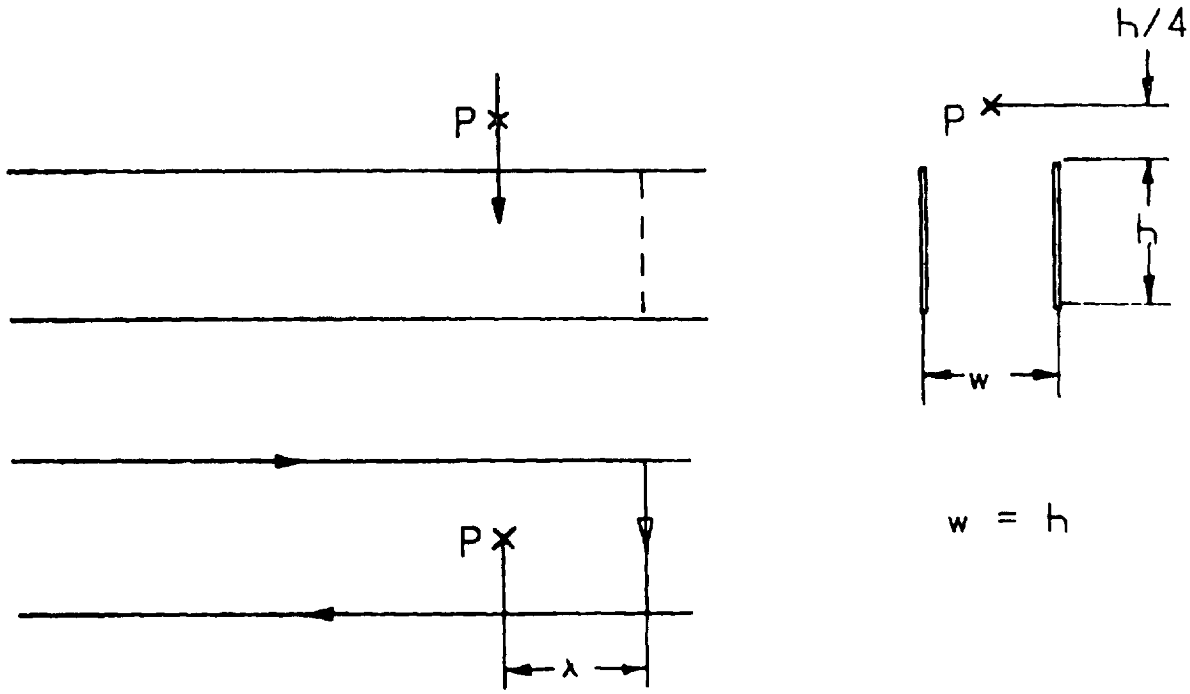


FIGURE 3-33. THE PRINCIPLE OF AUTOMATIC SYNCHRONOUS SWITCHING OF ENERGY STORES IN A DES RAILGUN LAUNCHER

The shape of the magnetic wave which accompanies the armature along the launcher is surprisingly sharp, as can be seen in Figure 3-34. The vertical component of the magnetic field at a point P which is on the centerline of the launcher and situated one quarter rail height above the top edges of the rails. The current assumed is three uniform current sheets, the two rails coming from the left of the armature, and the armature which is a distance x in front of P. As can be seen from Figure 3-34, the field changes from slightly backwards (i.e., the direction that causes a rearward force on a current at P which flows in the same direction as that in the current sheet) to full forward for an armature travel of a little less than half the launcher bore, a favorable state of affairs for automatic synchronization.



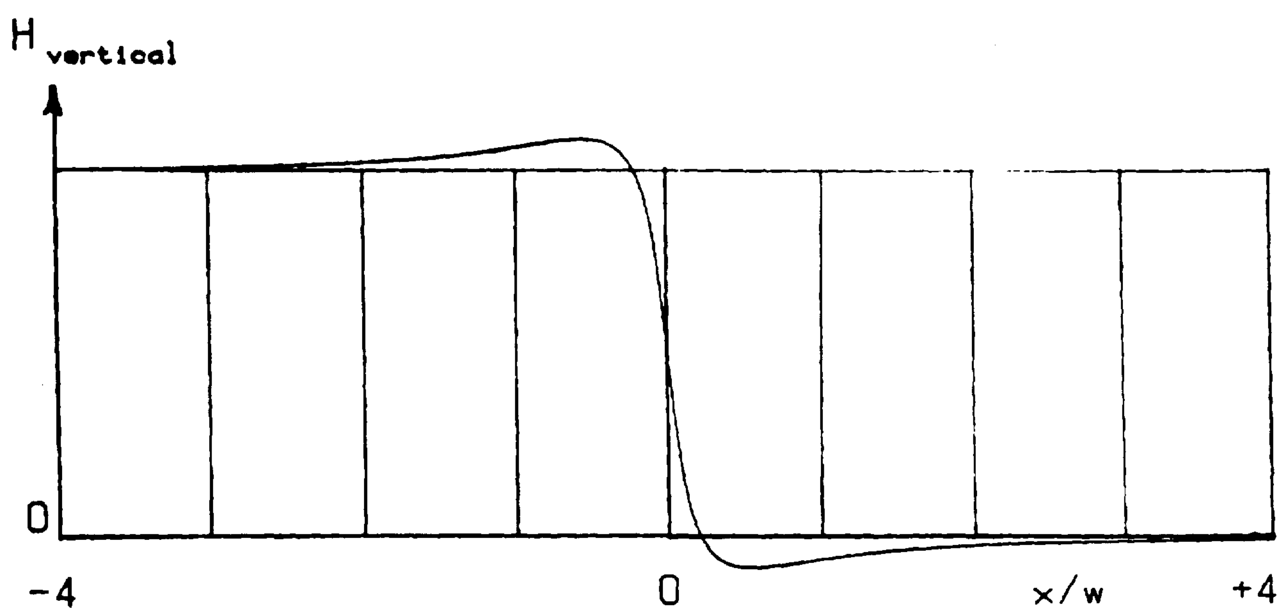


FIGURE 3-34. SHAPE OF THE VERTICAL COMPONENT OF MAGNETIC FIELD OF RAIL-LAUNCHER, AT LOCATION P, NEAR THE ARMATURE

The switching sequence required to connect an energy store to the launcher, and to disconnect it when it is discharged, are as follows. With the homopolar generator (HPG) fully charged (i.e., rotating at full speed), energy is transferred to the inductor (see Figure 3-35) by closing the bypass contacts. It will probably be desirable to have two of these, one above and one below the rails. This transfer is made before the projectile arrives. Then the bypass contacts are opened, drawing arcs between their contacts, just before the projectile passes. As the armature passes, the magnetic wave then sweeps the arc along to join the armature plasma, the current from the inductor then flowing from rail to rail. As the projectile continues on down the launcher the bypass contacts would continue to move apart to reduce the likelihood of restrike.

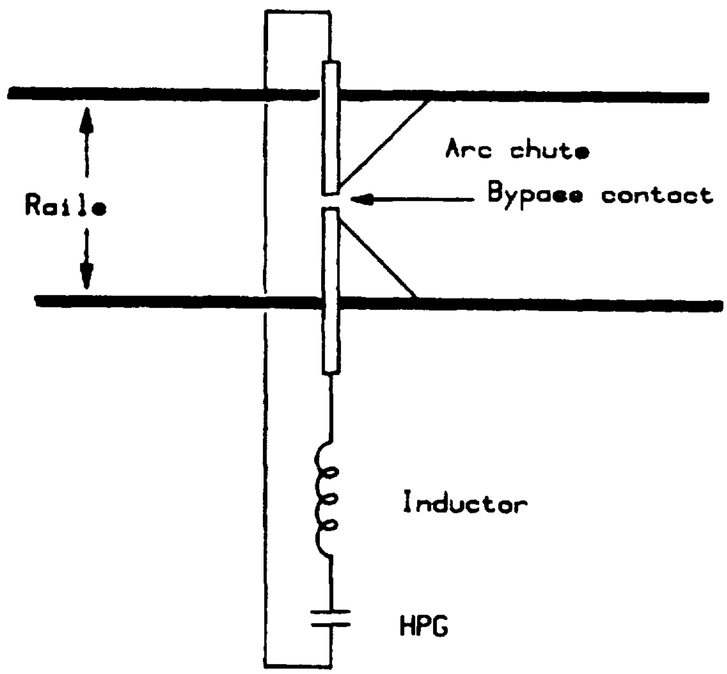


FIGURE 3-35. THE ELEMENTS OF AN ENERGY STORE AND SWITCH

There are two other requirements. The energy store system must be electrically isolated from the rest of the launcher while energy is being transferred from HPG to inductor. The reason for this is that (for continuous rails) the rail to rail voltage ahead of the projectile is equal to the armature volt drop, i.e., several thousand volts. This voltage is a lot higher than the voltage of a fully charged HPG and its presence would prevent the energy store from operating.

The second extra requirement is that when discharged, the energy store system must be electrically disconnected from the launcher to prevent reverse flow of energy. A method of doing this is discussed below.

The two extra switching requirements can be met by using chevron-shaped segments to make up the whole rails, as shown in Figure 3-36, the shaded area being one such chevron. Each chevron would be electrically insulated from its neighbors. The circles indicate the positions of the bypass contacts, and the dashed lines show the position of the launcher top and bottom (insulating) boundaries.

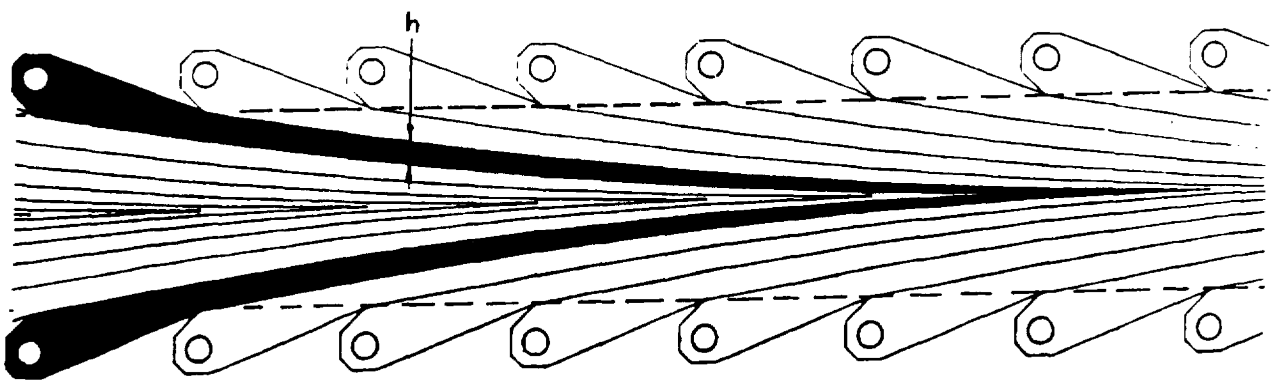


FIGURE 3-36. ESRL RAIL CONSTRUCTION

The two requirements are met by the chevrons being insulated from their neighbors. The energy store connected to any chevron knows nothing of what is going on in the launcher until the armature reaches the chevron, and after the armature has left its tip.

The chevron would be shaped in such a way that the height h is proportional to the current being delivered by that energy store when the armature is in that position. This would be workable because the thickness of the plasma in the direction of its motion is small compared with the bore of the launcher. Done correctly the flow pattern of electric current in the armature would be the same as if the rails were continuous sheets, and the launcher's, performance would be unaffected.

3.2.5 ESRL Technical Uncertainty

The greatest technical uncertainty in the ESRL concept is whether energy store switches for the duty described above can be made to work. There is every reason to believe that such switches will be practicable in one form or another, but more detailed study is needed, particularly concerning the way in which all components of the magnetic fields might affect the switch arc motion.

Because of technical uncertainty associated with the armature, a more detailed investigation of the stability of the large plasma armature is required.

3.3 Single Energy Store Railguns (IAP Research)

Dr. John Barber of IAP Research, Inc., Dayton, Ohio, was subcontracted by Battelle to provide information on railgun technology as well as some point designs for a few ESRL concepts. The basic information that Dr. Barber presented at the 12-13 August 1981 ESRL Concept Definition meeting is contained in Appendix D and is not repeated further here. Dr. Barber was asked to conceptualize some point design single energy store launcher systems. The results of this activity is reviewed in the next two sections.

3.3.1 Summary

The results of single energy store railgun analysis are summarized in Table 3-5 and described in the following paragraphs. The basic requirements assumed for several cases of interest are those listed in Table 3-3 (see Section 3.2). The projectile mass and diameter for each case is indicated in Table 3-3. For the purposes of this investigation the projectile cross-section indicated in Figure 3-37 was assumed.

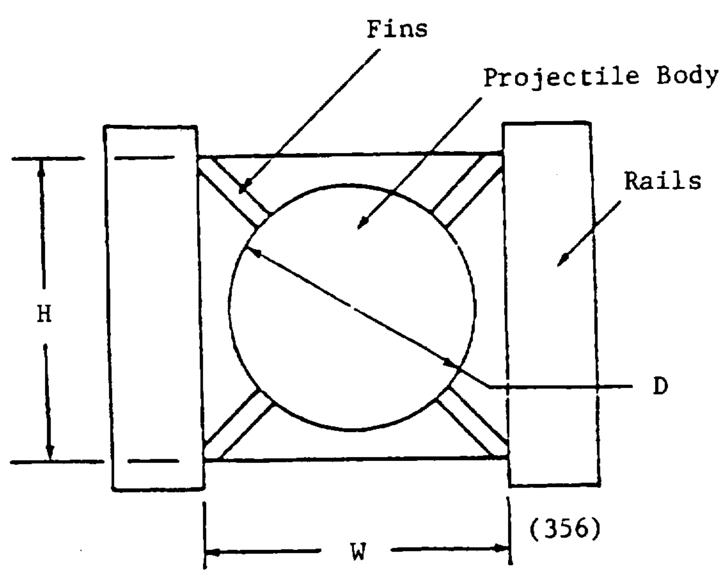


FIGURE 3-37. PROJECTILE/BORE CONFIGURATION

The projectile base stress was computed from the projectile mass, m, and the projectile base area. The acceleration, a, was taken to be 10,000 g's for the solar system escape mission (Mission A) and 2500 g's for the Earth orbital mission (Mission B). The base stress, σ_b, is given by:

$$\sigma_b = 4ma/\pi D^2 \tag{1}$$

The base stress indicated is extremely high for all but Cases A-2 and B-1.

TABLE 3-5. PARAMETER SUMMARY FOR SINGLE STAGE ESRL

Parameter/Case No.	A-1	A-2	A-3	A-4	A-5	A-6	B-1
Base Stress, N/cm2	274,000	86,500	848,000	266,000	848,000	266,000	68,500
Current, MA	51	29	29	16	16	9	25
Bore Height, cm	100	58	58	32	32	18	55
Kinetic Energy, GJ	1,300	411	411	130	130	41	81-325
Launcher Length, m	3,000	3,000	3,000	3,000	3,000	3,000	750-3,000
Stored Energy, GJ	5,200	1,600	1,600	520	520	160	324-1,300
Peak Power, GWe	26,000	8,400	8,400	2,600	2,600	800	1,600-3,100

The acceleration current can be readily calculated from the simple formula:

$$I = (2ma/L')^{1/2} \qquad (2)$$

where L' is the inductance per unit length of the launcher. The values shown in Table 3-5 assume that $L' = 0.5$ μH/m.

The bore height is determined from the bore stress, σ_r, which may be different from the base stress because of the sabot. The calculation follows from,

$$\sigma_r = L'I^2/2wh \qquad (3)$$

where w and h are the width and height of the bore respectively. Assuming that $w = h$ and $L' = 0.5$ μH/m, the following equation is obtained:

$$h = (L'/2\,\sigma_r)^{1/2} I \qquad (4)$$

where I is taken from Equation (2). The values shown in Table 3-5 were computed using σ_r equal to the highest yield strength quoted for copper-zirconium alloy (AMZIRC) of 41,400 N/cm^2 (60,000 psi). As expected, the bore is larger than the projectile diameter for all but the lowest base stress cases (Cases 4-2 and B-1). The next column in Table 3-5 shows the launch package kinetic energy. The power supply/energy store must supply at least this amount of energy to the accelerator to obtain the required velocity.

The launcher length, is related to the muzzle velocity and the acceleration. The minimum launcher length, x_o, will be obtained when the maximum acceleration, a_m, is maintained throughout launch. The minimum launcher length is given by:

$$x_o = v^2/2a_m \qquad (5)$$

In a single energy store system, a tradeoff must be made between launcher length and energy storage requirements. An infinite amount of energy must be stored to achieve the desired velocity in the minimum launcher length. Minimum energy storage requirements are obtained with an infinitely long launcher. IAP simulations indicate that the ratio of stored energy to kinetic energy is related to the ratio of launcher length to minimum launcher length approximately as indicated in Table 3-6. The results shown in Table 3-5 were computed assuming that $x/x_o \approx 4.0$. These are approximations and should be treated with some caution. In a more detailed analysis a quantitative tradeoff would be made between launcher length and energy storage requirements.

TABLE 3-6. ENERGY STORAGE REQUIREMENTS FOR SINGLE STAGE ESRL

$E_{store}/E_{kinetic}$	x/x_o
∞	1.0
8	1.2
4	1.5
2.5	2.0
2.0	∞

The final row in Table 3-5 shows the peak power required during acceleration. This power was derived from the back emf developed by the ESRL rails and is given by:

$$P = L'vI^2 \tag{6}$$

This power will, of course, also have to be supplied in a distributed store (see Section 3.2) or segmented system (see Section 3.4). The required peak power is enormous, clearly indicating the attractiveness of energy storage (rather than direct conversion). The peak voltage (back emf) developed by the accelerator varies from about 90 kV (Case B-1) to over 500 kV (Case A-1). These high voltages pose severe problems for switching of distributed or segmented guns especially near the muzzle.

Some crude estimates on size and mass of major components have been estimated. A HPG might store 0.1 GJ/m^3. At a density of 7800 kg/m^3 this equates to approximately 10 kJ/kg. To store the required energy in homopolars, a total mass ranging from 16,000 MT (for Case A-6) to 520,000 MT (for Case A-1) would be required. The corresponding volume ranges from 2000 m^3 to 67,000 m^3. Inductive energy storage can have an energy density comparable to that of homopolar generators at 0.1 GJ/m^3 (higher energy density can be achieved at high stresses). However, the mass of an inductor can be very low (dictated by resistive losses and stress) with an effective "density" of perhaps 1000 kg/m^3. The mass of inductive energy storage would, therefore, vary from about 1,600 MT (Case A-6) to 52,000 MT (Case A-1). The volume occupied would be similar to homopolar generators. The launcher tube might weigh from a few thousand MT to a few tens of thousands of MT depending on the bore size.

The most mass and volume efficient method of storing energy is chemical storage. In a single stage ESRL, the acceleration time is long enough that such a scheme might be possible (e.g., pulsed MHD). This would greatly reduce the power system size and difficulty.

3.3.2 Conclusions

The single energy store system is large, and unavoidably so, given the energy requirements imposed by the projectile and mission. No major technological impediment exists to developing a large single-stage accelerator. The energy stores would have to be large, however, no new technological concepts are required. Switching is relatively straightforward for a single-stage railgun, as the difficult switching tasks are all done at the beginning of acceleration where they are the easiest to do. The current levels are high, but not so far beyond existing experience that they cannot be contemplated with some equanimity. Launcher losses, and subsequent cooling, will probably limit firing rates, but no difficulty is anticipated in obtaining a few shots per day.

3.4 Segmented Energy Store Railguns (LLNL)

Mr. Ron Hawke of Lawrence Livermore National Laboratory (LLNL) was subcontracted by Battelle to provide information on railgun technology as well as some point designs for a few ESRL concepts. The basic information that Mr. Hawke presented at the 12-13 August, 1981 ESRL Concept Definition meeting is contained in Appendix E and is not repeated further here. Mr. Hawke was asked to conceptualize several point designs for segmented energy store rail launcher systems. The results of this activity is reviewed in the sections that follow.

3.4.1 Summary

The results of the segmented energy store railgun analysis are summarized in Table 3-7, and described in the following paragraphs. The basic requirements assumed for several cases of interest are those listed in Table 3-3 (see Section 3.2).

TABLE 3-7. PARAMETER SUMMARY FOR SEGMENTED STAGE ESRL

Parameter/Case No.	A-1	A-2	A-3	A-4	A-5	A-6	B-1
Initial Kinetic Energy, GJ	3.25	1.03	1.03	0.3725	0.325	0.103	3.25
Launched Kinetic Energy, GJ	1300	1.03	411	130	130	41.2	81.3-325
Current, MA	56.4	31.7	31.7	17.8	17.8	10.0	28.2
Rail Height, cm	128	72	72	40	40	23	64
Sabot No. 1 Mass, kg	3338	647	426	111	100	20	254
Sabot No. 2 Mass, kg	1093	279	332	99	101	36	301
Single Stage Efficiency, %	37	31	31	23	23	17	25
Stored Energy Required, Single Stage, GJ	3532	1346	1346	559	559	247	1309
100 Stage Efficiency, %	64	61	61	58	58	53	59
Stored Energy Required 100 Stage, GJ	2710	874	874	289	289	98	722

3.4.2 Details

3.4.2.1 Energy Requirements

The initial kinetic energy provided to the railgun section is the first row in Table 3-7. It was assumed that 1 km/s was required to avoid erosion on the rails. The launch kinetic energy, based upon the projectile mass and the velocity at the exit of the rail launcher is given in the next row of Table 3-7.

3.4.2.2 Current

The equation below defines the force as a function of inductance and current in a railgun:

$$F = L'I^2/2 \qquad (1)$$

Acceleration is given as:

$$a = L'I^2/2m \qquad (2)$$

Therefore, the current is a function of the mass, the acceleration and the inductance as given in Equation 3.

$$I = [2ma/L']^{0.5} \qquad (3)$$

For calculations here, it was assumed that L' is equal to 0.4 μH/m. A value of 0.6 μH/m is theoretically possible for L'. However, a vaue of 0.4 μH/m represents a typical railgun value. Given the value of mass, acceleration, and L', the current required to accelerate the projectile is then calculated. See Table 3-7.

3.4.2.3 Rail Height

Next the rail height is calculated. Table 3-8 provides information on the rail material assumed, AMZIRC (Engineering Alloys Digest, Inc., 1961).

3.4.2.4 Rail Design Limits

Joule heating of the rails is given by the following relationship

$$\frac{I}{h} = \left[\frac{\pi\rho Cv\Delta T}{2\mu o} \frac{1}{\ln\left(1 + \frac{\pi}{2}\sqrt{\frac{Cv\eta_o}{\mu_{ok}}}\right)}\right]^{1/2} \qquad (4)$$

where,

$$\Delta T \simeq \frac{2\mu_o I^2}{\pi\rho Cvh^2} \ln\left[1 + \frac{\pi}{2}\sqrt{\frac{\rho Cv\eta_o}{\mu_o k}}\right] \qquad (5)$$

TABLE 3-8. PROPERTIES OF COPPER AND AMZIRC

Property	Copper	AMZIRC
$\eta(\Omega m)$	1.7 (10^{-8})	183 (10^{-8})
$\alpha(\Omega m/K)$	11.6 (10^{-11})	8.4 (10^{-11})
$\rho(g/cm^3)$	8.92	8.89
C_v(J/kgK)	385	--(a)
k(w/mK)	3.98	343
$\sigma y(N/cm^2)$	30,500 (hard)	42,100 (85% cold worked)
$y(N/cm^2)$	11,700,000	12,900,000
T_m(C)	1,083	--(a)

(a) C_v and T_m for AMZIRC were not available; values for pure copper are probably similar and were used in the calculations.

Based upon the temperature stress data provided in the reference data sheet on AMZIRC (Engineering Alloys Digest, Inc., 1961). A maximum temperature limit in the rails was assumed to be 450 C. Substituting this value and others into the above equations, we arrive at a limit due to Joule heating of rails of the order of 25 MA/m. To calculate the launcher stress, the Lorentz pressure on the rails is given as:

$$P_L = \frac{\mu o I^2}{2\pi h^2}\left[2h \tan^{-1}\left(\frac{h}{w}\right) - w \ln\left(\frac{w^2 + h^2}{w^2}\right)\right] \tag{6}$$

For a square bore where the w equals h, this equation reduces to:

$$P_L = 0.44 \frac{\mu_o I^2}{\pi h^2} \tag{7}$$

Based upon the assumption that the rail is heated to 450 C and the maximum allowable stress is reduced by roughly 70 percent of the ambient temperature value of stress, the maximum rail stress is given as 33,700 N/cm^2 (49,000 psi). Solving for the value I/h, we arrive at a current density limit of 44 MA/m.

$$\frac{I}{h} = \left[\frac{\pi P_M}{0.44\ m_o}\right]^{1/2} \tag{8}$$

$$\frac{I}{h} = 44\ MA/m$$

When one considers the sabot, one can also arrive at stress limits. For Lexan with a σ_y of a 7600 N/cm^2, we arrive at a current density limit of 19.5 MA/m. For a carbon/carbon filament with a stress limit of 100,000 N/cm^2, the current density is equal to 72 MA/m. Table 3-9 provides a summary of the various stress limits.

TABLE 3-9. ESRL STRESS LIMITS

Type of Limit	I/h (MA/m)
Joule Heating of Rails	25
Rail Stress	44
Sabot Stress ($P = \sigma y$)	19.5 (polycarbonate)
Sabot Stress	72 (carbon/carbon)

The Joule heating calculation is for a step function rise in the current, where the projectile is moving very fast and the arc is infinitely thin. Let the plasma length be approximately 1 meter (a reasonable assumption for a 1 to 2 meter bore railgun). At 20 km/s the rise time of the current in the rails is approximately 50 μs (σ approximately equal to 0.7 mm). Hence, the actual temperature rise will be less than calculated above and will permit a higher current concentration, especially at lower velocities. Therefore, in the calculations that follow, the limit imposed by rail stress is used, that is, 44 MA/m. The corresponding rail heights as calculated based on this value are given in Table 3-7.

3.4.2.5 Sabot Mass Estimates

The following discussion relates to estimating the mass of sabots. Two sabot concepts are shown in Figure 3-38. Table 3-7 provides the mass characteristics of these two sabot concepts.

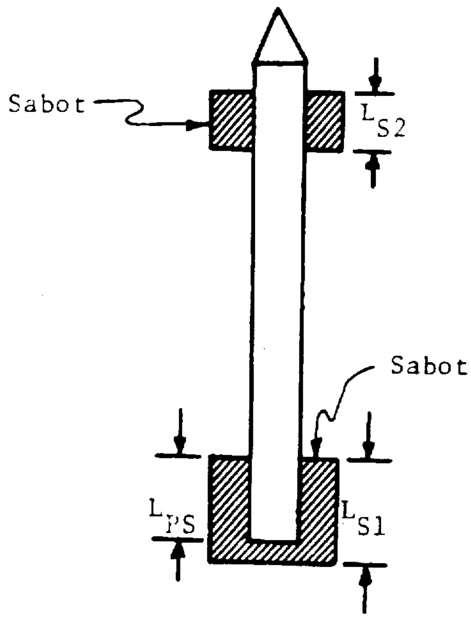
3.4.2.6 Power and Energy Requirements

Power and energy requirements are estimated here. The energy loss in the rails for constant current is given as:

$$E_R = \left(\frac{32}{15hI}\right)\left(\pi\mu_o\eta\right)^{1/2}\left(m/L_1\right)^{3/2}\left(v\right)^{5/2} \quad (9)$$

Inductive energy in the railgun, although in principal, can be recovered, it must be provided to accelerate the projectile. The energy is equal to the kinetic energy (E_{KE}).

$$E_I = 1/2LI^2 = 1/2\, L'\, z\, I^2 = 1/2mv^2 \quad (10)$$



$$m_S = \rho_S L_S \left(hw - \frac{\lambda \pi d_p^2}{4} \right)$$

Where:

$$L_S = L_{S1} + L_{S2}$$

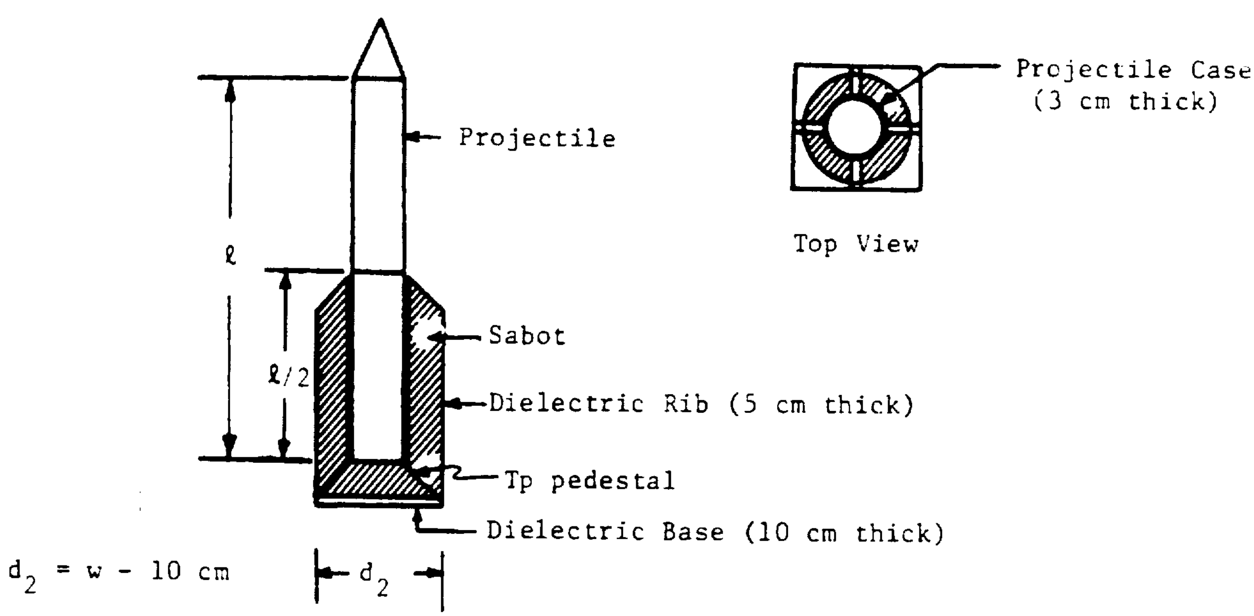
$$\lambda = \frac{L_{PS} + L_{S2}}{L_{S1} + L_{S2}}$$

$$\lambda = 0.8$$

$$\rho_S = 1.8 \text{ MT/m}^3$$

(for carbon/carbon)

a. Sabot Concept #1



b. Sabot Concept #2

FIGURE 3-38. SABOT CONCEPTS

The values of energy loss in the rails and the overall efficiency for a single-stage rail launcher is given in Table 3-10 below.

TABLE 3-10. SINGLE-STAGE RAILGUN EFFICIENCY

Mission Case No.	E_{KE}, GJ	E_I, GJ	E_R, GJ	Overall Efficiency, percent	E, GJ
A-1	1,300	1,300	932	37	3532
A-2	411	411	524	31	1346
A-3	411	411	524	31	1346
A-4	130	130	299	23	559
A-5	130	130	299	23	559
A-6	41	41	165	17	247
B-1	325	325	659	25	1309

For multistage railguns with and without inductive energy recovery, the efficiencies for 1, 10 and 100 stages are shown in Table 3-11. Based upon these data, the overall stored energy required for one stage, 10 stages, and 100 stages, is given in Table 3-12. Note that the energy per stage is very similar for Missions A-2, A-3 and B-1. The largest HPG made to date was of the order of 500 MJ. It is reasonable to assume that a 1-2 GJ HPG is within the state of the art and that a 5 GJ HPG or equivalent cluster of small HPG's could be developed and operated in the next 10 years. An assumption was made that a 5 GJ HPG module would be available and that a 4.5 GJ of energy could be transferred to a storage inductor and 4.0 GJ of energy transferred into a railgun stage. The number of stages needed is listed in Table 3-13. Assuming a 1.2 GW_e output power plant is used to energize the homopolar generators at an 80 percent efficiency, the times to charge the homopolars are given in Table 3-13 for the various missions. Table 3-14 summarizes the components for Missions A-2, A-3, and B-1.

Figure 3-39 shows the layout of major components in a single, HPG storage inductor-shuttle switch railgun stage. The primary power source motors up the homopolar generators. Switch S_1 is closed on each segment. At peak current, Switch S_2 is closed on each segment. Switch S_3 is

TABLE 3-11. MULTISTAGE EFFICIENCY WITH AND WITHOUT INDUCTIVE ENERGY RECOVERY

Mission Case No.	Number of Stages 1	10	100
With Inductive Energy Recovery			
A-1	37[a]	45	48
A-2	31	42	47
A-3	31	42	47
A-4	23	37	45
A-5	23	37	45
A-6	17	31	42
B-1	25	38	45
With 50 % Inductive Energy Recovery			
A-1	45[a]	58	64
A-2	36	53	61
A-3	36	53	61
A-4	26	45	58
A-5	26	45	58
A-6	18	36	53
B-1	28	47	59

[a] In percent.

TABLE 3-12. ENERGY STORAGE REQUIREMENTS FOR MULTISTAGE RAIL LAUNCHERS

Mission Case No.	Number of Stages 1	10	100
A-1	3,532[a]	2,890	2,710
A-2	1,346	980	874
A-3	1,346	980	874
A-4	559	350	289
A-5	559	350	289
A-6	247	130	98
B-1	1,309	855	722

[a] In GJ.

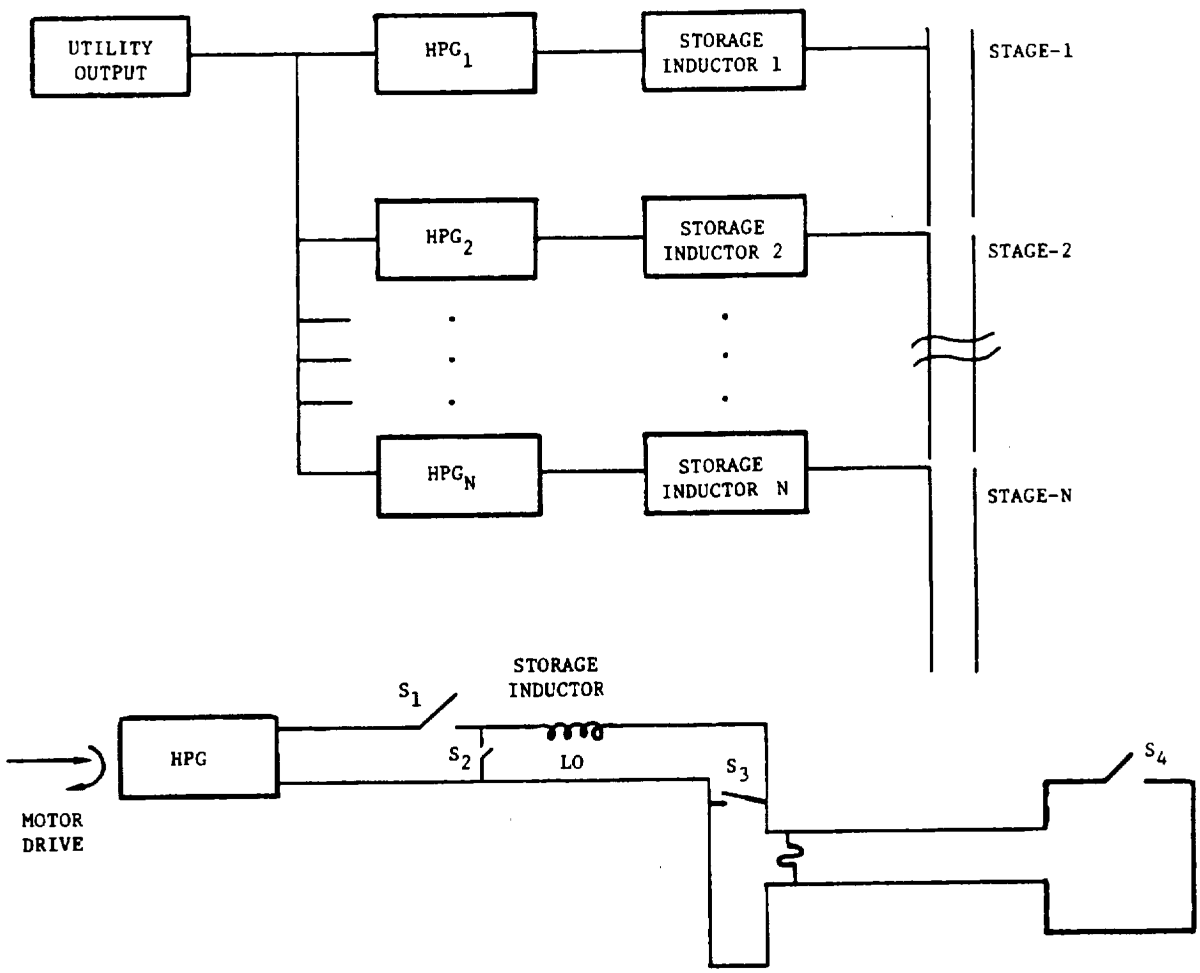


FIGURE 3-39. DIAGRAM OF CIRCUIT AND SWITCHING CONCEPT

TABLE 3-13. STAGE LENGTH, NUMBER AND TIME TO CHARGE STORES FOR 1.2 GW_e CHARGE RATE

Mission Case No.	Number of Stages[(a)]	Stage length, m	Charge Time, min
A-1	640	3.1	43
A-2	205	9.8	13.7
A-3	205	9.8	13.7
A-4	74	27	4.9
A-5	74	27	4.9
A-6	29	69	1.9
B-1	170	12	11.3

[(a)]Based on a 5 GJ HPG module per stage and 80 percent HPG charge up efficiency.

TABLE 3-14. CHARACTERISTICS OF ESRL SYSTEM

System Parameter	Missions A-2 and A-3	Mission B-1
Launcher		
Current, MA	31.7	10
Bore, cm	72	23
Number of Stages	205	170
Storage Inductors		
Inductance, H	9.0	90
Stored Energy, GJ	4.5	4.5
Homopolar Generators		
Output Voltage	200	200
Effective Capacitance	250	250
Stored Energy, GJ	5	5

sequentially triggered to shuttle across the side feed, fuses the breech portion of each rail stage, and has the projectile passes the input of each stage. As the projectile exits each stage, Switch S_4 is closed to extinguish the arc. The remaining energy in the rails and storage inductor would be used to motor up the homopolar generator for the next launch by closing Switch S_1 and opening Switch S_2 (with a shuttle switch).

The rail cross-sectional dimensions, structure dimensions and materials are illustrated in Figure 3-40. The intersection of the rails and dielectric should be at a 45 degree angle (see Figure 3-40). The back side of the rail should be circular and the rail thickness should be at minimum 3 cm at the corners. The interior dielectric might be ceramic (aluminum oxide, boron nitride, or titanium oxide), polymide (which has worked well on tests), or perhaps Delrin or G-10/11. The hoop dielectric provides support to the rails and interior dielectric and spacing between the outer casing and the rails. The minimum thickness of the hoop, Δr, is given as

$$\Delta r/w = P_b/2P_h$$

where P_B is the bore pressure resulting from the Lorentz forces on the rails and the hydrodynamic forces from the plasma and P_h is the allowable stress in the hoop. If the hoop is made of carbon/carbon or Kevlar filament and has a strength of 100,000 N/cm^2, then,

$$\Delta r/w = 0.19 \quad .$$

The values of bore height, where bore height equals the width in the opposite direction, are listed in Table 3-7 for all seven mission candidates. There is no need for a metal case for a hoop stress point of view, however, a case may be needed to provide stiffness and hoop positioning. An aluminum case should be adequate and could serve as a vacuum vessel. The dielectric hoops could be short cylinders which are slipped over the interior dielectric and rails. The interior dielectric and rails could be short sections which are fastened together. The whole structure could be disassembled for maintenance.

3.4.3 Conclusions

Mr. Ron Hawke concluded that the ESRL concept would be a very large system, but that it appears feasible. The multistage approach has definite advantages over the single-stage launcher because one can size energy stores into smaller reasonably sized modules (that exist or will soon exist) and distribute the current over the rail length. There are energy storage problems with the single stage launcher approach--stores must be large and/or concentrated at the breach of the launcher. Better efficiency is available because of the higher level of current that can be maintained along the launcher. Also, the launcher is shorter for distributed energy systems. Another advantage of the multistage system is that it is possible that even with an injection one may not have sufficient velocity for injection (preboost) into the rail acceleration portion of the launcher to avoid all rail damage or dielectric damage. One may desire to stay below a critical current threshold in the early phase of the launch. However, as the projectile moves, higher currents can be applied. Therefore, one can tailor the current pulse to allow the launcher to have its maximum efficiency. The multistage device requires multistage switching and it can be automatic with the projectile. However, there must always be an open switch some place to use inductance energy storage, and there is a technology that is coming close to maturity where one can use moving plasma arcs to provide the opening plasma switch.

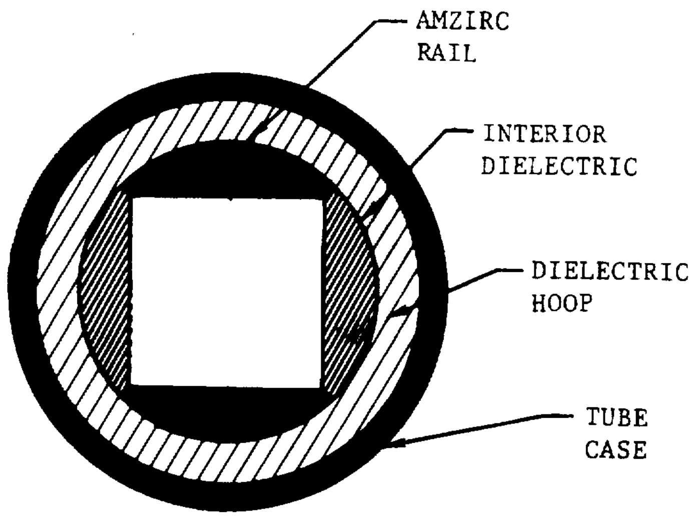


FIGURE 3-40. CROSS-SECTION OF RAILS, SPACERS, AND HOOP

3.5 Sabot/Projectile Considerations (H. F. Swift, PAI)

A brief sabot/projectile analysis was conducted was conducted by Mr. H. F. Swift, Physics Applications, Inc. (PAI) of Dayton, Ohio. His work was supported by NASA/LeRC for the purpose of providing technical information to this study. The material, as presented at the 12-13 August 1981 ESRL Concept Definition Meeting held in Columbus, Ohio, is included as Appendix F. A brief summary of important aspects, along with conclusions and recommendations provided by Mr. Swift are given below.

3.5.1 Summary

A brief sabot/projectile analysis was performed for possible ESRL configurations. Topics that were addressed and are discussed here are:

(1) Drag coefficients and critical mass considerations

(2) Sabot concepts

(3) Aerodynamic heating and ablation

(4) Assessment of launch through non-ideal atmospheric conditions

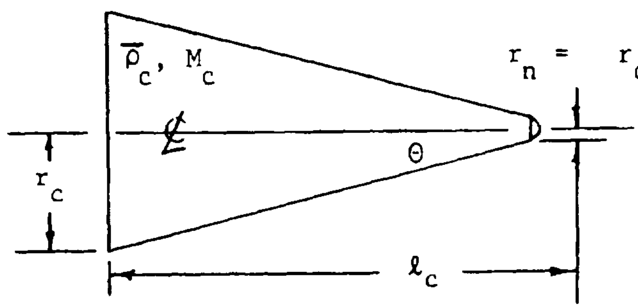
(5) A sample point design.

3.5.1.1 Drag Coefficient and Critical Mass Concentrations

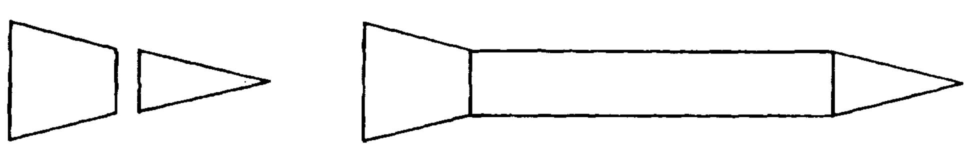
The drag coefficient of a projectile, C_D, represents one of the most important parameters in the accomplishment of the ESRL mission. The drag coefficient is made up of many components. These include Newtonian pressure drag, base drag, and skin friction drag. Mr. Swift's assessment indicates that for hypersonic flight through dense atmosphere, the base drag is much less than the pressure drag and that the skin drag is negligible. For spherically blunted cones and cone darts, as shown below in Figure 3-41, the drag coefficient, C_D, is 1.83 $\text{Sin}^2\theta$. (The drag coefficient for a 15 degree half cone angle is 0.12.) Based upon the relationship of velocity loss due to drag for flight in the atmosphere, relationships were developed that relate the cone angle to the total mass of the projectile for a given critical ballistic coefficient. Plots of the critical mass of a projectile as a function of the half cone angle are given in Appendix F for both cones and cone darts. The results of these calculations indicated that there is a wide range of cone angles and masses that are possible for this mission.

3.5.1.2 Sabot Concepts

Another area that Mr. Swift discussed related to the projectile/ sabot survivability during launch. His assessment was conducted by using the assumption (initially provided by Battelle) of a 30,000 g acceleration limit, however, this has now been changed to 10,000 g's. He investigated the



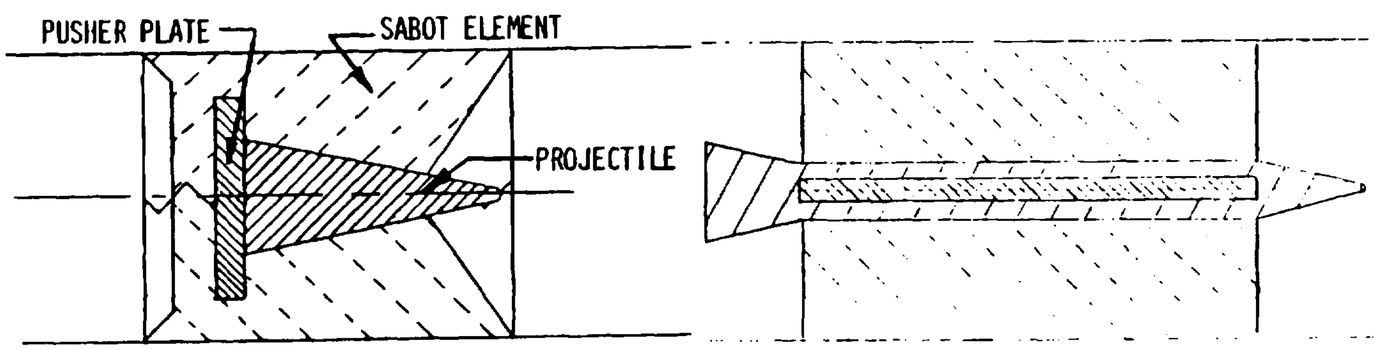
a. SPHERICALLY BLUNTED CONE



b. CONE DART

FIGURE 3-41. SPHERICALLY BLUNTED CONES AND CONE DARTS FOR ESRL PROJECTILE SHAPES

stresses in the sabot and in the projectile for various types of sabots. The two types of saboting concepts are shown in Figure 3-42. These are the base-loading sabots and the side-loading sabots. Information that was provided during this presentation was considered in the formulation of the current sabot/projectile reference concept. Mr. Swift recommended that the acceleration limit be reduced to below the 30,000 g limit to aid in the saboting feasibility. (This recommendation was followed).



a. Base-Loading Sabot b. Side-Loading Sabot

FIGURE 3-42. SWIFT'S ESRL SABOT CONCEPTS

3.5.1.3 Aerodynamic Heating and Ablation

Another topic that was discussed was aerodynamic heating and ablation. Based on his preliminary evaluation, the total heat input is considerably less than what one would expect in a typical ballistic reentry from low Earth orbit. The difference is related to the time that the heating is available to the payload. For decaying type reentries, a much longer time is required for the object to impact the surface (hundreds of seconds through the atmosphere). However, for this particular rail launch application, the time is extremely short (a few seconds). Mr. Swift calculated some values of ablation for Novalak. He estimated that at the stagnation point, less than 1 cm of material would be ablated away for a vertical 20 km/s launch velocity. An additional calculation also supported justification for a high-strength nose cone material because of the relatively high compressive force predicted at the stagnation point.

3.5.1.4 Assessment of Launch through Non-ideal Atmospheric Conditions

The assessment of launches through non-ideal atmospheric considerations was also conducted. The preliminary assessment indicated that wind, and even very high winds, would not affect the launch of an Earth-to-space projectile. For non-precipitating clouds the problem appeared to be substantial. Initial estimates indicated that approximately 80 cm of nose recession were possible. This means that launches should not occur for space disposal missions (20 km/s) through clouds. Additional work is needed to verify this assessment. It is also possible to choose a tougher nose cone material, which would not oxidize or be easily eroded. For precipating clouds, launches should not be made.

3.5.1.5 A Sample Point Design

Mr. Swift also accomplished a point design calculation. He assumed a cermet payload of 842 kg with a radiation shield mass of the order of 5300 kg. He also assumed a cone dart configuration with a half cone angle of 13.2°. The total projectile mass was on the order of 9 MT. Using a side-loading sabot, he estimated that a 35 MT sabot would be necessary. This concept is shown in Figure 3-43.

3.5.2 Conclusions

Based upon the preliminary sabot/projectile analysis done by Mr. Hal Swift of PAI, he has drawn the following conclusions.

(1) The projectile must have a ballistic coefficient of the order of 1×10^5 kg/m^2 to properly fly out of the atmosphere with minimal velocity losses due to drag

(2) The projectile shape should utilize a spherically blunted cone or a cone dart type configuration

(3) Aerodynamic drag is almost exclusively Newtonian pressure drag

(4) Projectiles must be saboted during launch

(5) Base loading sabots are possible, but compressive stress limits the size of the projectile

(6) Side-loading sabots have much wider stress limitations and easily accommodate cone dart type projectiles

(7) Aerodynamic heating is extremely intense, but the total heat input is less than the typical orbital reentry.

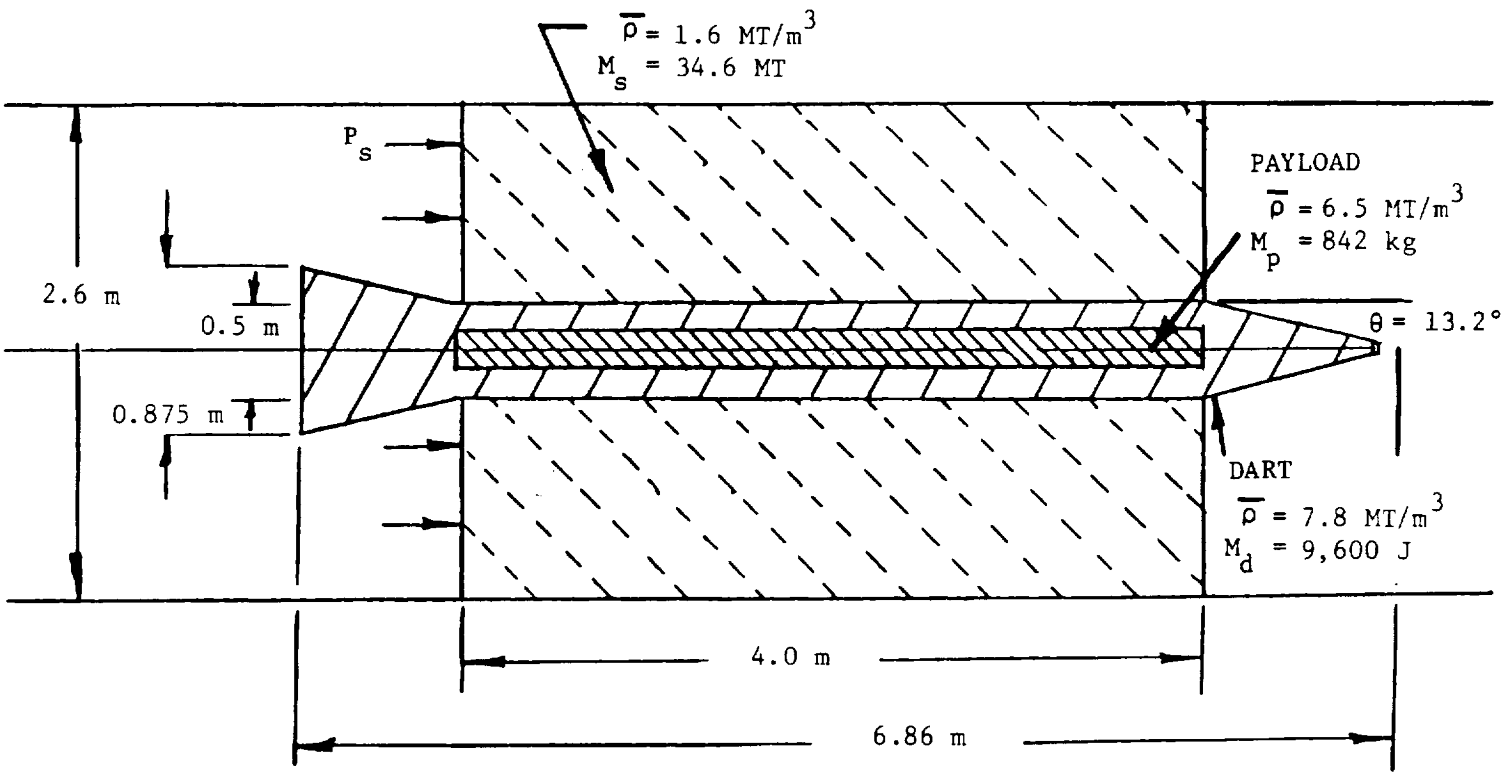


FIGURE 3-43. SWIFT'S ESRL PROJECTILE/SABOT CONCEPT

(8) Stagnation point recessions for fine quality ablators are of the order of 1 cm

(9) Compressive and sheer stresses from aerodynamic forces will limit the ablator choices

(10) Erosion from particulates and light clouds is devastating to high-performance ablators

(11) No fundamental objections to the Earth-to-Space Rail Launcher concept have been found.

3.5.3 Recommendations

Mr. Swift recommended several areas where additional work was needed. The area of aerothermal analysis and ablation performance need significant additional analysis. Refractory metals and oxidation-resistant alloys need to be considered for the nose cone. Stresses that are expected in practical sabot designs will have to be analyzed, most likely with some NASTRAN-type 2D or 3D finite element calculation. Finally, additional work is needed in the area of predicting erosion of high velocity projectiles flying through clouds.

3.6 Projectile/Aerodynamic Heating (LLNL)

Dr. Al Buckingham of Lawrence Livermore National Laboratory was contracted by NASA/Lewis Research Center to provide suggestions relating to projectile design at the ESRL Reference Concept Definition meeting on 12-13 August 1981. Some of his thoughts are summarized in the following section and the vugraphs that he provided at the August meeting are contained in Appendix G.

3.6.1 Summary

Dr. Buckingham discussed many issues and ideas that relate to Earth-to-space launch projectiles; a number of them are discussed in this section. Topics that were discussed are:

(1) Drag coefficient
(2) Concept for allowing plasma to move forward of sabot
(3) Concept suggestions
(4) Aerodynamic heating.

Dr. Buckingham expressed concern that even though the contributions of types of drag, other than the pure Newtonian drag, are small, they should be included in the total calculation. He indicated that for massive ablator blowing off the projectile (see Figure 3-44 for projectile concept), the drag coefficient can be altered substantially. He also suggested that cone darts were excellent candidates for the ESRL mission.

He suggested a concept for allowing plasma to move through and ahead of the sabot, which would allow distributed stress along the body of the projectile. He suggested that a properly designed gap, of the order of 100 microns, may allow this to occur. More study on this concept is required before it could be considered. Another concept that he had suggested was that the projectile employ a boattail at the rear. This would be a light-weight hollow section on the rear of the projectile. He also suggested that the mass of the projectile should be nose forward to aid in stability.

Dr. Buckingham discussed the problems of aerodynamic heating. Most of the discussion centered on radiation heating and the effect of altitude and the effect of ablation.

He suggested a concept for recovery during an abort mode. The concept basically involves the jettison of the back end of the projectile with explosive bolts. The lower section would look something like an Apollo heat shield. The abort would be signaled by an onboard accelerometer. The basic idea is to alter the characteristics of drag by splitting the body into pieces and shifting the center of gravity so that the rear would actually reenter. Another concept that was suggested was the concept referred to as a wave rider. This particular concept would allow the development of lift so that a piece of hardware could actually fly and land at a few meters per second.

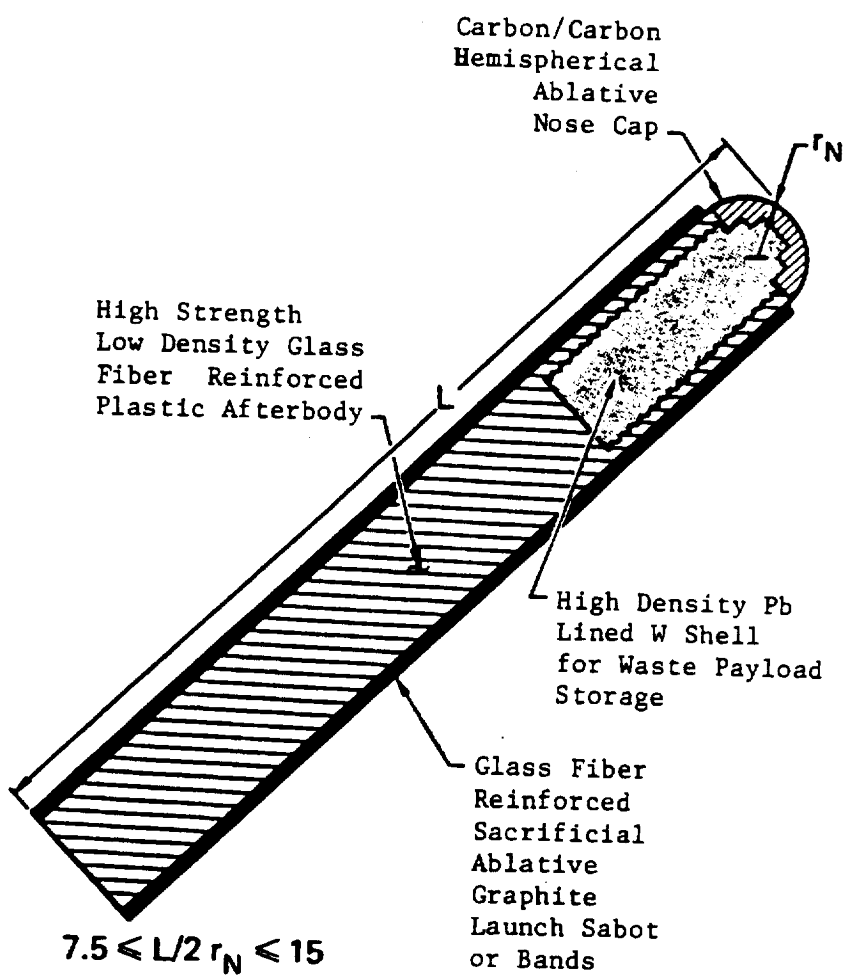


FIGURE 3-44. BUCKINGHAM'S ESRL PROJECTILE CONCEPT

Dr. Buckingham continued to suggest that a 100-200 kg projectile would be a reasonably-sized mass for space applications and disposal.

3.6.2 Recommendations

Dr. Buckingham recommended that testing be started in existing rail gun systems for configurations that include current candidates for ESRL sabots. Work could be done using x-ray radiography to detect and measure ablation and deformation from the sabots themselves. The sabots should be constructed of materials which are candidates for the Earth-to-space rail launcher system.

3.7 Aerodynamics Considerations (The Ohio State University Aeronautical Research Laboratory)

Dr. John Lee of The Ohio State University, Aeronautical and Astronautical Research Laboratory was contracted by NASA/Lewis Reserach Center to provide suggestions on the aerodynamics of ESRL projectiles. Some of his thoughts are summarized in the following section and the vugraphs that he provided at the 12-13 August 1981 ESRL Reference Concept Definition Meeting are contained in Appendix H.

The behavior of a vehicle on a transatmospheric coasting flight may be determined to the first order by means of relatively accurate approximations. The results may be later refined when the results of such an analysis are examined, that is, where the consequence of the approximations may affect the mission. Figure 3-45 illustrates some geometric aspects assumed.

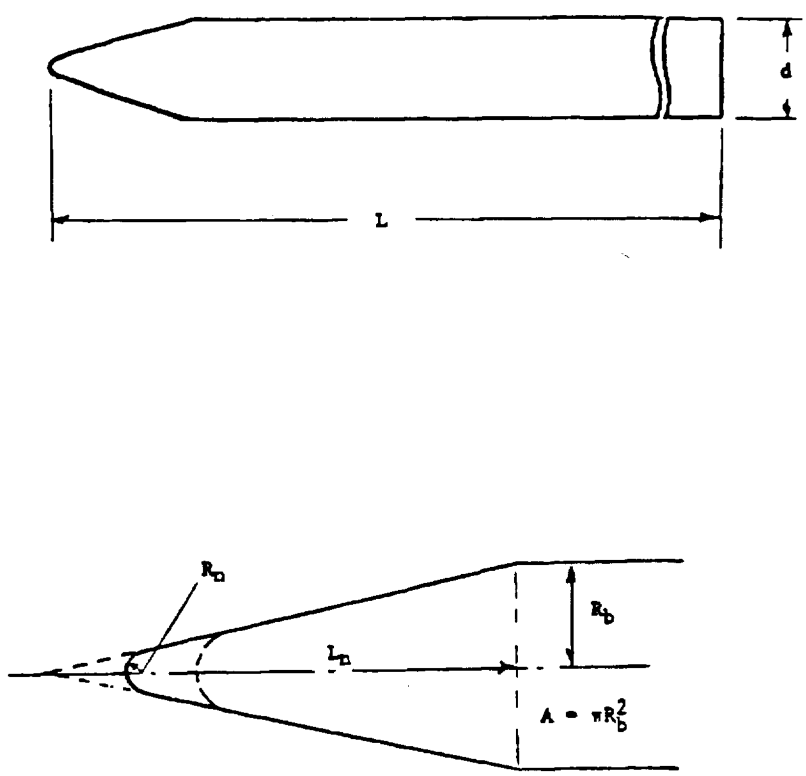


FIGURE 3-45. ESRL PROJECTILE GEOMETRY

The equation of motion,

$$m(dV/dt) - mg - D = 0 \tag{1}$$

may be integrated by assuming: (1) vertical flight; (2) an isothermal atmosphere; (3) a constant drag coefficient; (4) a cylindrical vehicle body; and (5) constant gravitational acceleration.

$$D = C_D\ (\rho/2)v^2A$$

$$\rho = \rho_o\ e^{-h/a}$$

$$m = \rho_m LA$$

where,

C_D = projectile drag coefficient
ρ_o = surface air density
ρ_m projectile mean density
A = vehicle cross-sectional area
L = projectile length
h = altitude
a = atmospheric constant, 6705 m
g = acceleration due to gravity
v = launch velocity.

Equation (1) allows straightforward evaluation of the influences of the vehicle parameters, ρ_m, L, and C_D. The assumption of an isothermal atmosphere has been shown to be sufficiently accurate (Enkenhus, 1959); for this application accuracy may be improved by selecting a value for ρ_o which will weight the match to low altitudes.

The critical aerodynamic item is the drag coefficient; it will be shown that the pressure drag dominates in the most critical phase of the flight (low altitude) but an evaluation of the frictional drag is necessary also for some cases, to obtain an accurate value for the velocity at exit from the atmosphere.

3.7.1 Pressure Drag

The principal pressure drag arises from the nose cap, with minor contributions from the base, surface proturberances, and stabilizing devices.

The base drag is negligible,

$$C_{D_{BASE}} = \frac{\rho_\infty - \rho_\beta}{0.7\ \rho_\infty\ M^2} \approx 10^{-4}$$

(assuming the base pressure ρ_β = 0 absolute).

The nose cap C_D is in the range of 0.05 for a blunted 10-degree cone (half-angle) to 1 for a hemispherical cap (Cox and Crabtree, 1965; Hoerner, 1965; Enkenhus, 1959). The blunted cone is the most realistic candidate. For bluntness ratios up to about 0.2 (tip radius to base radius) both theory and experiment show no increase in pressure drag (Hoerner, 1965). Small bluntness is consistent with the heat transfer analysis. Also, the entropy layer generated by the blunt nose results in a decrease in skin friction drag.

The geometry of other parts of the projectile (surface roughness, fins, etc.) may be used with Hoerner's correlations to provide a realistic estimate of the pressure drag arising from them. A clean projectile should not have a contribution of more than 0.01 to 0.02 from such sources.

3.7.2 Viscous Drag

The extremes of temperature and pressure so affect the properties of the air in and above the boundary layer as to make the results of any analysis questionable. However, the results of some drastic simplifying assumptions are useful.

With a constant velocity (the launch value), the isothermal atmosphere, and the viscosity varying with the (ideal) temperature to the 0.75 power, the reference Reynolds number for a distance x from the nose may be approximated.

$$Re_\infty = 113 \times 10^6 \, v_i \, e^{-h/a} \qquad (2)$$

Again, assuming also a boundary layer on the projectile flank at local atmospheric pressure and fully submerged under an entropy layer from a tip normal-shock, a local Reynolds number becomes

$$Re = Re_\infty \left[\frac{5}{M_\infty^2} \left(1.075 M_\infty^{0.57} - 1\right) \right]^{1.75} \qquad (3)$$

This, in combination with the excessively high air-to-surface temperature ratios encountered, would indicate a laminar boundary layer.

Using surface-temperature based correlations (Schlichting, 1968) simplified for a surface temperature of 2000 K and integrating over the surface length of the projectile, an approximation results:

$$C_{D_F} = 3 \times 10^{-4} \frac{L/d \; e^{h/2a}}{\sqrt{v_i \cdot L}} \tag{4}$$

where, v_i is in km/s and L is in meters.

Thus, the frictional drag may be ignored except for small and/or very slender projectiles, for which a more definitive analysis showed be made. The effects of blowing and/or ablation roughness should be negligible in view of the heat transfer results.

3.7.3 Applications

In view of the above analyses, a drag coefficient may be realistically assumed based on the nose cone, e.g., 0.10 to 0.15 for a projectile with a slightly blunted 12° (half-angle) conical nose.

The velocity at exit from the atmosphere, ignoring the gravitational effect, is a useful parameter to consider. This gives the fraction of the initial velocity lost due to aerodynamic drag:

$$= 1 - e^{-\frac{C_D}{2} \frac{\rho_o a}{\rho_m L}} \tag{5}$$

$$= 1 - e^{-\frac{1}{2B}} \tag{6}$$

where B appears as the primary parameter of the projectile

$$B = \frac{\rho_m L}{\rho_o a} \frac{1}{C_D} \tag{7}$$

The curve plotted in Figure 3-46 thus may be considered as the characteristic description of this problem. The velocity fraction along the trajectory is shown for three projectiles, as typified by the projectile parameter, in Figure 3-47. This consideration ignores the gravity term which must be retained to obtain the correct value for the velocity, and may affect the calculations for all aspects of the flight.

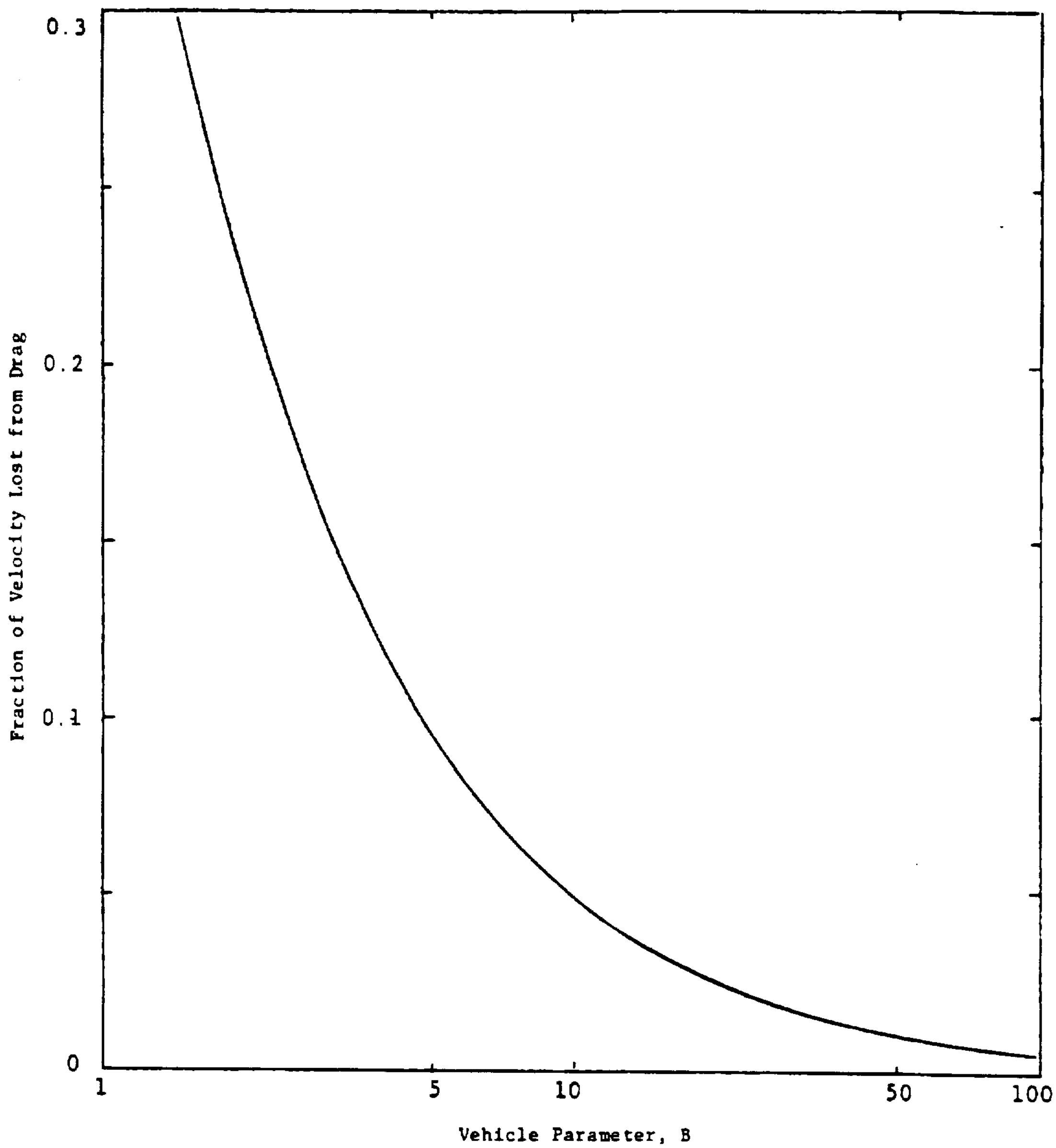


FIGURE 3-46. DRAG-LOSS ONLY IN TRANSATMOSPHERIC FLIGHT AS FRACTION OF LAUNCH VELOCITY

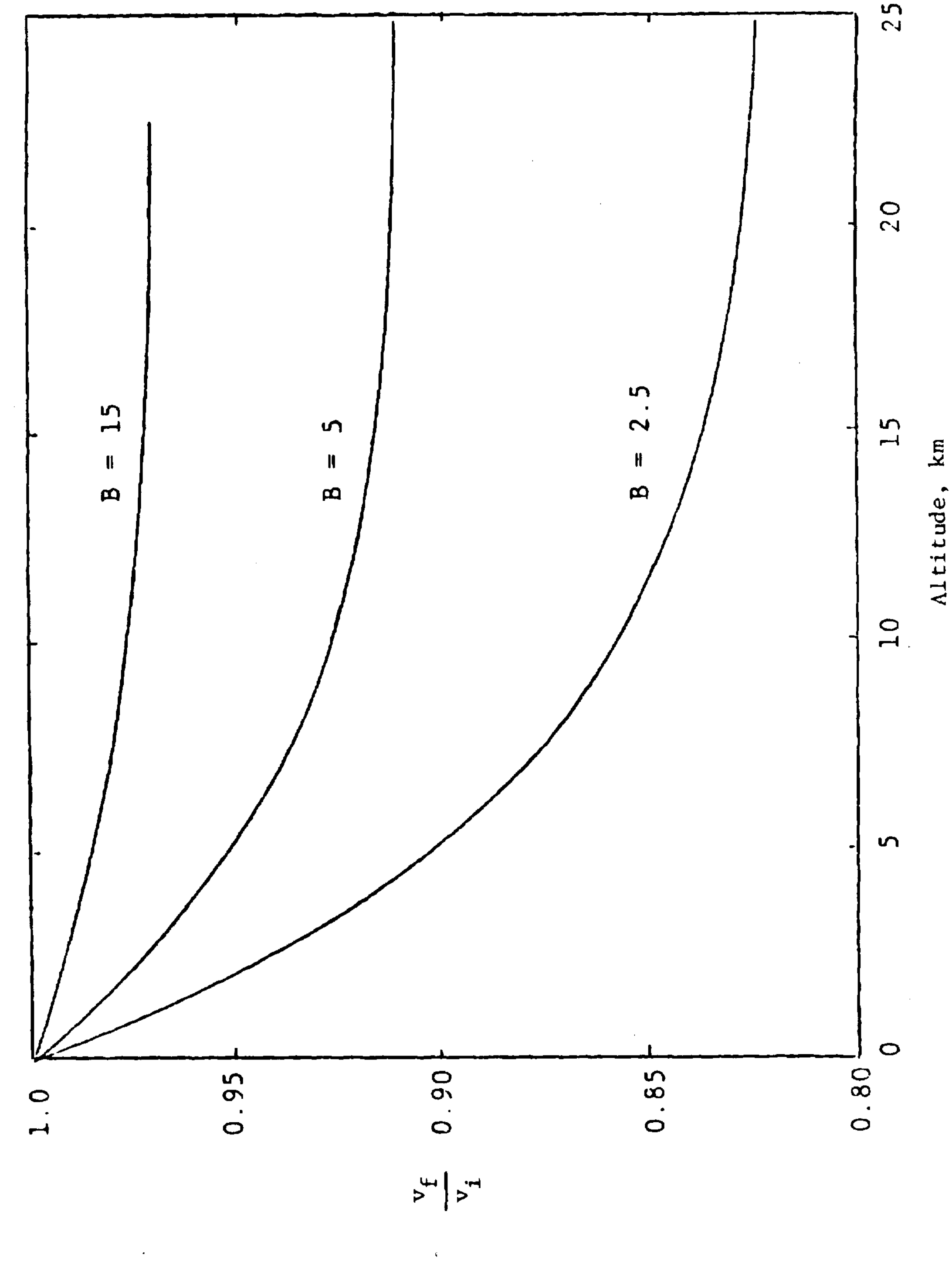


FIGURE 3-47. VELOCITY ALONG FLIGHT PATH FOR DIFFERENT PROJECTILE PARAMETER VALUES

3.7.4 Heat Transfer

By entrapolating the available information on convective heating and radiation (Cox and Crabtree, 1965) the values at the stagnation point may be estimated from:

$$\dot{q}_r = 4.9 \times 10^{-5}\, r_n\, v^{8.5}\, e^{-h/a} \tag{8}$$

$$\dot{q}_c = 2.72 \times 10^{-3} \frac{v^{4.5}}{\sqrt{r_n}}\, e^{-h/2a} \tag{9}$$

Both values may be effectively reduced by blowing from ablation (Holden, 1981) and by absorption within the gas cap (Cox and Crabtree, 1965), so that the above formulations will be conservatively high. Their combined effect may be determined by the following integration scheme.

For a given initial geometry, i.e. a cone topped by a tangent sphere (see Figure 3-45), a small incremental recession dx is introduced. The incremental volume of material is calculated and, from that, the heat absorbed by melting (latent heat of fusion). For the current radius of curvative the stagnation-point values for the convective and radiation heat transfer rates are calculated, summed and considered applicable over the entire surface of the spherical tip for the heat input. Equating this input heat to that absorbed in fusion, a time increment, dt, is calculated. Now an altitude increment is calculated since $dh/dt = v$. With the new altitude, the heat transfer rates may be calculated for the next step. It is noted that the above values so calculated will be conservatively high.

For some typical cases examined (see Reference Concept, Section 4.0) the material lost was quite small. For example, with a 15° cone of tungsten (latent heat of 44 cal/g) and a 1000-step integration across the atmosphere, about 1 cm of material was lost.

3.7.5 Vehicle Stability

The behavior of a cylindrical projectile with a conical or ogival nose cap may be estimated from available experimental data on supersonic and hypersonic vehicles (Savin, 1955; Perkins and Jorgensen, 1956; NAVWEPS, 1961). A summary of the effective center-of-pressure is given in Figure 3-48. Initially the force develops from an asymmetric flow over the nose cap and, at higher attack angles, a boundary layer separation develops from the cross-flow on the afterbody. At small angles, the nose-region has a normal force coefficient of about 0.034 per degree. These data may be combined to give a maximum pitching moment coefficient of about 0.05 about the projectile midpoint (i.e. the center-of-mass for a uniform, cylindrical projectile) at an attack angle of 8 degrees.

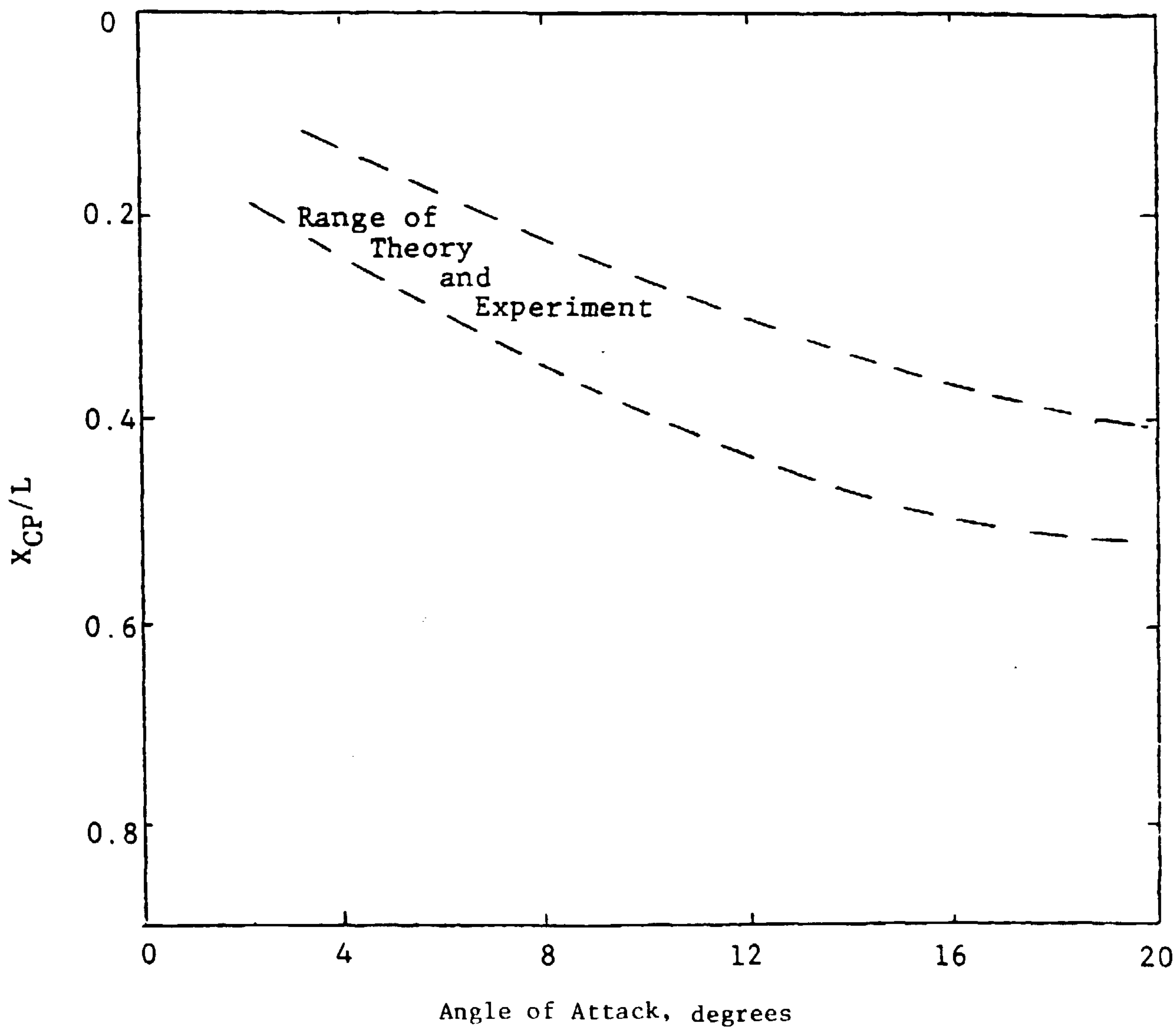


FIGURE 3-48. CENTER OF PRESSURE VARIATION FOR A SLENDER BODY

The pitch-onset is of more importance due to the extreme loads. Again from the given data, the pitching moment at small angles was estimated to be 0.0135 per degree. Assuming a mean density for the projectile of 0.75 that of iron, at sea level this translates to an angular acceleration:

$$\ddot{\alpha} \cong \frac{v^2}{L^2} \alpha \quad 10^{-3} \quad (\text{deg/s}^2) \tag{10}$$

which is seen to be very high even for an initial deviation of a fraction of a degree.

Figure 3-48 implies that the center-of-mass cannot be used to stabilize the projectile. However, a relatively small fin structure near the base would suffice to provide aerodynamic stability for a fraction of a second without appreciably affecting the drag.

3.8 Summary

After the accomplishment of various technical analyses on the proposed ESRL system concepts, it appears that it is technically feasible to develop such a system. The best long-term choice for a rail launcher system appears to be a distributed (integral) energy store (DES) system. It has a higher potential for performance than the single and multistage segmented energy store systems. Integrating the results of Sections 3.2, 3.3 and 3.4 into a comparison of data for the single, multistage-segmented, and multistage-distributed energy store rail launchers, results in Table 3-15. Energy storage is minimal with the multistage-distributed rail launcher. A summary of basic advantages and disadvantages of each are given in Table 3-16.

TABLE 3-15. COMPARISON OF LAUNCHER LENGTH AND TOTAL STORED ENERGY REQUIREMENTS FOR THE THREE RAIL LAUNCHER CONCEPTS

	Single Energy Store	Multistage Segmented Energy Store	Multistage Distributed Energy Store
Mission A			
Launcher Length, m	3000	2040	2040
Total Stored Energy, GJ	1600	874	571
Mission B			
Launcher Length, m	3000	2040	2040
Total Stored Energy, GJ	1300	722	450

TABLE 3-16. ADVANTAGES AND DISADVANTAGES OF THE VARIOUS OVERALL RAIL LAUNCHER TYPES

	Single Energy Store	Multistage Segmented Energy Store	Multistage Distributed Energy Store
Advantages	1. Switching is much simpler than other concepts	1. Shorter launcher due to leveled currents	1. Shorter launcher due to leveled current
	2. Considerable experimental experience demonstrated	2. Good efficiency	2. Best efficiency
	3. Can easily accommodate round bore	3. Can use small modular HPG's and inductors	3. Can use small modular HPG's and inductors
	4. Basically simple	4. Can easily accommodate round bore	4. Affords ability to tailor rail current to minimize rail damage to startup
		5. Affords ability to tailor rail current to minimize rail damage in startup	
Disadvantages	1. Longer launcher required	1. Switching is complex	1. Switching is complex
	2. Poorest efficiency	2. No significant experimental experience is available	2. No significant experimental experience is available
	3. High voltage drop due to current drop off	3. Somewhat complex	3. Has difficulty accommodating round bore concept
	4. Concentrated energy storage at breach of launcher		4. Significantly complex

4.0 ESRL REFERENCE CONCEPT DEFINITION

This section describes the Earth-to-Space Rail Launcher (ESRL) system that has evolved over the course of the study, and that is the basis for this preliminary feasiblity assessment. The concept is very preliminary and considerable additional analytical work is necessary to develop an optimum and detailed system description. However, it does represent a pooling of railgun expert opinion, engineering judgement, and properly defined mission requirements. The following subsections describe: (1) how the Reference Concept was selected; (2) the overall mission definition; and (3) specific ESRL element definitions.

4.1 Reference Concept Selection

The current Reference Concept for the Earth-to-Space Rail Launcher (ESRL) system has been developed from a considerble number of system options that were identified in the course of the study. A summary of the various options identified is shown in Figures 4-1, 4-2, and 4-3. The options selected for the Reference ESRL Concept are shown in the blocks with asterisks, other options are given below each category, in relative order of preference.

The Reference Concept is based, for the most part, on a concensus of opinion at the ESRL Concept Definition Meeting held at Battelle's Columbus Laboratory on August 12-13, 1981. Expert "Railgun" opinion was offered and considered in the selection process. The key individuals that participated in the selection process are listed below along with their respective experience and organizational affiliation.

Name	Experience	Organization
J. P. Barber	Railgun Technology	IAP Research, Inc.
A. C. Buckingham	Aerodynamics	Lawrence Livermore Labs
R. S. Hawke	Railgun Technology	Lawrence Livermore Labs
W. R. Kerslake	Electric Propulsion	NASA/LeRC
J. D. Lee	Aerodynamics	Ohio State University
R. A. Marshall	Railgun Technology	University of Texas
E. E. Rice	Propulsion	Battelle Columbus Labs
H. F. Swift	Ballistics	PAI Corporation
F. F. Terdan	Propulsion	NASA/LeRC
A. E. Weller	Combustion	Battelle Columbus Labs

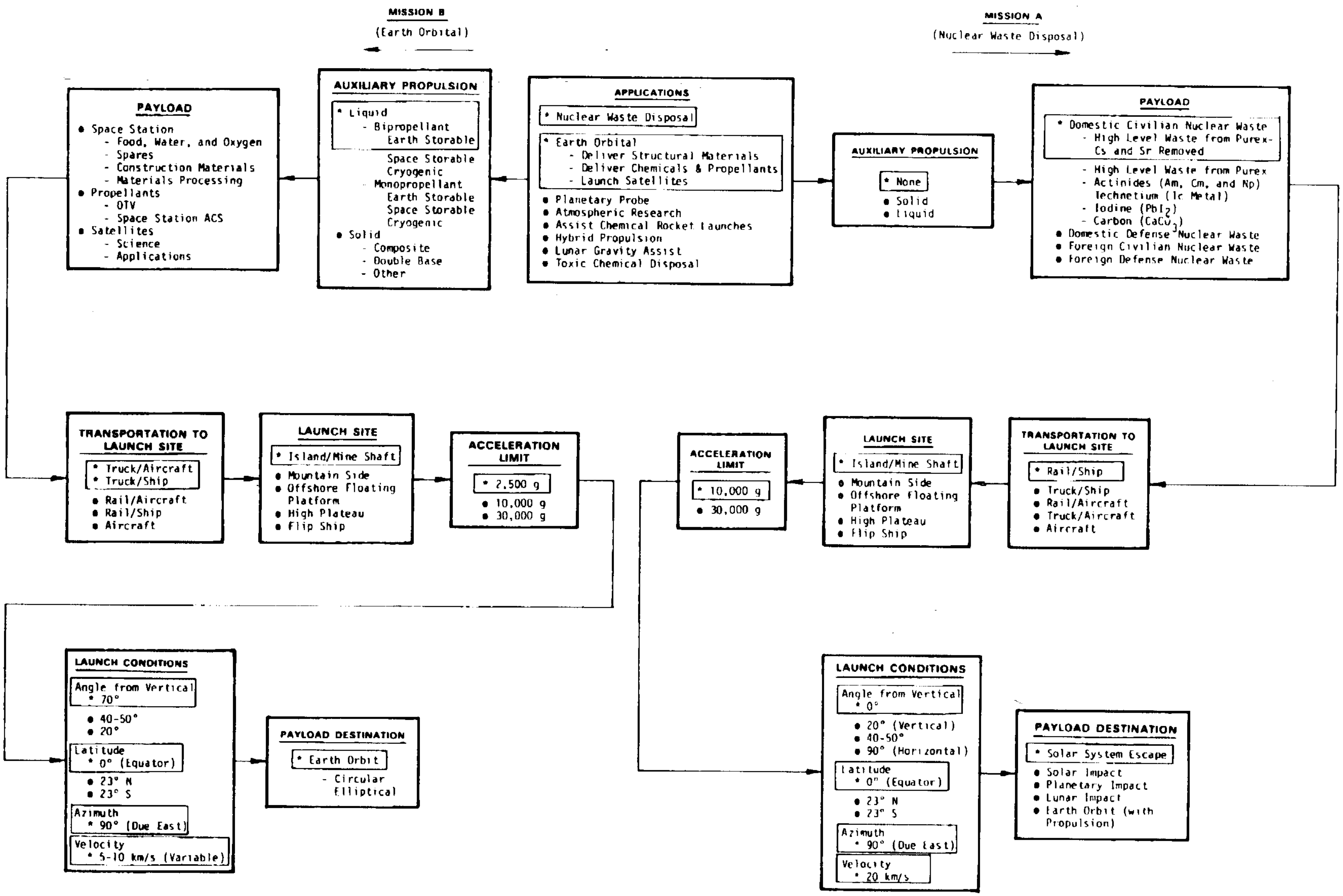


FIGURE 4-1. OVERVIEW OF ESRL REFERENCE CONCEPT OPTIONS

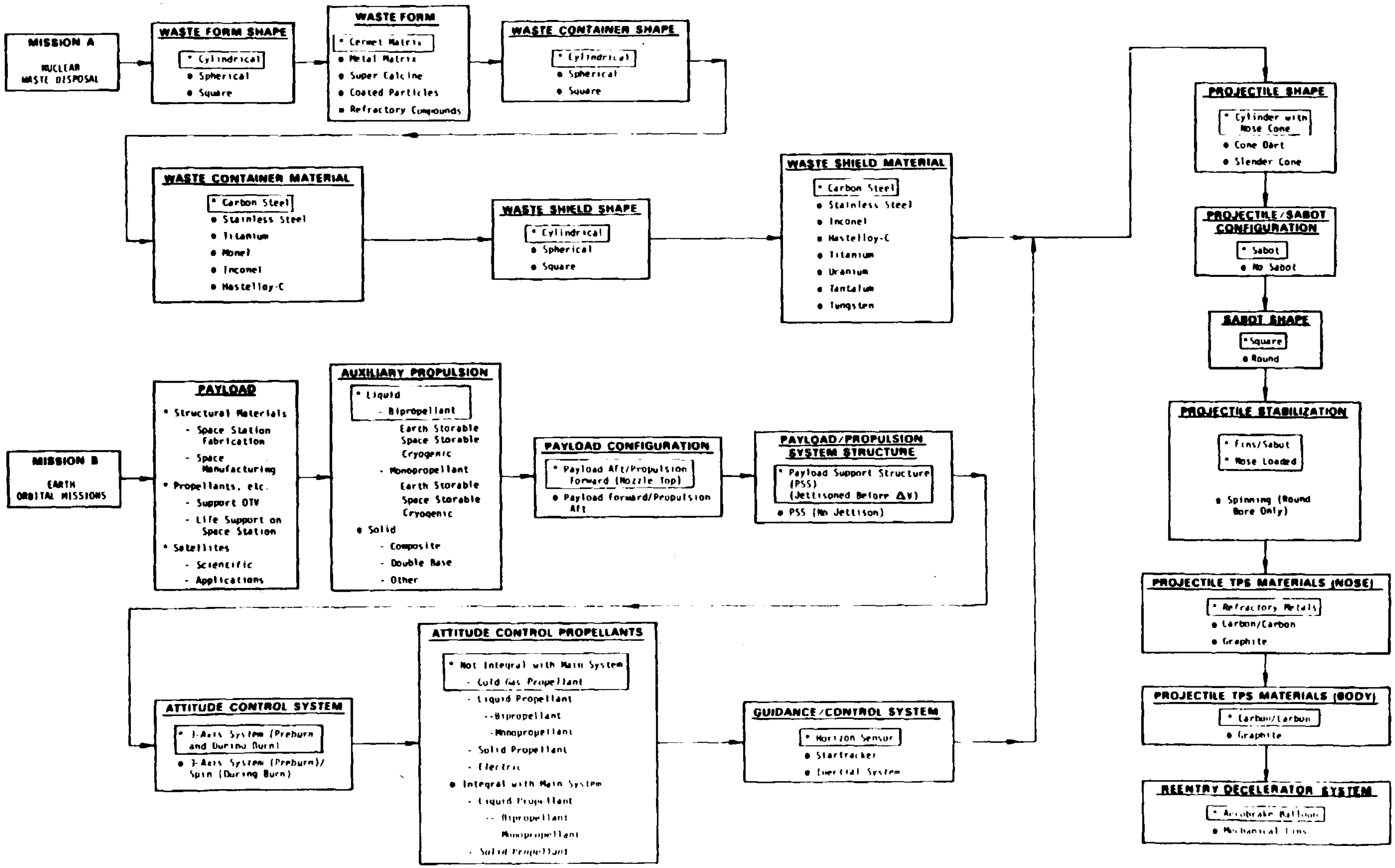


FIGURE 4-2. ESRL PAYLOAD/PROJECTILE OPTIONS

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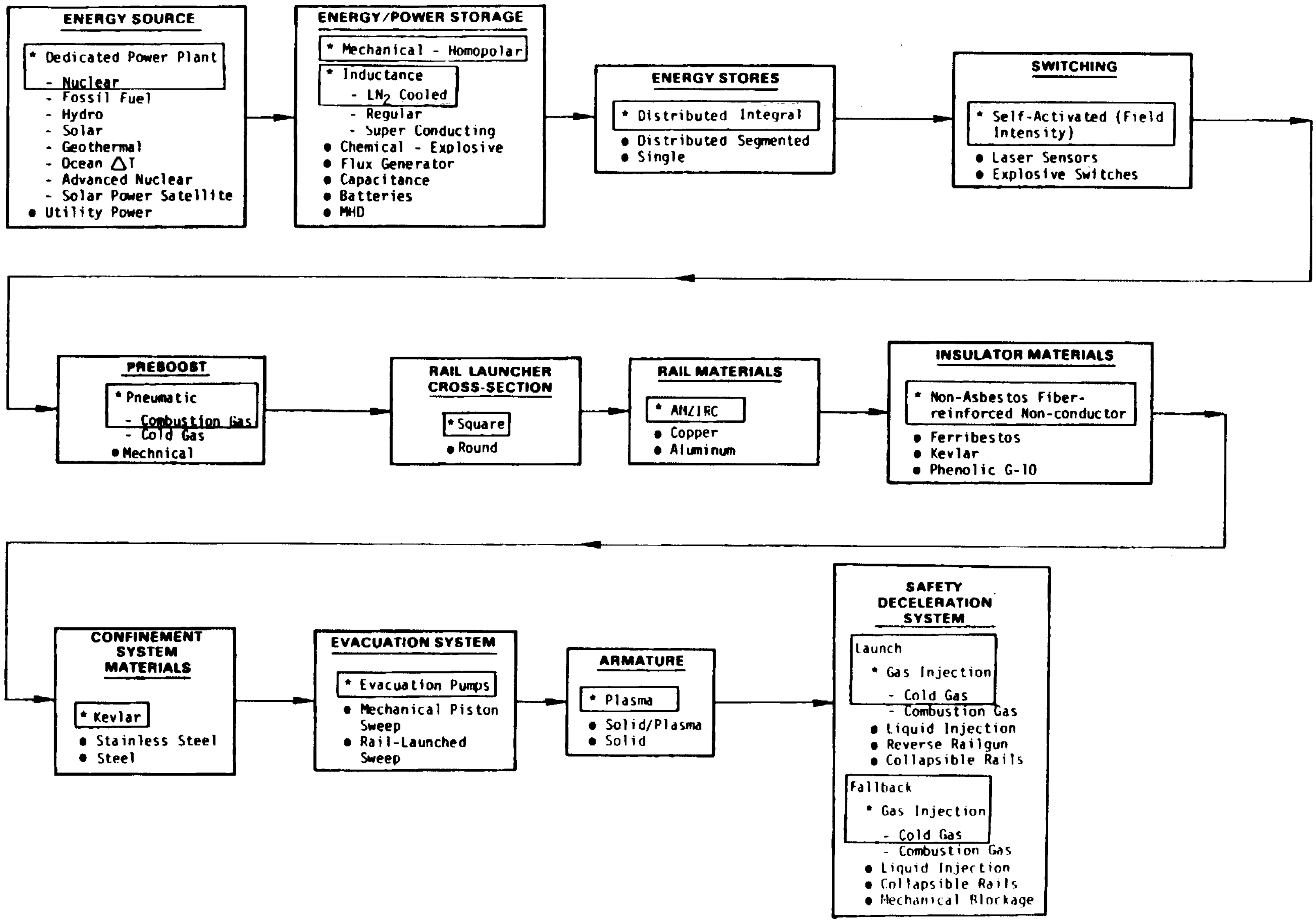

FIGURE 4-3. ESRL SYSTEM OPTIONS

At the end of the working meeting, after many concepts had been discussed, a concensus of opinion was reached on the choice of options for the ESRL concept. For the most part, these selected options are indicated in Figures 4-1, 4-2, and 4-3. Additional evaluation by Battelle and NASA resulted in a few modifications and additions.

For the primary "candidate" mission application, nuclear waste disposal in space, the definition of waste mix, waste form and space destination were guided by the study assumptions. The selections in these categories (shown in the figures) are in keeping with the current thinking within the space nuclear waste disposal study program (Rice et al, 1980, 1981 and 1982). Solar system escape is believed to be the most logical destination for space disposal within the capabilities of the ESRL system and the general safety requirements (see Section 2.3.1). With this primary candidate mission application, it is also possible to launch planetary flyby probes into the solar system. These missions would be handled by the same ESRL systems as used by the primary mission.

A secondary mission was selected to become part of the ESRL concept. Because of the excess power available throughout the major portion of the day (Mission A drives the peak power level needed), an Earth-orbital mission capability was believed to be warranted. This capability would be provided by a secondary rail tube and support systems. The general support functions would be the same as those for the overall ESRL system.

4.2 Overall Reference Concept Definition

The Reference ESRL Concept has been divided into five major activities for each of the two candidate missions. These are:

Mission A. Nuclear Waste Disposal in Space

(1) Nuclear Waste Processing and Projectile/Payload Fabrication
(2) Nuclear Waste Projectile/Payload Surface Transport
(3) Nuclear Waste Projectile/Payload Preparation at the Launch Site
(4) Rail Launch Operations
(5) Trajectory Monitoring.

Mission B. Earth Orbital Applications

(1) Projectile/Payload and Propulsion System Fabrication
(2) Projectile/Payload Surface Transport
(3) Projectile/Payload Preparation at the Launch Site
(4) Rail Launch Operations
(5) Trajectory Monitoring and On-Orbit Operations.

Consideration of rescue and recovery operations for Mission A are discussed in Section 2.4. Definitions and requirements for individual system elements are discussed in Sections 3.1 and 2.2.

4.2.1 Nuclear Waste Disposal in Space (Mission A)

The nuclear waste disposal in space mission (Mission A) was selected as the primary mission because it represents a large amount of mass that could be delivered to space and, because of the nature of a surface-based launcher, represents the most likely application, in the absence of onboard propulsion systems. Planetary flyby probes could also be performed by a Mission A ESRL system. However, the traffic alone would not justify ESRL development for only this application (planetary probe). The following subsections describe the concept of disposing of nuclear waste in space using an ESRL system. Later in this report this concept will be compared on the basis of risk and cost against the "conventional" way of performing the mission (e.g., via the Space Shuttle).

4.2.1.1 Nuclear Waste Processing and Projectile/Payload Fabrication

Spent fuel rods from domestic power plants would be transported to the waste processing and payload fabrication site via conventional shipping casks. Using the Purex process, high-level waste containing fission products and actinides, including 0.5 percent plutonium and 0.1 percent uranium, would be processed from these spent fuel rods (see McCallum et al, 1982). Then 95 percent of the Cs and Sr would be removed from the waste and taken to a mined geologic repository. After this separation, the high-level waste would be formed into a cermet matrix by a calcination and hydrogen reduction process. The waste form would then be fabricated into a 250 kg cylindrical waste form, with a partial cone toward the nose. Within a remote shielded cell, the waste form would be loaded into the flight container and radiation shield of similar shape. They would then be closed and sealed, inspected, and decontaminated. An auxiliary radiation assembly, which reduces the radiation dose to 1 rem/hour at 1 meter distance, would be used to transport the shielded cylinder to a projectile assembly area. Projectile components to be added to the basic structure include the side-body carbon/carbon thermal protection system, the instrument package, the dielectric system, the tungsten nose cone, the fins, and front and rear sabots. Several projectile assemblies would then be placed in a shipping cask with a passive cooling system for transport to the launch site. The shipping cask would be capable for use in both the rail and ocean transport portions of the surface transport activity. Auxiliary shields that can be used to allow safe handling at the launch site, would shield the projectiles in the shipping cask.

4.2.1.2 Nuclear Waste Projectile/Payload Surface Transport

The shipping cask, which provides appropriate additional shielding, thermal, and impact protection to comply with the Nuclear Regulatory Commission/Department of Transportation regulations, would then be loaded onto a specially designed railcar for transporting the assembled projectiles from the waste processing and projectile fabrication site (on the mainland) to a ship which would then transport the cask to the island rail launch facility (see Figure 4-4). Once the cask reaches the launch site, it would be offloaded into a nuclear projectile storage and checkout facility.

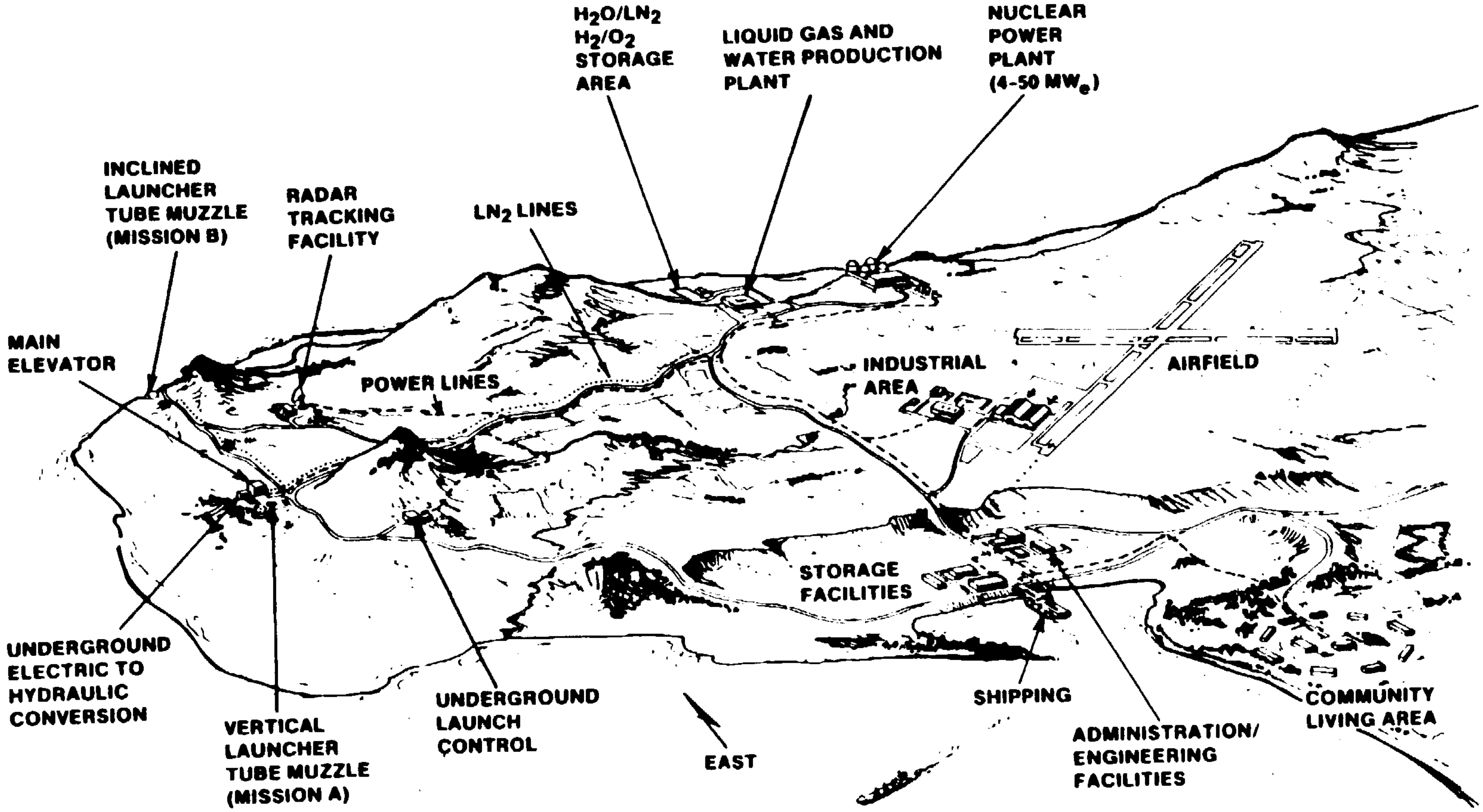

FIGURE 4-4. ARTIST'S CONCEPT OF THE ESRL REMOTE ISLAND LAUNCH SITE

4.2.1.3 Nuclear Waste Projectile/Payload Preparation at the Launch Site

The nuclear projectile storage and checkout facility would provide interim storage capability for 60 nuclear waste projectiles. This would afford sufficient capacity for unplanned delays (the expected launch rate is 2 per day). During storage, additional radiation shielding, thermal control, monitoring and inspection of the waste container would be provided. The integrated payload would then be stored in a shielded vault until the time of the launch. Prior to launch, systems checkout and inspection would occur.

4.2.1.4 Rail Launch Operations

In preparation for launch, the nuclear projectiles would be taken down the main elevator shaft (see Figure 4-5) to a temporary underground payload storage facility. The projectile would be kept there in temporary storage (capacity to store up to 10 nuclear waste projectiles). At the proper time (based upon launch windows and operational time lines), homopolar generators (HPG's) are then run up to speed over a period of several hours, the launcher tube (for Mission A) is evacuated, and proper liquid nitrogen cooling is provided to the inductors. The initial accelerator system would be checked, and all other systems would be readied and checked out prior to the beginning of the final launch countdown.

Before the final countdown, the weather and wind direction would be checked. The area would be cleared of all air and sea traffic. NORAD clearance will then be requested. (No satellites and manned space stations would be in the path of the projectile). A siren or alarm would be sounded, and all personnel on the island would enter the designated safe areas. Potential adversaries would be notified that the launch of nuclear waste payload is about to occur.

After all precautions are taken, the launch is initiated. The launch sequence would be computerized and automatically controlled. A liquid propellant-driven (H_2/O_2) piston accelerator system would be used to provide the projectile its initial velocity impulse of 1000 m/s. Verification of the attainment of this velocity, within reasonable tolerances, would then allow the automatic dumping of current into the first segment of rails as the projectile passes. A plasma armature would be formed behind the projectile. Automatic electronic switching would then be employed to dump power progressively into the rails as the projectile accelerates at 10,000 g's up the rail launcher tube. Surface tracking systems would be used to verify proper flight and velocity conditions as the projectile leaves the launcher muzzle. The nominal muzzle velocity would be 20,000 m/s and the fixed rail launcher tube would allow a vertical launch out of the atmosphere.

4.2.1.5 Trajectory Monitoring

An existing orbital radar satellite system would be used to monitor the trajectory of the waste payload as it would leave the vicinity of the

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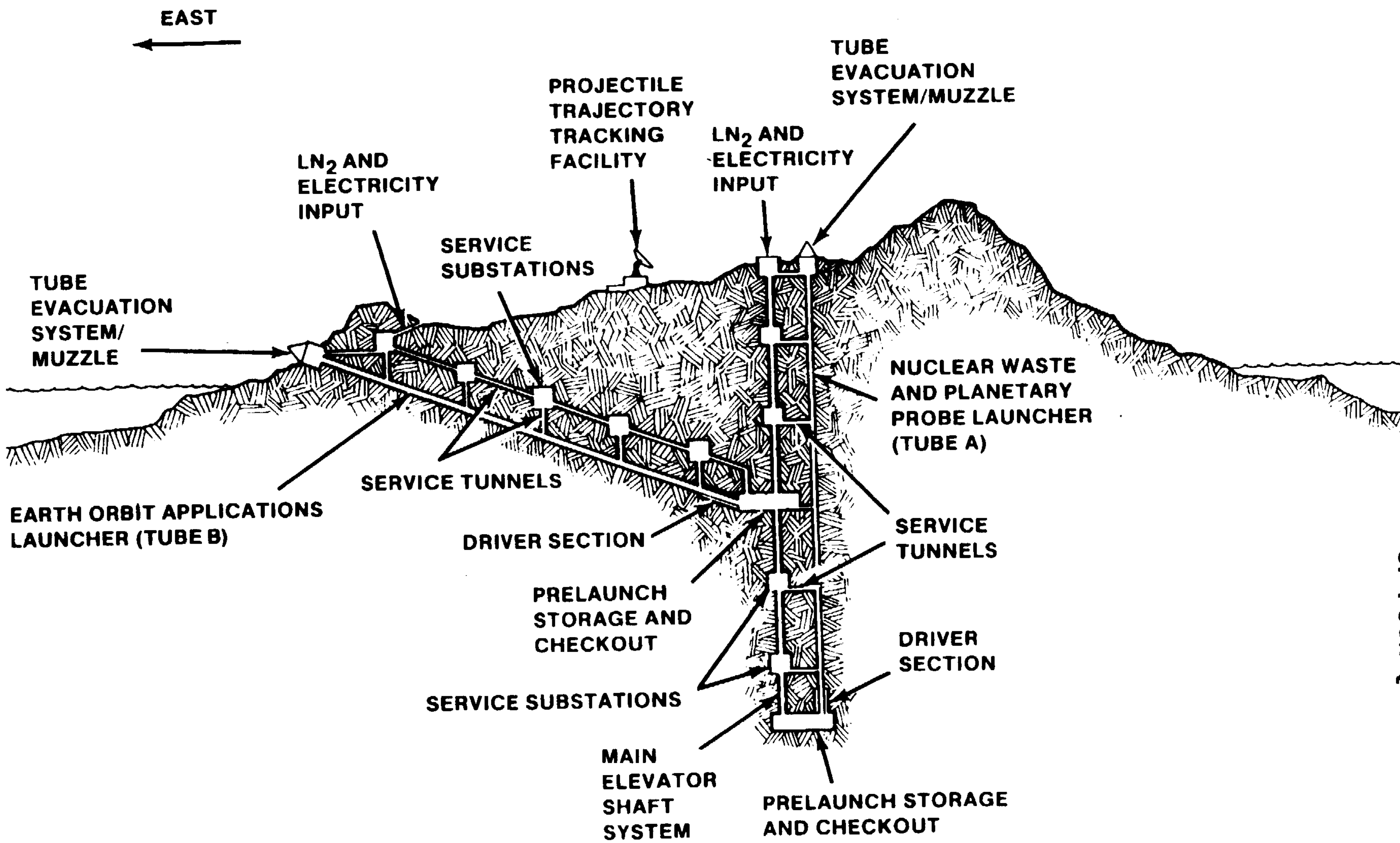


FIGURE 4-5. CROSS-SECTIONAL SIDE VIEW OF ESRL SYSTEM

Earth. Telemetry would be relayed back to Earth that would allow assurance that proper projectile velocity and direction were obtained to escape the solar system. A small radar would be used to provide tracking near the launch site.

4.2.2 Earth Orbital Applications (Mission B)

Earth orbital applications envisioned are space station construction and resupply, orbit transfer propulsion system propellant logistics, and small satellite launches. These are discussed in greater detail in Section 8.1. Discussion of the general mission is given below.

4.2.2.1 Projectile/Payload and Propulsion System Fabrication

The respective payload and onboard propulsion system would be assembled and prepared for transport to the remote launch site. All assembled components would be integrated and checked out prior to the systems being transported to the launch site.

4.2.2.2 Projectile/Payload Surface Transport

Projectile/payload surface transport to the remote island launch site would be conducted by ship and/or air. The payload, upon arrival, would be placed in proper storage until prelaunch preparation begins. Water payloads would originate on the island (island-based distillation plant provides source.)

4.2.2.3 Projectile/Payload Preparation at the Launch Site

As time approaches for its scheduled launch, the projectile/payload would be properly checked out and prepared for flight. On-board propulsion systems would be inspected. Earth orbital applications projectiles, nearing launch time, would be stored at the base storage facility (underground). Launch operations would typically be conducted during a 16-hour period, at times not interfering with nuclear waste launches.

4.2.2.4 Rail Launch Operations

When it is time to launch (based upon proper launch windows and preparation times), the projectile is loaded into the breech of the rail launcher (see tube at 20 degree angle in Figure 4-5). The main homopolar generators are then started, the launcher tube is evacuated, and all other systems are readied and checked out prior to the beginning of the final launch countdown.

Before the final countdown, the weather and wind direction would be checked. The area would be cleared of all air and sea traffic. NORAD clearance would then be requested. (No satellites and manned space stations should be in the predicted path of the projectile). A siren or alarm would be sounded, and all personnel on the island would enter the designated safe areas.

After all precautions are taken, the launch is initiated. The launch sequence would be computerized and automatically controlled. An initial velocity impulse of 1000 m/s would be provided by chemical means (see discussion in previous section). Verification of the attainment of this velocity, within reasonable tolerances, would then allow the automatic dumping of current into the first segment of rails as the projectile passes. A plasma armature would be formed behind the projectile. Automatic electronic switching would then be employed to dump power progressively into the rails as the projectile accelerates at 2,500 g's up the rail launcher tube. Surface tracking systems would be used to verify proper flight and velocity conditions as the projectile leaves the launcher muzzle. The maximum muzzle velocity would be 10,000 m/s (5000 m/s minimum) and the fixed rail would be aimed 20 degrees from the horizontal in an easterly direction. The launch tube would be constructed to match the orbital inclination of a space station activity; no orbital inclination specification is given here.

4.2.2.5 Trajectory Monitoring and On-Orbit Operations

An existing orbital radar satellite system would be used to monitor the trajectory of the projectile as it leaves the atmosphere and approaches the altitude where the payload propulsion system provides the necessary ΔV to attain the desired Earth orbit. The 3-axis attitude control system would be activated to provide the proper attitude prior to and during the propulsion maneuver. Telemetry would be relayed back to Earth that would allow measurement of the resulting orbital parameters. An orbit transfer system dispatched from an orbital base could then rendezvous and dock with the payload and transport it to its final Earth orbit destination. Reasonable sophistication of the projectile's guidance system could also permit a drifting rendezvous with the final destination.

4.3 Reference System Element Definition

The definitions for the Reference Concept system elements are described below. They are given in terms of both "generic" driver missions that are considered here. For each mission, a number of major system elements have been identified for discussion. These are:

- Payload Characteristics
- Projectile Characteristics
- Surface Transport Systems
- Launch Site Support Facilities
- Rail Launcher System
- Monitoring Systems
- Accident Recovery Systems (to support Mission A)
- Space Destination.

4.3.1 Nuclear Waste Disposal in Space (Mission A)

The various characteristics of major ESRL system elements relating to the nuclear waste disposal in space mission are described below. Most of the system elements are common with Mission B's needs.

4.3.1.1 Payload Characteristics

4.3.1.1.1 Nuclear Waste Source and Mix. The primary waste source is nuclear waste generated by the operation of U.S. commercial nuclear power plants. Table 4-1 provides data showing the quantity of waste for space disposal over the first twelve years of waste availability (Rice et al, 1982). The waste mix to be disposed of in space is reprocessed high-level waste (HLW--containing 0.5 percent of the Pu and 0.1 percent of the U that is present in the fuel rods at the time of reprocessing) that has been out of the reactor for 10 years. Also, at the time of reprocessing, 95 percent of the Sr and Cs is removed. Gases and transuranic (TRU) wastes, plus 95 percent of Sr and Cs, would be placed in a mined repository. The space waste mix defined here was that used as the Reference Case in the most recent "standard" space disposal of nuclear waste (see Rice et al, 1982).

TABLE 4-1. HIGH-LEVEL U.S. COMMERCIAL NUCLEAR WASTE AVAILABLE FOR SPACE DISPOSAL (IN CERMET FORM)

Year Waste Available for Disposal*	Kilograms of HLW, Less 95% Cs and Sr (Cermet Waste Form)
1	279,000
2	85,000
3	100,000
4	115,000
5	131,000
6	149,000
7	164,000
8	166,000
9	188,000
10	198,000
11	206,000
12	212,000
	1,993,000

Source: Adapted from data in Rice et al, 1982.

*Year one is 1989; storage and aging allows easier handling and lower accident risk as shorter lived isotopes decay.

4.3.1.1.2 Waste Form and Shape. The reference waste form for space disposal is the Oak Ridge National Laboratory iron/nickel based cermet (Rice et al, 1982). A cermet is a dispersion of ceramic particles in a continuous metallic phase. The reference cermet is formed by a process involving dissolution and precipitation from molten urea followed by calcination and hydrogen reduction to produce a continuous metallic phase (Rice et al, 1980). Non-hydrogen reducible oxides would form the ceramic portion of the ceramic/metal matrix waste form. This waste form has been shown to have superior properties as compared to other potential waste forms for space disposal (Rice et al, 1980). The iron/nickel-based cermet has high waste loading (67.4 percent), a thermal conductivity 9.5 Watts/m-C), a high density (6.5 g/cc), and a high structural integrity (Rice et al, 1982). The waste form would be made in the form of a cylinder/cone 25 cm in diameter and 95 cm in length (see Figure 4-6). The form would have a mass of approximately 250 kg. During the formation process, the waste form would be pressed and formed in a 0.5 cm thick steel container with an enclosed end. After formation, an end cap would be electronic beam welded to the main container rim. This activity would be conducted in a hot cell.

4.3.1.1.3 Waste Container. The primary containment for the radioactive waste will be a $\sim$30 kg stainless steel cylindrical container, 0.5 cm thick. This container provides primary containment for the waste form during the various defined mechanical and thermal loads to which the total payload is subjected in anticipated normal and accident conditions. These loads would be mitigated in varying degrees by the waste form itself, by the cylindrical flight radiation shield (also the auxiliary radiation shield during storage or surface transport and ground handling), and by the shipping cask which provides additional protection for surface transportation. To protect structural integrity, the primary steel container should not exceed a temperature of 416 C during normal conditions (Rice, 1981).

4.3.1.1.4 Radiation Shield. The container will be housed in a steel flight radiation shield. The shield is intended to limit radiation to no more than 10 rem per hour at 1 meter from the shielding surface under normal conditions. The shield would be approximately 11.5 cm thick, conform to the container shape, and have a mass of about 1100 kg. Auxiliary shielding would be designed such that radiation exposure limits for ground personnel are not exceeded during operations (this would be 1 rem/hr at 1 meter). For normal conditions, the temperature limit for flight radiation shield is 416 C (Rice, 1981). During accident conditions, the shield should not exceed 1280 C (Rice, 1981).

4.3.1.1.5 Waste Processing and Payload Fabrication Facilities. The waste processing and payload fabrication facilities are assumed to be co-located in the continental U.S. The reference waste mix would require a waste processing facility utilizing the Purex process. After separation and generation of the aqueous waste stream (5-year old waste), approximately 5 years of storage would occur before further processing would occur. The waste will then be put into its final cermet waste form.

DIMENSIONS, cm		ESTIMATED MASS CHARACTERISTICS, kg	
PROJECTILE LENGTH	170	WASTE FORM	250
WASTE FORM LENGTH	95	SHIELD/CONTAINER	1140
WASTE FORM DIAMETER	25	NOSE CONE	440
SHIELD/CONTAINER THICKNESS	12	AFT SABOT	40
PROJECTILE DIAMETER	51	FORWARD SABOT	100
SABOT THICKNESS	22 - 8	TPS	25
OVERALL DIAMETER	67	INSTRUMENTS	50
		FINS	10
		TOTAL	2055

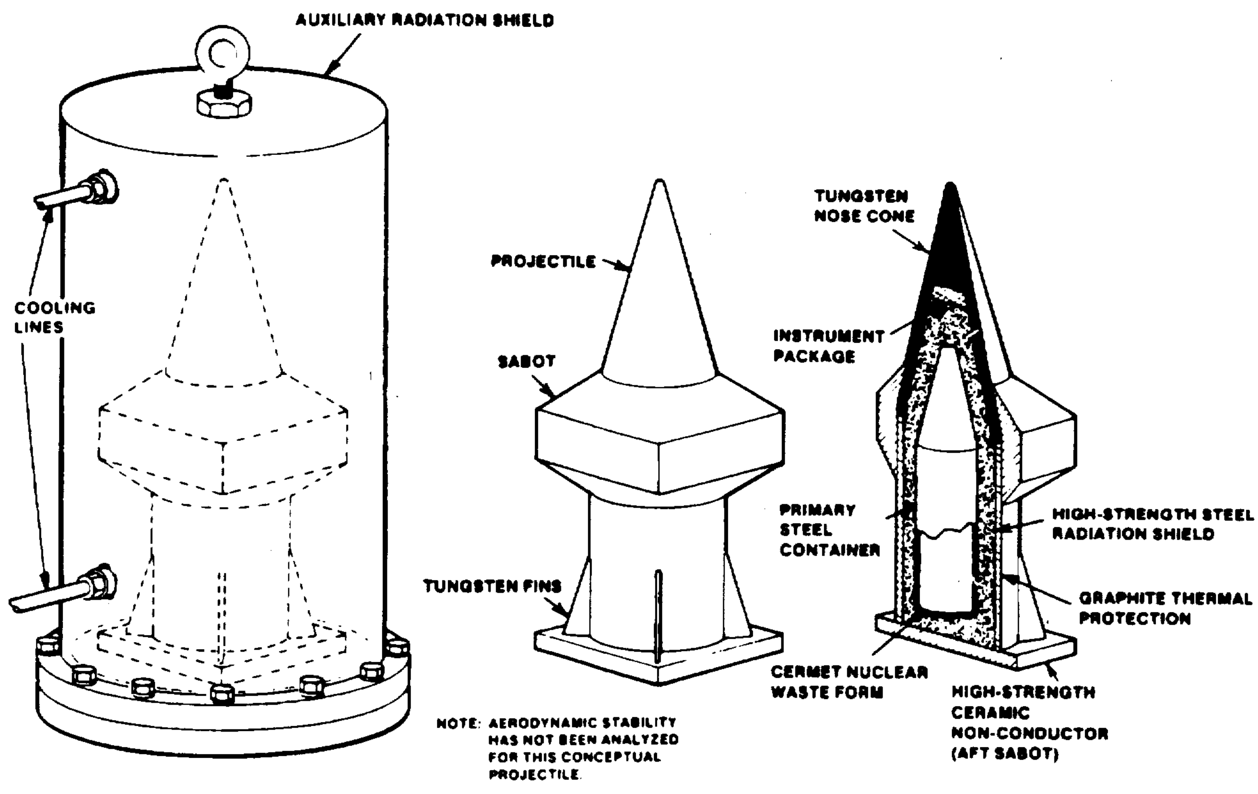


FIGURE 4-6. ESRL PROJECTILE CONCEPT FOR NUCLEAR WASTE DISPOSAL IN SPACE

The waste payload fabrication facilities would provide a series of interconnected, shielded cells for loading the waste form into the cylindrical containers, closing, welding, inspecting, decontaminating containers, and ultimate insertion into the flight and auxiliary radiation shield assemblies. Each cell would have provisions to connect the waste container and flight shield to an auxiliary cooling system. Each facility will provide interim storage for a number of shielded waste packages and equipment/systems for cask handling and railcar loading.

4.3.1.2 Projectile Characteristics

The overall projectile is depicted in Figure 4-6. The nose tip of the projectile would be slightly blunted and would be constructed of tungsten. As the projectile traverses the atmosphere, the tungsten metal is expected to begin melting cleanly, leaving an eroded, but smooth nose surface. The body of the projectile is the radiation shield covered with about 1 cm of carbon/carbon material applied in such a way to provide strength and thermal protection.

For stabilization during flight, four small stabilization fins would be attached to the rear of the projectile (see Figure 4-6). Also, at the rear of the projectile, a jettisonable, high-strength, ceramic non-conducting sabot would be used to: (1) protect the projectile and fins from excessive heating from contact with the driving plasma armature, and (2) proper positioning in the rail launcher tube.

A radio transmitter beacon will be located in the instrument package under the nose cone, along with an aerobraking decelerator system to be deployed automatically after the projectile leaves the atmosphere. This would allow a low velocity reentry if a misfire occurs, otherwise the payload will continue along its escape trajectory.

The assembled projectile, with fins, would be supported by a small sabot (forward and aft) for the acceleration portion of the launch. After the projectile leaves the ESRL, the sabot components would automatically be separated away in the initial contact with the atmosphere, leaving the projectile body and the exposed fins.

The total mass of the projectile, with its payload, is estimated to be about 2055 kg.

4.3.1.3 Surface Transport Systems

Surface transport systems used to support the operation of the ESRL system includes:

(1) Special equipment for supporting nuclear payloads
(2) Ships for hauling supplies and payloads
(3) Aircraft for transporting high priority material and people
(4) Launch site transport vehicles.

For transport from the waste fabrication facility to the launch site, the nuclear waste projectiles with auxiliary shielding would be housed in shipping casks which would afford additional shielding, thermal and impact protection to meet the Nuclear Regulatory Commission/Department of Transportation regulations. The maximum outside diameter of the shipping cask would be 3.05 meters. It is expected that perhaps as many as four projectiles could be transported in one shipping cask.

The cask would be transported from the projectile assembly facilities to the ocean front on a specially designed rail car which would adequately support and distribute the weight of the cask and provide acceptable tie downs. From the coast to remote island launch site, the cask would be transported by ship, also with acceptable tie downs. International guidelines and regulations would also be satisfied.

An airfield (see Figure 4-4) with two runways would permit landings of supply aircraft and passenger aircraft during non-critical launch operation periods. Aircraft should not be operating in the area when a launch is made. A hanger to provide adequate servicing for large jet aircraft is assumed necessary.

To support the workers and families at the ESRL launching site an adequate number of transport vehicles would be provided. Because of the aspects of isolation, few personnel vehicles would be required. Trucks and heavy transporters would be needed to service ESRL system hardware (replace homopolars, inductors, etc.).

4.3.1.4 Launch Site Support Facilities

The launch facilities used for the nuclear waste disposal mission would be located on a dedicated remote equatorial island. The island should be selected such that no uncontrolled population centers would be nearby (within radius of 50 to 100 km). Figure 4-4 is a concept of the ESRL launch site. Facilities which would be needed are discussed below.

4.3.1.4.1 Power Plant. A dedicated, 200 MWe, power plant is needed to supply the electrical power requirements of an ESRL system and supporting functions. As currently envisioned, the power plant facility would be comprised of four 50 MW_e nuclear reactors. These reactors would be similar to those on Navy ships. The current estimates of all electrical energy needs, both baseload and peaking have indicated that only about 100 MW_e is required at peaking. Four power plants have been assumed here to provide what is believed to be adequate backup during unscheduled reactor shutdowns and normal reactor maintenance.

4.3.1.4.2 Nuclear Projectile Storage and Checkout Facility. A secure, sealed, environmentally controlled, nuclear projectile storage and checkout facility would be required to store at least 60 projectiles, cool, monitor, and checkout the nuclear waste projectile systems, from the time the shielded projectile arrives at the launch site until the projectile is moved to the underground storage facility at the breach of the launcher.

4.3.1.4.3 General Storage Facilities. To support the supply needs of the staff and ESRL activities, general storage facilities are needed. Items to be stored include food, clothing, paper, gasoline, ESRL spares, etc.

4.3.1.4.4 Administration/Engineering Facilities. Buildings to house the administration and engineering activities of the island launch facility would be needed. These would be located near the community living and the industrial areas.

4.3.1.4.5 Industrial Area and Airfield. Various industrial facilities would be co-located at the two-runway airfield, and near the shipping docks, to support the maintenance and refurbishment activities of the remote island launch site. Facilities would include a homopolar repair/refurbishment shop, vehicle maintenance, aircraft maintenance, etc.

4.3.1.4.6 Community Living Area. A community living area, located at a practical distance away from the ESRL muzzles, would include the necessary housing, schools, shops and entertainment facilities to support the ESRL work force and their families. Housing would likely be in the form of apartment type structures.

4.3.1.4.7 Liquid Gas and Water Production Plant. A liquid gas (nitrogen, as well as small amounts of oxygen and hydrogen) and water (distilled sea water) production is needed to support the overall ESRL operation. Liquid nitrogen is required for cooling the ESRL inductors, liquid hydrogen and oxygen are required for the ESRL preboost, distilled (fresh) water is needed for (1) water supplies for the launcher base, (2) the hydraulic operation of the homopolars, and (3) as a payload for Earth orbit applications (Mission B). The plant would be located near the power plant facility. LN_2 lines would directly transfer the LN_2 to the ESRL system. LO_2 and LH_2 would be transported via truck. Fresh water would be distributed by an underground plumbing system.

4.3.1.4.8 Other ESRL Facilities. Other facilities needed to support ESRL operations include: (1) a radar tracking facility, (2) an underground electric-to-hydraulic conversion facility, (3) an underground launch control center, (4) a main elevator system, and (5) the basic rail launcher system itself (see next section).

4.3.1.5 Rail Launcher System

The preliminary Earth-to-Space Rail Launcher (ESRL) System concept developed during this study would indeed be a very large and complex system. Various aspects of the system are discussed below.

The rail launcher system (Mission A) would accelerate the nuclear waste projectile (see Figure 4-6) to solar system escape velocity by supplying an Earth surface muzzle velocity of the order of 20 km/sec in the proper direction. Figure 4-7 shows a cross-sectional view of the rail launcher tube. Figure 4-8 shows a side view of the current concept. The rail launcher would have a square bore (67 cm across). The materials to be used include AMZIRC (a copper alloy) for the rails, a non-asbestos, fiber-reinforced material as the insulator, and a Kevlar tube to confine the rails and insulator.

The ESRL system would be powered with some 10,200 homopolar generators (HPG's)/inductor units. These units would be distributed along the length of the launcher (see Figure 4-8). Self-activated switches would control the release of the 28.4 MA of current from the inductors to the rails. A combustion-gas-driven accelerator preboost system (200 meters long) would be used to obtain 1000 m/sec initial velocity.

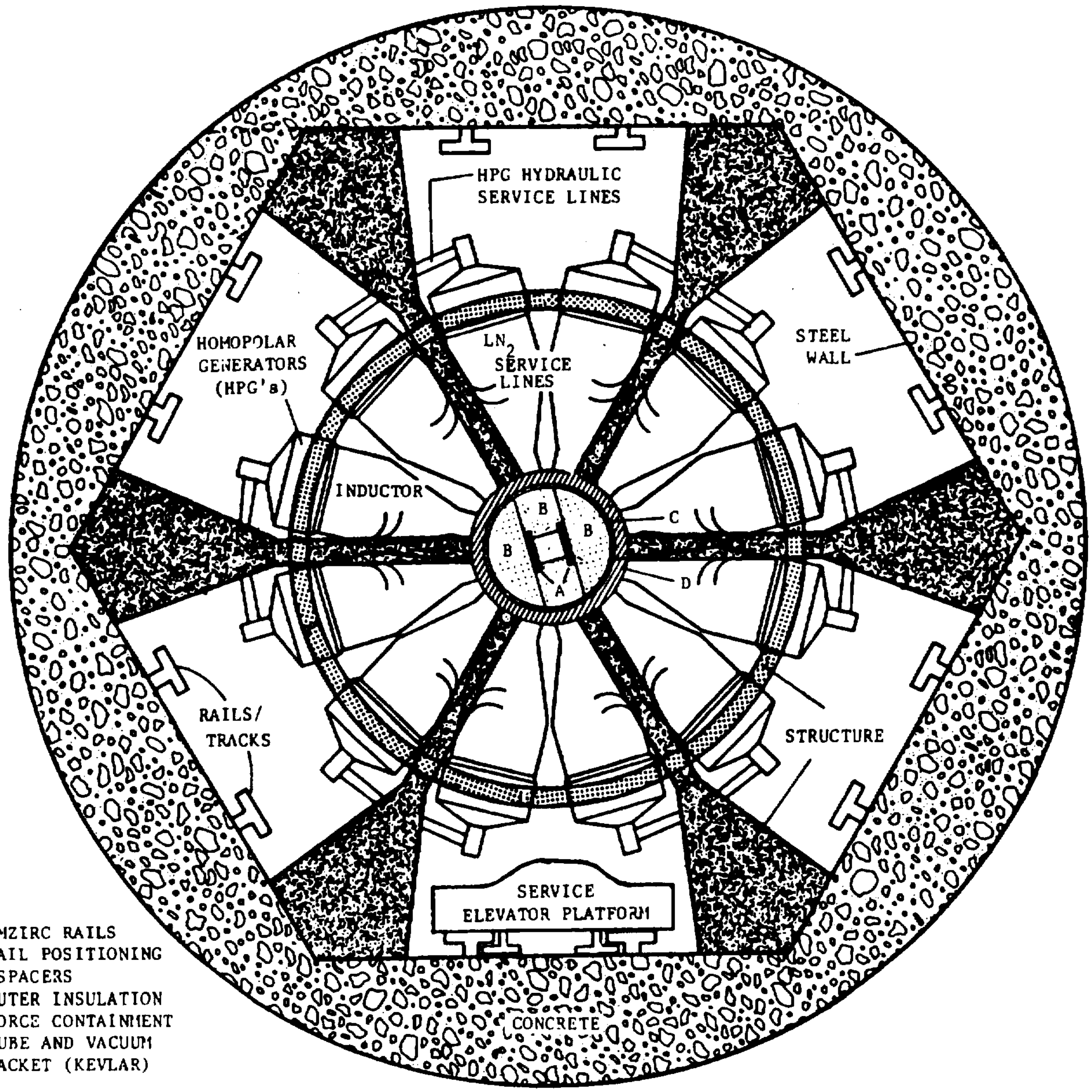
The ESRL system would have an emergency gas injection system to slow down and possibly stop the payload in the launcher tube if a misfire occurs during the initial part of the launch. Also, a gas injection system would be used to cushion a projectile falling back on to the end of the rail after an early misfire.

The ESRL system would be underground (see Figure 4-5), with access to it by tunnels. Provisions for maintenance and repair have been included.

The following paragraphs briefly discuss the rail launcher subsystems that have been conceptualized for this ESRL application. These are:

- Bore/rails
- Homopolar generator/inductor units
- Launcher/tube/support structure
- Preboost system
- Switching and control
- Storage facilities
- Service and access systems.

4.3.1.5.1 Bore/Rails. The pressure supplied to the base of the projectile is also exerted on the walls of the bore. Because the force expected on the projectile (2055 kg at 9.8 x 10^4 m/s^2) is 202 MN (45.4 Mlb), and it is assumed that the walls of the bore (AMZIRC rails) can withstand 44,800 N/cm^2 (65,000 psi), then the bore size would be 67.1 cm across. This would then require a sabot to match the round (51 cm diameter) projectile with the square 67.1 cm bore (see Figure 4-6). AMZIRC was selected for the rail material because of its excellent strength and high conductivity. Some properties of AMZIRC are listed below (Engineering Alloys Digest, Inc., 1961):



LEGEND:

A - AMZIRC RAILS
B - RAIL POSITIONING SPACERS
C - OUTER INSULATION
D - FORCE CONTAINMENT TUBE AND VACUUM JACKET (KEVLAR)

FIGURE 4-7. CROSS-SECTIONAL VIEW OF ESRL TUBE CONCEPT

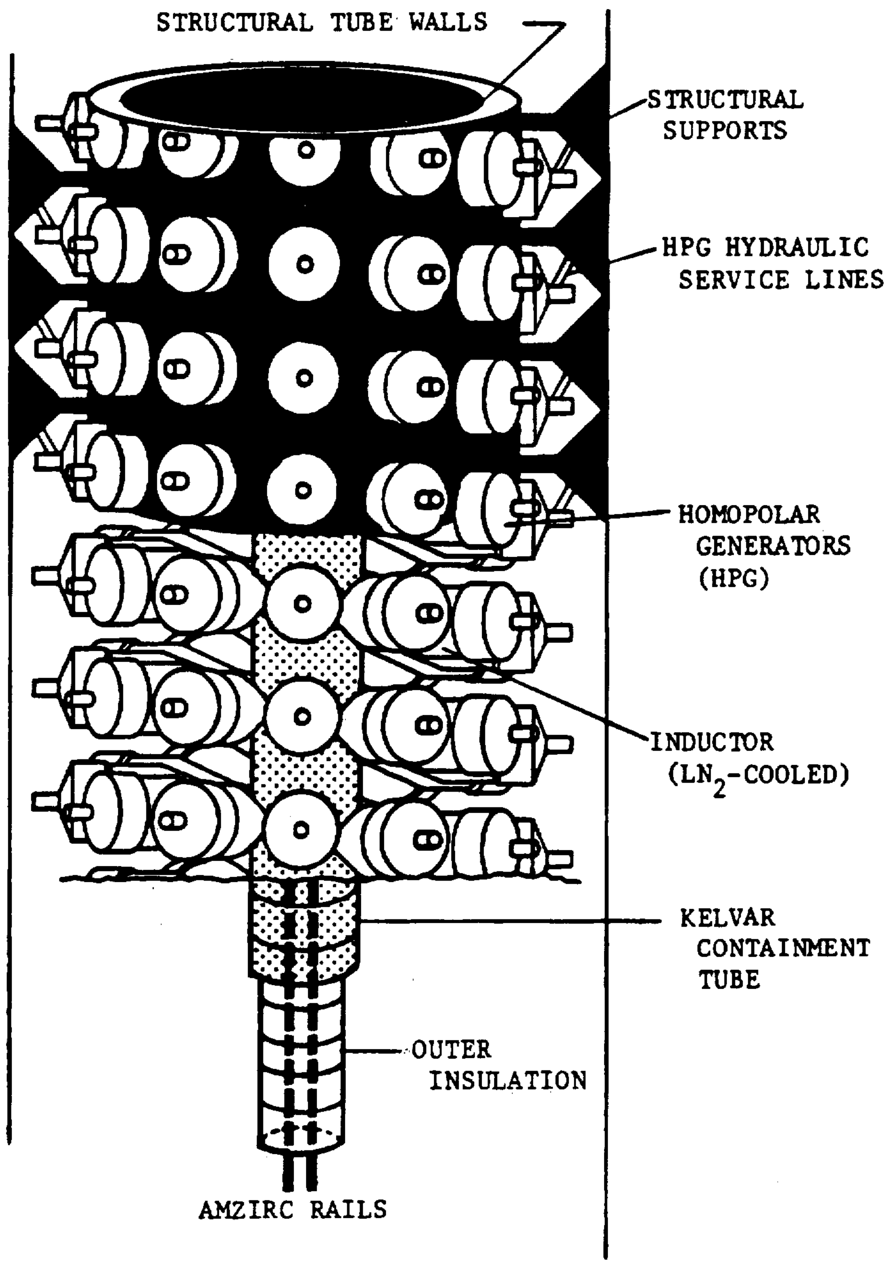


FIGURE 4-8. SIDE VIEW OF ESRL TUBE CONCEPT

Composition	- 0.1-0.15 percent zirconium
	- 99.9-99.85 percent copper
Density	- 8.89 g/cc
Electrical Conductivity	- 90-95 percent IACS
Tensile Strength (room temp)	- 48,260 N/cm^2 (70,000 psi)
Yield Strength (room temp)	- 42,056 N/cm^2 (61,000 psi)

Figure 4-7 shows the bore, rails, insulation and spacers in the center. The insulation and spacers would be made of a non-asbestos, fiber reinforced material. The outer force containment tube would be made of Kevlar.

4.3.1.5.2 Homopolar Generators (HPGs)/Inductor Units. The conceptual ESRL HPG/inductor unit for this application is shown in Figure 4-9.

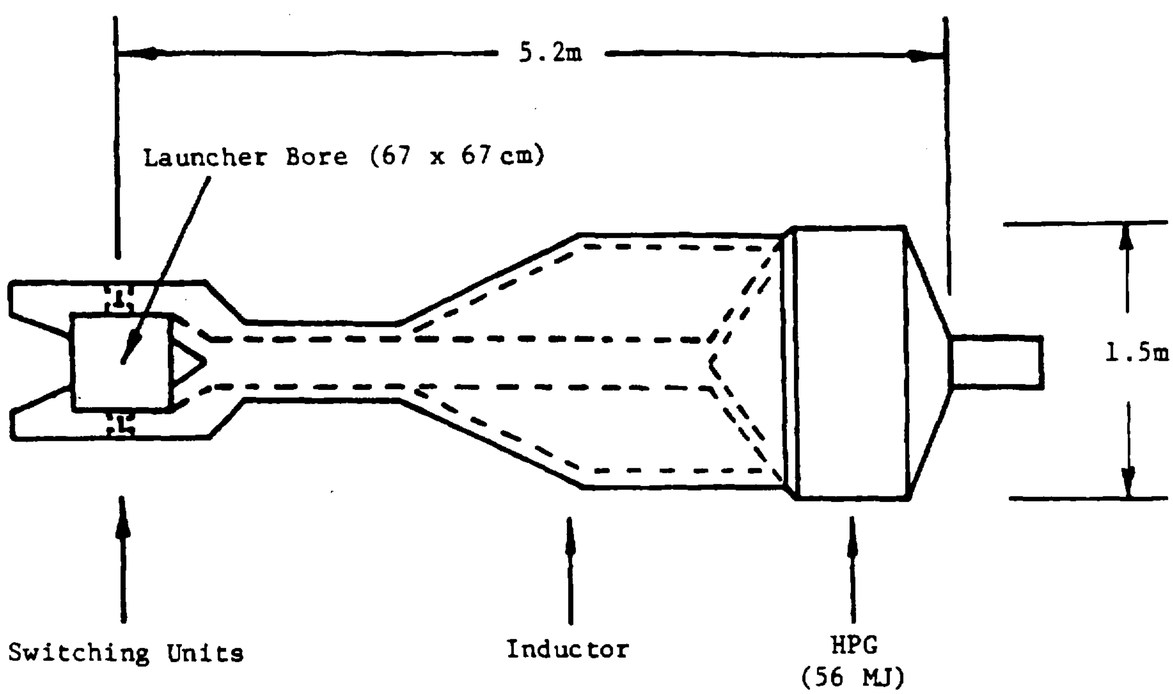


FIGURE 4-9. ESRL HPG/INDUCTOR UNIT

This concept was developed by R. Marshall (see Section 3.2). It has been assumed that an inductance of 0.5 μH/m is achievable in the rails. For mass efficiency, liquid nitrogen cooled aluminum inductors have been selected for the reference concept. Each inductor is expected to have a mass of 1.0 to 1.5 MT and about match the volume of a HPG. Based upon the required force of 202 MN to accelerate the projectile, and a 72 percent efficiency from the homopolar to the plasma armature, an input energy of 280 MJ per meter of rail is required for Mission A (see Figure 4-7). It has been assumed that the inductors and HPG's would be placed as close to the rails as possible, at 5 units for every meter of rail. Thus, for the entire length of the tube (2040 m), 10,200 units would be required. For Mission A, each homopolar generator would need to store about 56 MJ. The estimated mass for one HPG this size is about 10 MT (R. Marshall--see Section 3.2).

4.3.1.5.3 Launcher Tube/Support Structure. The launcher tube would be constructed by drilling out an 18 m diameter hole in the island bed rock. Steel structure and concrete would be employed to form the proper structural interface between the natural rock and the inner launcher structure. A preliminary supporting structure concept is shown in Figure 4-7. (No structural analysis has been conducted to support the concept). The weight of the HPG's, inductors, and core structure must be supported by the walls of the tunnel. The structure would also support the service lines for the LN_2 (for cooling the aluminum inductors) and the water-based hydraulic fluid to drive the HPG's.

4.3.1.5.4 Preboost System. The preboost system is needed to prevent damage to the rails during the initial acceleration. A desired initial velocity into the rail section is 1000 m/s (R. Marshall--see Section 3.2). A preboost concept for ESRL is shown in Figure 4-10.

It involves the continuous high pressure combustion $\sim$1500 N/cm^2 (2200 psi) of liquid hydrogen and liquid oxygen to force a movable piston against a hydrogen/nitrogen gas mixture, which in turn causes the sabot projectile to be accelerated up the tube. The concept is similar to a gas gun, but is continuously driven by the combustion process. The system can be properly designed such that the movable piston does not reach the rail section.

4.3.1.5.5 Switching and Control. The details of switching in the ESRL concept are still not developed to any degree of confidence. For the concept to be viable, this will have to be resolved. The current thought is that the projectile's movement/arrival would trigger the release of current from the inductors into the rails. R. Marshall discusses this concept in Section 3.2.

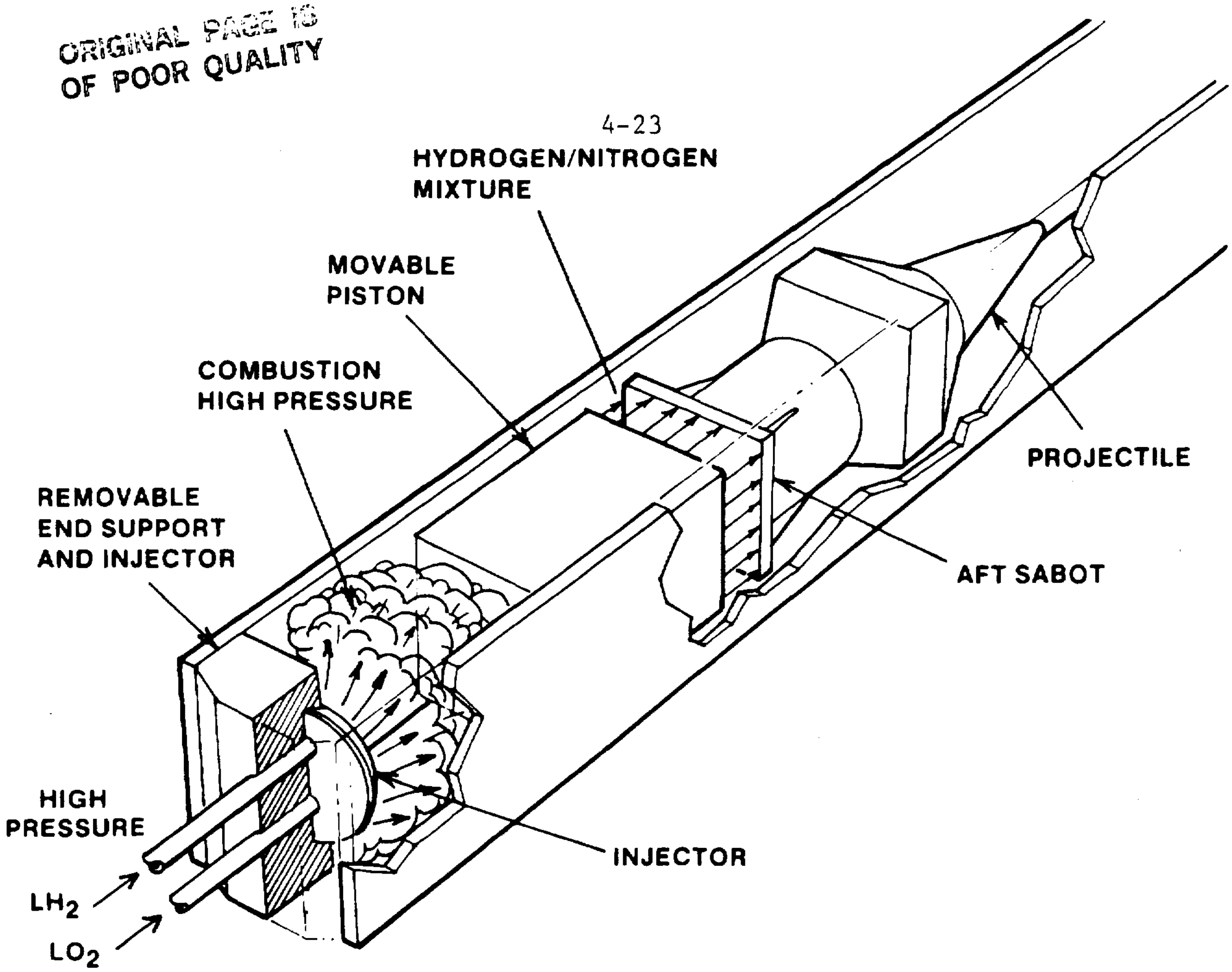


FIGURE 4-10. ESRL PREBOOST CONCEPT

4.3.1.5.6 Storage Facilities. Several storage facilities would be part of the rail system. Storage of the following items would need to be accomodated in the large facility at the breech or at intermediate level storage areas:

- 10 projectiles
- HPG, inductor and other spares
- LH_2 and LO_2 for preboost
- LN_2 for inductor cooling service
- H_2O for HPG hydraulic service.

4.3.1.5.7 Service and Access Systems. Figure 4-7 shows a service concept for the launcher tube. Six elevator systems allow servicing and/or replacement of malfunctioned ESRL subsystems. Elevator rails/tracks on the ESRL tunnel will permit vertical movement of the service platform. Access tunnels are shown in Figure 4-5 for both rail launcher tubes.

4.3.1.6 Monitoring Systems

Various monitoring systems would be used for the overall ESRL system and conduct of the mission. These monitors include devices for measuring radiation, acceleration, and temperature. A ground-based radar tracking system, an on-orbit satellite radar system to track the projectile, and instruments to provide data for tracking the payload after it leaves the Earth's influence are also part of the concept.

4.3.1.7 Accident Recovery Systems

Accident recovery teams would be made part of the operational disposal system. They would be responsible for all accident recovery operations, including accidents involving processing, payload fabrication, projectile assembly, railroad or ship transport, projectile/payload preparation at the rail launcher site, launch, and possible reentry.

4.3.1.8 Space Destination

The space "destination" for the nuclear waste would be to escape the solar system with an excess velocity of 1 km/s. The minimum ideal velocity requirement from the Earth's surface for this mission is 16.67 km/s, including the 1 km/s excess velocity at escape. The muzzle velocity of 20.0 km/s at the surface coupled with a drag coefficient (C_D) of 0.1, implies a velocity loss of about 1 km/s (C_D = 0.2 gives a 2 km/s loss).

4.3.2 Earth Orbit Applications Mission

The various characteristics of major ESRL system elements relating to the Earth orbit applications mission are presented below. Only items peculiar to this mission application are discussed here.

4.3.2.1 Payload Characteristics

Payloads which are envisioned for launch for Earth-orbital missions include:

(1) Structural materials
(2) Propellants and chemicals
(3) Satellites.

Structural materials could include metals, components, or plastics for delivery to low-Earth orbit (LEO) to manufacture space stations or platforms, or parts thereof. The structural members would be launched by the ESRL and injected into LEO. An orbit transfer system could intercept the payload and transport it to the space station building activity.

The ESRL also could be used to launch propellants to orbit. These propellants could be used to refuel on orbit propulsion systems. Another fuel use would be to power support systems on space stations.

Satellites could be launched on the ESRL. Prime candidates would be scientific satellites which operate in the LEO regime. Examples include remote sensing satellites and observation satellites. Section 8.0 discusses ESRL applications in more detail.

4.3.2.2 Projectile Characteristics

The Earth-orbital applications projectile would consist of the following subsystems (see Figure 4-11):

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

The forward and aft sabot, nose cone, instrument package, thermal protection system, and fins are basically the same as described in the previous section for the Mission A projectile.

The payload support structure (PSS) serves a dual purpose. First it would have an aerodynamic shape and provide the structural integrity of the projectile. Second, attached fins would stabilize the projectile during atmospheric flight. The PSS also would provide the structural support for the propulsion system.

The liquid propulsion system (ClF_3/N_2H_4) would be in the forward part of the PSS, with the nozzle forward. The payload is aft, and attached to the propulsion system. After atmospheric flight, and prior to the circularization burn, the PSS would be jettisoned. A cold gas attitude control system (ACS) would provide the proper altitude for the on orbit burn and for proper altitude control while waiting for the arrival of the orbit transfer system. An astronics system coupled with a horizon sensor would be located near the nozzle.

The mass of the Earth orbital projectile is 6,500 kg, providing a maximum payload mass of approximately 650 kg.

4.3.2.3 Surface Transport System

Surface transport systems used to support Mission B activities include:

(1) Aircraft for transporting high-priority materials, payloads, and people

(2) Ground transport vehicles for local transportation

(3) Ships for hauling supplies and bulk material.

DIMENSIONS, cm	
PROJECTILE LENGTH	360
PROJECTILE DIAMETER	90
SABOT THICKNESS	26 - 5
OVERALL DIAMETER	100

ESTIMATED MASS CHARACTERISTICS, kg	
INSTRUMENTS	30
MAXIMUM PAYLOAD	650
ASTRIONICS	25
ACS	50
PROPULSION SYSTEM (DRY)	425
PROPELLANT	1150
NOSE CONE	1020
FORWARD SABOT	200
AFT SABOT	100
PSS	2730
TPS	100
FINS	20
TOTAL	6500

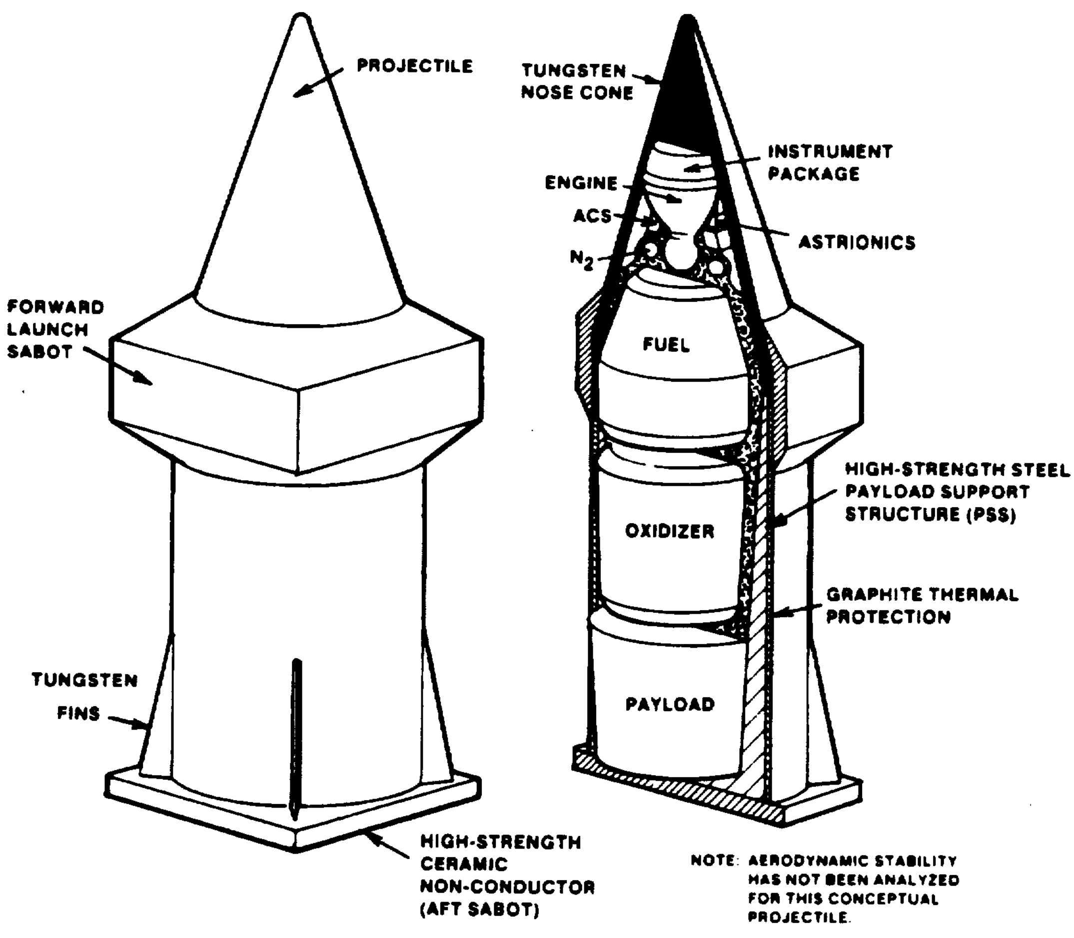


FIGURE 4-11. PROJECTILE CONCEPT FOR EARTH ORBITAL APPLICATIONS

The payload, propulsion, and projectile systems components would likely be assembled on the mainland and transported by truck to aircraft or ships to be transported to the remote island. Aircraft could be used to transport the projectiles, ESRL personnel, and high-priority materials to the launch site. Ships could also be used to transport supplies and bulk-material payloads, such as materials to be launched for space station fabrication.

4.3.2.4 Launch Site Facilities

The launch facilities used for the Earth orbit mission would also be located on the same dedicated remote island launch site, as previously discussed. The launch site would be shared with Mission A (see previous section).

4.3.2.5 Rail Launcher System

The rail launcher system would accelerate the projectile at no more than 2,500 g's to velocities on the order of 5-10 km/s, depending upon the exact Earth orbital mission requirements (see Section 3.1).

The rail launcher would have a square bore ~100 cm wide (see Figure 4-7). The materials to be used include AMZIRC for the rails, a non-asbestos, fiber-reinforced material as the insulator, and Kevlar to confine the system. The rail launcher for Mission B would be placed near the one for Mission A such that the main elevator shaft could be shared. The rail launcher tube would be 2040 m long and be pointed east, and have an elevation angle of 20° (20 degrees from horizontal). The ESRL system would be underground (see Figures 4-4 and 4-6), with access to it by mine shafts. Each homopolar generator/inductor unit would be accessible for repair and/or replacement--see Figure 4-7. The plasma current is slightly smaller (25.2 MA) than used in Mission A. The kinetic energy at launch is 325 GJ, which is 79 percent of that of Mission A. As in Mission A, 10,200 HPG/inductor units are required but they would have to supply only 44 MJ/HPG. To simplify operations and maintenance, HPG/inductor systems in Tube B would be identical to those in Tube A.

Self-activated switches would control the release of the energy in the inductor stores. A combustion gas accelerator preboost system (see previous section) would be used to obtain 1000 m/s initial velocity. A gas injection system would be used to cushion a projectile falling back onto the aft end of the rail after an early mission misfire.

4.3.2.6 Monitoring Systems

For Earth orbital payloads, monitoring systems will also be needed but, they are not as critical as for Mission A. For those missions which require monitoring, such as satellites and payloads which would be intercepted by orbit transfer systems, monitoring systems would be part of the payload, and would include on-board telemetry, such that the payload could be tracked by stations on Earth and satellite systems on orbit.

4.3.2.7 Space Destination

The space destination for the payloads would be Earth orbit, with the prime mission being circular low-Earth orbits, but secondary eliptical orbits and higher altitude circular orbits (including geosynchronous) are possible (see Section 3.1).

5.0 SAFETY AND ENVIRONMENTAL IMPACT ASSESSMENT

This section documents the preliminary safety and environmental impact assessments for the ESRL Reference Concept (see Section 4.0 for concept definition). Since nuclear waste disposal in space mission is a major driver in the conceptualization of the current ESRL system, it was important to provide a preliminary assessment of the safety and risk aspects of this concept against the "standard" Shuttle-based disposal of nuclear waste in space (see Rice et al, 1982). Also, it was important to identify any environmental impact "show stoppers" or potential problem areas for normal and accident ESRL operations.

This section has been divided up into three major sections: (1) Identification of Possible ESRL System Failure Modes (Mission A); (2) Preliminary Accident Response Analyses for Certain Major Failures (Mission A); and (3) Preliminary Environmental Impact Assessment of the ESRL Reference Concept.

5.1 Identification of Possible ESRL System Failure Modes (Mission A)

A preliminary evaluation of possible failure modes or accident events for the ESRL nuclear waste disposal mission was undertaken. The approach used considered: (1) the definition of the Reference Concept, as given in Section 4.0 of this report; (2) previous work on the radioactive material release risk for "standard" Shuttle-based nuclear waste disposal in space (Rice et al, 1982); and the use of top-level fault trees for ESRL mission phases to aid in the identification problem areas.

The fault tree approach was selected to help identify failures. The fault tree approach is a technique by which the component failures leading to system failure can be logically deduced. Application of the technique yields combinations of basic events whose occurrence causes the undesired failure events (containment breach). These event combinations can then be evaluated by various screening techniques to determine the high risk scenarios and their probability of occurrence (if data are available). For its application, the fault tree method requires probability information about all of the individual component failures and events. The fault tree technique is well suited to analyzing the rapid events (such as ESRL launches--which have discrete, but currently unknown probabilities). Because probability data are not available, for the ESRL concept, no risk calculation is possible at the present time; however, comparable to standard space disposal, it may be possible to suggest what the overall reliability of the ESRL system might have to be.

The first activity involved the definition of the various mission phases. Six ESRL mission phases for the nuclear waste disposal in space were defined as:

- Phase 1--Terrestrial Transport
- Phase 2--Prelaunch Activities

- Phase 3--Preboost
- Phase 4--ESRL Acceleration
- Phase 5--Sabot Jettison
- Phase 6--Atmospheric and Space Flight

Various accidents and malfunctions that could occur during these mission phases were identified and top-level fault trees were developed. Only top-level events that lead to the release of nuclear waste material into the Earth's biosphere are shown (see Figures 5-1 through 5-6). These are discussed in the following sections.

5.1.1 Phase 1--Terrestrial Transport (Mission A)

The two major candidate events (see Figure 5-1) which could lead to nuclear waste release to the biosphere from the radiation shield (primary container assumed to be included in the shield for purposes of discussion) are (1) shield breakage via a mechanical means (Event 101); or (2) shield corrosion (Event 102). The types of events that may cause shield breakage during terrestrial transport are related to: (1) a railroad transport accident; (2) an accident at the handling facilities; (3) a ship accident at sea (e.g., two ships collide); and (4) a transporter accident at the launch site. The probability of any of these events happening and causing shield breakage is extremely low and not considered to be a significant contributor to release risk. Release in sea water, with long-term corrosion, could occur as a result of a shipping accident at sea, where the ship with its cargo actually sink to the ocean floor and recovery activities ultimately fail. The shipping accident could be caused by severe weather, a critical ship accident, or a critical ship failure, followed by sinking. The probability of release in sea water from terrestrial transport is believed to be extremely low and is not considered a significant contribution to the total release risk.

5.1.2 Phase 2--Prelaunch Activities (Mission A)

The two major candidate events which could lead to nuclear waste release to the biosphere are shown in Figure 5-2: shield breakage (Event 201) or shield melting (Event 202). Shield breakage during prelaunch activities could occur from a transporter accident, handling accident, or elevator system failure. The consequences of any of these are not considered significant and that only a very small quantity of material would be released if the shield actually were to breach. Therefore, these are not considered significant contributors to the total release risk. For shield melting, two scenarios have been identified: (1) an external melt caused by a severe fire, and (2) an internal melt caused by a critical cooling loss. Melting due to a severe fire is considered to have a very low probability because of the precautions that would be expected to be taken to prevent such an occurrence, and the low amounts of combustible material that would be available to feed the fire. An internal melt would not be expected to be a problem because of the reduced thermal energy generation of the high-level nuclear waste (a period of from 30 to 50 years for aging the high-level nuclear waste is assumed and as a result,

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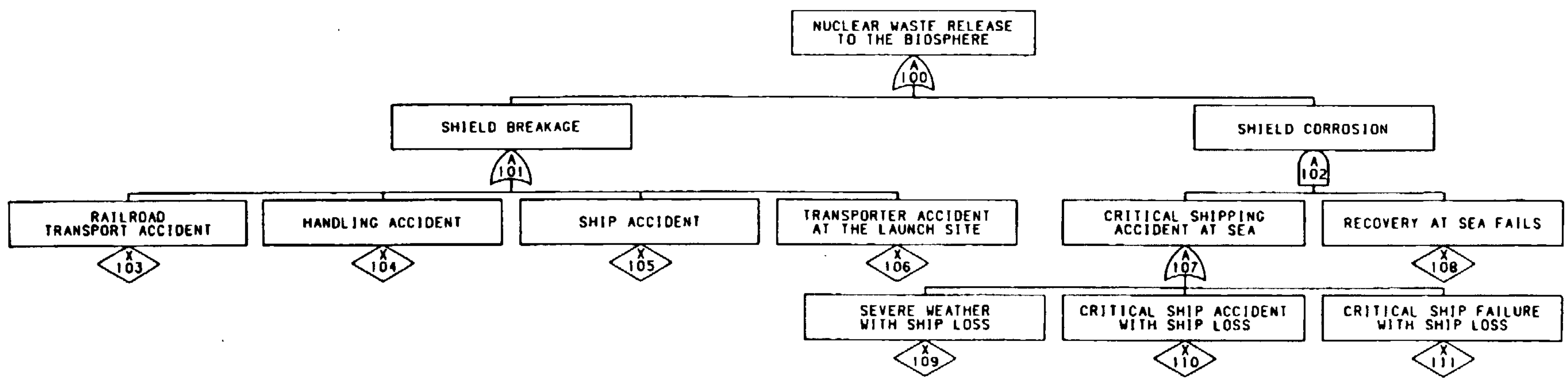

FIGURE 5-1. PHASE 1 FAULT TREE FOR TERRESTRIAL TRANSPORT

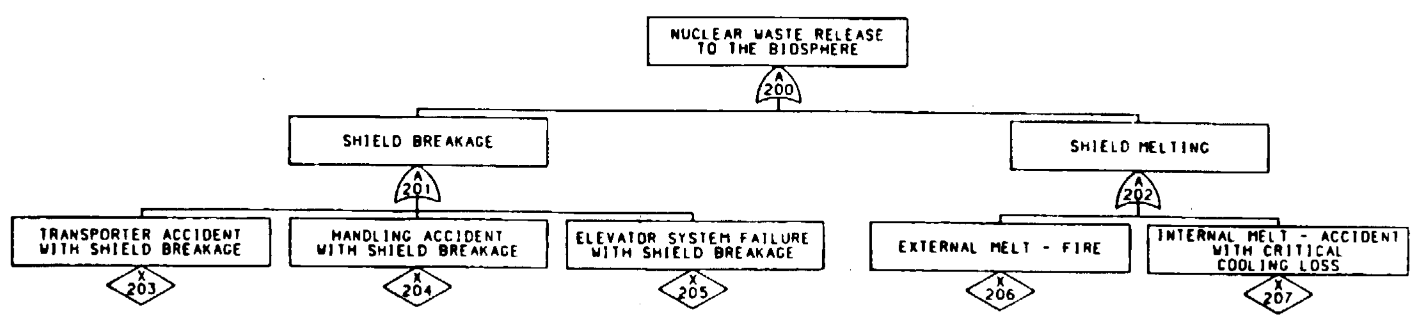

FIGURE 5-2. PHASE 2 FAULT TREE FOR PRELAUNCH ACTIVITIES

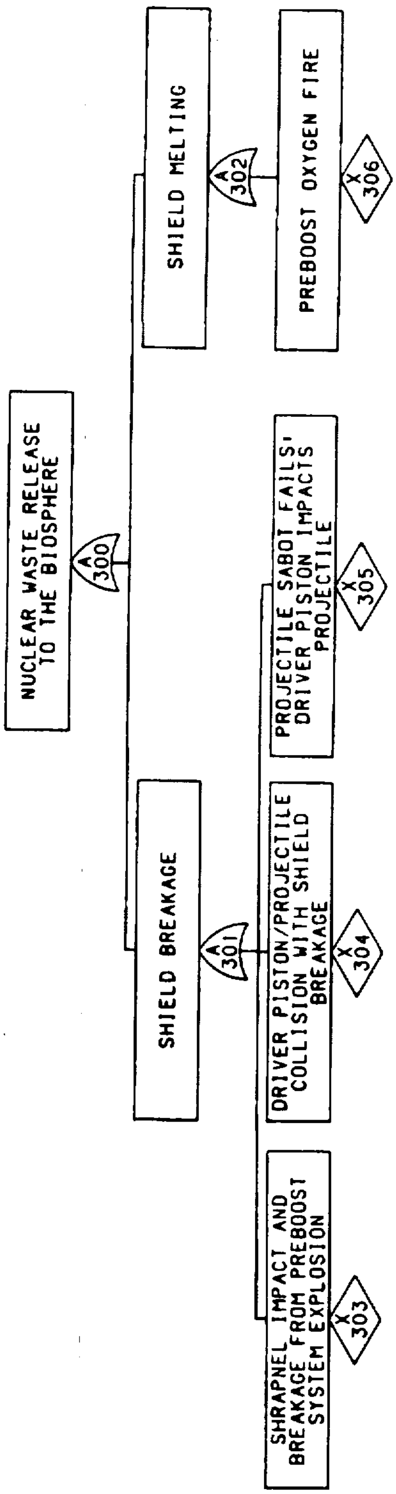

FIGURE 5-3. PHASE 3 FAULT TREE FOR PREBOOST

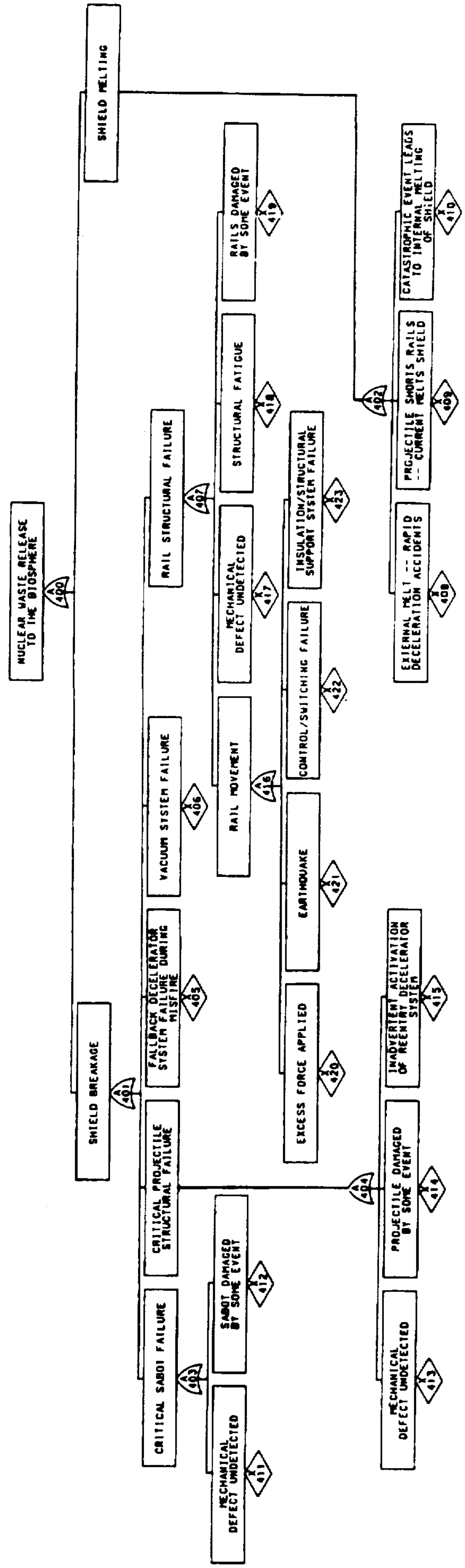

FIGURE 5-4. PHASE 4 FAULT TREE FOR ESRL ACCELERATION

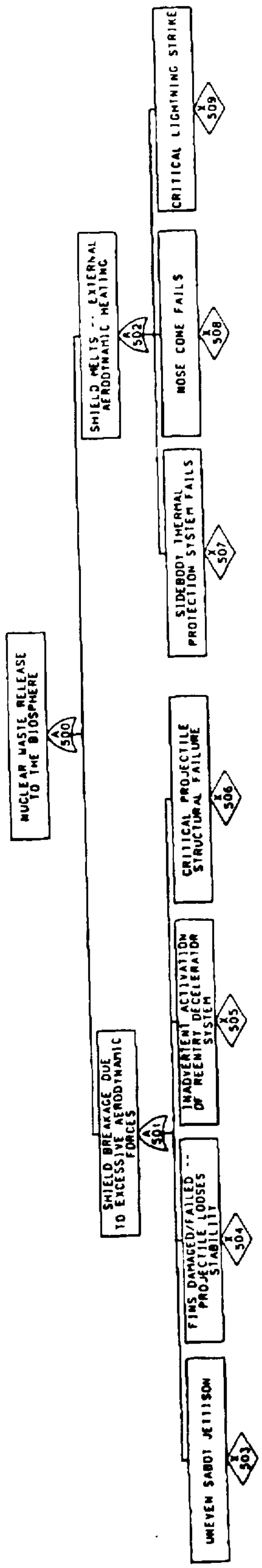

FIGURE 5-5. PHASE 5 FAULT TREE FOR SABOT JETTISON

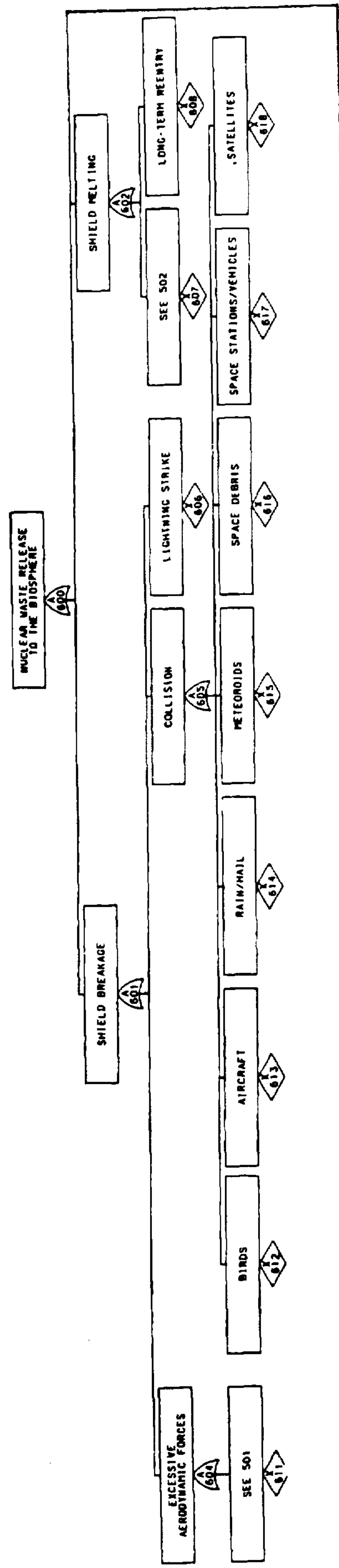

FIGURE 5-6. PHASE 6 FAULT TREE FOR ATMOSPHERIC AND SPACE FLIGHT

the internal heating rate is much reduced over that previously studied-Rice et al, 1980).

5.1.3 Phase 3--Preboost (Mission A)

Again, the two types of failure and release of nuclear waste to the biosphere relate to shield breakage and shield melting (see Figure 5-3). Shield breakage could be caused by: (1) shrapnel impact and damage from a preboost system explosion; (2) driver piston/projectile collision with shield breakage; and (3) projectile/sabot fails and the driver piston impacts the projectile. All of these events would occur in the rail launcher itself, therefore, any release of radioactive material could likely be contained and not released to the biosphere. There would be a very low probability of releasing material to the biosphere from these events. For shield melting, the major contributor would be expected to be an oxygen fire, related to the preboost system function, where oxygen actually burns away and melts the shield with the release of radioactive material in the launcher.

5.1.4 Phase 4--ESRL Acceleration (Mission A)

Shield breakage and shield melting could lead to nuclear waste release to the biosphere (see Figure 5-4). Shield breakage could occur from: (1) a critical sabot failure; (2) a critical projectile structural failure; (3) an event where the fallback decelerator system fails during a misfire; (4) the vacuum system fails; or (5) the rail structure fails under loads. The events that could lead to these failures are indicated in Figure 5-4. If the shield fails as it is being accelerated out the launcher, then it is possible that a significant release to the biosphere could occur. The amount of radioactive material released into the atmosphere would be a function of the velocity that the payload mass had achieved during the acceleration process. If the payload is held within the launcher tube, then it is possible to decontaminate the launcher tube without a significant release to the biosphere. Shield melting could occur: (1) during rapid deceleration accidents; (2) during short circuiting of the rails (through projectile); or (3) a catastrophic event leads to internal melting of a shield. Many of these melting type of events are directly related to events listed under shield breakage.

5.1.5 Phase 5--Sabot Jettison (Mission A)

During the period as the sabot/projectile leaves the rail launcher, shield breakage could occur due to excessive aerodynamic forces or the shield could melt due to external aerodynamic heating (see Figure 5-5). Excessive aerodynamic forces could occur if: (1) there is an unbalanced or uneven sabot jettison; (2) the aerodynamic fins are damaged or fail and the projectile loses its stability; (3) there is an inadvertent activation of the reentry decelerator system; or (4) there is a critical projectile structural failure causing a change in the aerodynamic characteristics. At this point in the evaluation, little can be said for the probability of these events in contribution to total release risk. The other potentially major contributions to the release risk during this phase relates to: (1) the failure of the side-body thermal protection system; (2) the failure of the nose cone to perform; and (3) a critical lightning strike. Without detailed analysis of all these aspects, little can be said for their contributions to total release risk.

5.1.6 Phase 6--Atmospheric and Space Flight (Mission A)

During this phase, shield breakage, shield melting and shield corrosion is possible (see Figure 5-6). Shield breakage can be caused by: (1) excessive aerodynamic forces during flight; (2) a collision with another object; or (3) a critical lightning strike. Collisions could involve birds, aircraft, rain, hail, meteoroids, space debris, space stations, space vehicles, or satellites. The collision probability of all these, except for meteroids and space debris could be adequately controlled by selection of launch time and appropriate launch constraints. The probability of collisions with meteoroids or space debris in the near-Earth vicinity is considered extremely small due to the fact that the residence time is extremely small (see Rice et al, 1982). Shield melting is also a possibility. It can be caused by: (1) a sidebody thermal protection system failure; (2) a nose cone failure; (3) a critical lightning strike; or (4) a payload reentry where payload does not escape the Earth or does not escape the solar system and it does not get rescued. Little can be said for the potential release risk of this event without performing additional analysis. The third contributor to release of nuclear waste to the biosphere during Phase 6 is shield corrosion. Short-term corrosion of the shield can occur from a mechanical or thermal failure which results in reentry of some kind, coupled with recovery failure. Also, long-term corrosion can occur, due to a misfire, where there is no critical damage to the shield and the short-term recovery activity fails to find the payload. Little can be said about the probability of these events occurring without a detailed systems analysis and additional technology work. However, the consequences of certain major events were evaluated and are discussed in the next section.

5.2 Preliminary Accident Response Analysis for Certain Major Failures (Mission A)

This section discusses the work done in assessing certain major accidents for Mission A (nuclear waste disposal in space). Because of the limited resources allocated for this activity and the complexity of the problem, emphasis was placed upon the corrosion/leaching and reentry problems.

5.2.1 Corrosion/Leaching Analysis

In the event of an ESRL accident where the waste payload ends up lost intact (or damaged) in the ocean, it is desirable to determine the time history of the radioactive release to the biosphere (see Events 102 and 603 in Figures 5-1 and 5-6, respectively).

One possible consequence of an ESRL launch deployment accident (misfire) is that the nuclear waste payload could return to the Earth's surface intact (i.e., without significant breakup) and be deposited in a "wet" environment, such as the ocean. For short-term accidents, the expected response would be to recover the payload from the ocean, but, if such recovery were to fail, then long-term radioactive releases would occur. Corrosion of the radiation shield barrier and subsequent leaching of waste form material represent a time-delay mechanism for eventual release of radioactivity to the biosphere.

For corrosion followed by eventual waste form leaching, the waste form is assumed to be packaged inside a highly corrosion-resistant shield of approximately 12 cm thickness. For purposes of analysis, it is assumed that the shield material would be selected to have a corrosion rate similar to Inconel-625, as assumed for standard space disposal concept--Rice et al, 1982. It is further assumed that waste form leaching does not begin until the shield is completely corroded away. A corrosion model is therefore quite simple, with the result stated in terms of the corrosion delay time equal to the thickness livided by rate of corrosion. The following table gives these data for the expected and bounded values of the corrosion rate for Inconel-625 (Rice et al, 1982).

	Corrosion Rate[a]	Corrosion Time, years
	0.01 mills/year = 2.54E-5 cm/year	472,000
expected	0.1 mills/year = 2.54E-4 cm/year	47,200
	0.3 mills/year = 7.62E-4 cm/year	15,800

[a]From Rice et al, 1982.

Note that even the shortest value of 15,800 years provides for a significant time for many of the isotopes in the cermet waste form to decay prior to release, via leaching, to the biosphere.

After the corrosion of the radiation shield, the waste form will begin to leach. Also, if the accident involves immediate breakage of the shield, sea water will enter and the leaching process will begin. Nominal leaching characteristics for the cermet waste form under evaluation have been estimated based upon discussions with DOE's waste form experts, although there is considerable uncertainty due to lack of experimental data for the specific physical and environmental conditions. The leach rate for cermet is estimated as 10^{-6} g/cm^2-day, with 90 percent confidence that it is within the range of 10^{-5} to 10^{-7} g/cm^2-day (Rice et al, 1982).

Consider the situations where (1) the shield has been breached and the radioactive waste can leach out directly into the ocean's biosphere, or (2) the shield corrodes over a long time. The only difference in these situations is the time of decay before release via leaching. The payload is cylindrical in shape with initial radius (r_o) and length (ℓ_o). To convert the area leach rate (L) given above to a mass loss rate (m), it is assumed that the cylindrical shaped waste form will reduce in proportion to its initial size, i.e.,

$$\ell = (\ell_o/r_o)r \qquad (1)$$

$$\dot{\ell} = (\ell_o/r_o)\dot{r} \qquad (2)$$

The mass loss rate can be stated in terms of the instantaneous surface area and the size/density parameters.

$$\dot{m} = LA = 2\pi r^2(1 + \ell_o/r_o)L \qquad (3)$$

$$\dot{m} = \rho\pi(2r\ell\dot{r} + r^2\dot{\ell}) = 3\rho\pi(\ell_o/r_o)r^2\dot{r} \qquad (4)$$

Equations (3) and (4) yields the constant value of $\dot{r}$ and the time for complete leaching.

$$\dot{r} = (2L/3\rho) \times (1 + r_o/\ell_o) \qquad (5)$$

$$t_L = r_o/\dot{r} \qquad (6)$$

For isotope (a) which has a half-life of h_a, the deposition rate to the biosphere is

$$\dot{m}_a = f_a(t)\dot{m} \qquad (7)$$

where $f_a(t)$ is the mass fraction of isotope (a) which exists in the leaching material:

$$f_a(t) = f_a(o)e^{-\lambda_a t} \tag{8}$$

where

$$\lambda_a = (\ell n\ 0.5)/h_a$$

Thus, one obtains

$$\dot{m}_a = f_a(o)e^{-\lambda_a t}\ \dot{m} = LAf_a(o)\left[(1 - \dot{r}t/r_o)^2\ e^{-\lambda_a t}\right] \tag{9}$$

Integrating Equation 9 from t' = 0 to t' = t, one obtains the cumulative release in grams of radioisotope a:

$$m_a(t) = \frac{LAf(o)}{\lambda_a}\left\{\left[1 - \frac{2\dot{r}}{r_o\lambda_a} + 2\left(\frac{\dot{r}}{\lambda_a r_o}\right)^2\right]\left(1 - e^{-\lambda_a t}\right)\right.$$
$$\left. + e^{-\lambda_a t}\left[\frac{2\dot{r}}{r_o}\left(1 - \frac{\dot{r}}{r_o\lambda_a}\right) t - \frac{\dot{r}^2 t^2}{r_o^2}\right]\right\} \tag{10}$$

Table 5-1 was constructed to display the quantities of the more hazardous isotopes in a 250 kg cermet payload (developed from data in Rice et al, 1982). Using these data coupled with half-life data for the various radioactive isotopes (Wang, 1969) and a cylindrical shape that matches the ESRL Reference Concept (Mission A) (see Section 4.0), the cumulative ocean releases as a function of time can be calculated. The results are given in Tables 5-2 and 5-3 plotted in Figures 5-7 and 5-8 for the cases of immediate leaching or delayed (by corrosion) leaching. Less than a 20 percent difference is apparent for cumulative releases of the two scenarios out to 1 million years. These releases are about a factor of 25 less than those for standard space disposal on a per mission basis.

5.2.2 Reentry Analysis

This subsection summarizes efforts in an attempt to predict the payload thermal response for projectile atmospheric reentry. Battelle's RETAC (Reentry Thermal Analysis Code) was used in an attempt to accomplish the thermal response analysis. High speed reentry cases were modeled, but because of the extreme conditions and coding, the computer program could not carry through the calculations. Although RETAC could have been modified, resources were not available to do so. The remainder of this section discusses the

TABLE 5-1. MASS AND CURIES OF 15 ISOTOPES IN ESRL WASTE FORM PAYLOAD AT LAUNCH

Isotope	Mass, kg[a]	Activity, Ci
Am-241	1.083	3,557
Am-243	1.262	241
Pu-238	0.0195	337
Pu-239	0.163	9.87
Pu-240	0.450	105
Pu-242	0.0273	0.107
Np-237	3.208	2.18
Ra-226	--	--
(AC)	(6.213)	(4,252)
C-14	0.001	2.96
Sr-90	0.034	4,680
Tc-99	3.843	67.8
Sn-126	0.103	2.89
Cs-135	0.065	0.082
Cs-137	0.089	7,733
I-129	--	--
(15 Isotopes)	(10.348)	(16,739)
Other Isotopes	151.939	248,337
TOTAL	168.500	269,328

[a] Based on 47.39 kg of waste form per 1 MTHM, and 250 kg per payload.

TABLE 5-2. CUMULATIVE OCEAN RELEASES (LOG_{10} CURIES) FOR ESRL ACCIDENT INVOLVING IMMEDIATE LEACHING OF REFERENCE CERMET WASTE FORM

Isotope	Years						
	1E0	1E1	1E2	1E3	1E4	1E5	1E6
Am-241	-1.43	-0.43	0.54	1.28	1.38	1.38	1.38
Am-243	-2.59	-1.59	-0.59	0.39	1.21	1.40	1.40
Pu-238	-2.46	-1.47	-0.61	-0.35	-0.35	-0.35	-0.35
Pu-239	-3.99	-2.99	-1.98	-1.00	-0.06	0.44	0.45
Pu-240	-2.96	-1.96	-0.96	0.02	0.82	0.98	0.98
Pu-242	-5.95	-4.95	-3.95	-2.95	-1.97	-1.14	-1.02
Np-237	-4.64	-3.64	-2.64	-1.65	-0.66	0.19	0.33
Ra-226	-10.07	-9.76	-8.76	-7.85	-7.40	-7.40	-7.40
(AC)	(-1.35)	(-0.35)	(0.61)	(1.36)	(1.68)	(1.80)	(1.81)
C-14	-4.51	-3.51	-2.51	-1.54	-7.60	-6.19	-6.19
Sr-90	-1.32	-0.36	0.27	0.31	0.31	0.31	0.31
Tc-99	-3.15	-2.15	-1.15	-0.15	0.83	1.63	1.73
Sn-126	-4.52	-3.52	-2.52	-1.52	-0.55	0.20	0.27
Cs-135	-6.07	-5.07	-4.07	-3.07	-2.08	-1.23	-1.09
Cs-137	-1.10	-0.14	0.50	0.54	0.54	0.54	0.54
I-129	--	--	--	--	--	--	--
15 Isotopes	(-0.76)	(0.21)	(0.96)	(1.47)	(1.78)	(2.05)	(2.10)

TABLE 5-3. CUMULATIVE OCEAN RELEASES (LOG_{10} CURIES) FOR ESRL ACCIDENT INVOLVING LONG-TERM CORROSION OF SHIELD FOR REFERENCE CERMET WASTE FORM

	Years					
Isotope	1E4	4.7E4	4.8E4	5.7E4	1.47E5	1.0E6
Am-241	0	0	-30.12	-30.01	-30.01	-30.01
Am-243	0	0	-1.54	-0.72	-0.52	-0.52
Pu-238	0	0	--	--	--	--
Pu-239	0	0	-1.60	-0.66	-0.15	-0.14
Pu-240	0	0	-2.17	-1.37	-1.21	-1.21
Pu-242	0	0	-2.99	-2.01	-1.18	-1.06
Np-237	0	0	-1.65	-0.67	0.18	0.32
Ra-226	0	0	--	--	--	--
(AC)	(0)	(0)	(-1.08)	(-0.17)	(0.42)	(1.17)
C-14	0	0	-4.02	-3.24	-3.10	-3.10
Sr-90	0	0	--	--	--	--
Tc-99	0	0	-0.22	0.76	1.56	1.66
Sn-126	0	0	-1.67	-0.69	0.06	0.13
Cs-135	0	0	-3.07	-2.09	-1.24	-1.10
Cs-137	0	0	--	--	--	--
I-129	0	0	--	--	--	--
15 Isotopes	0	0	-0.15	0.82	1.60	1.79

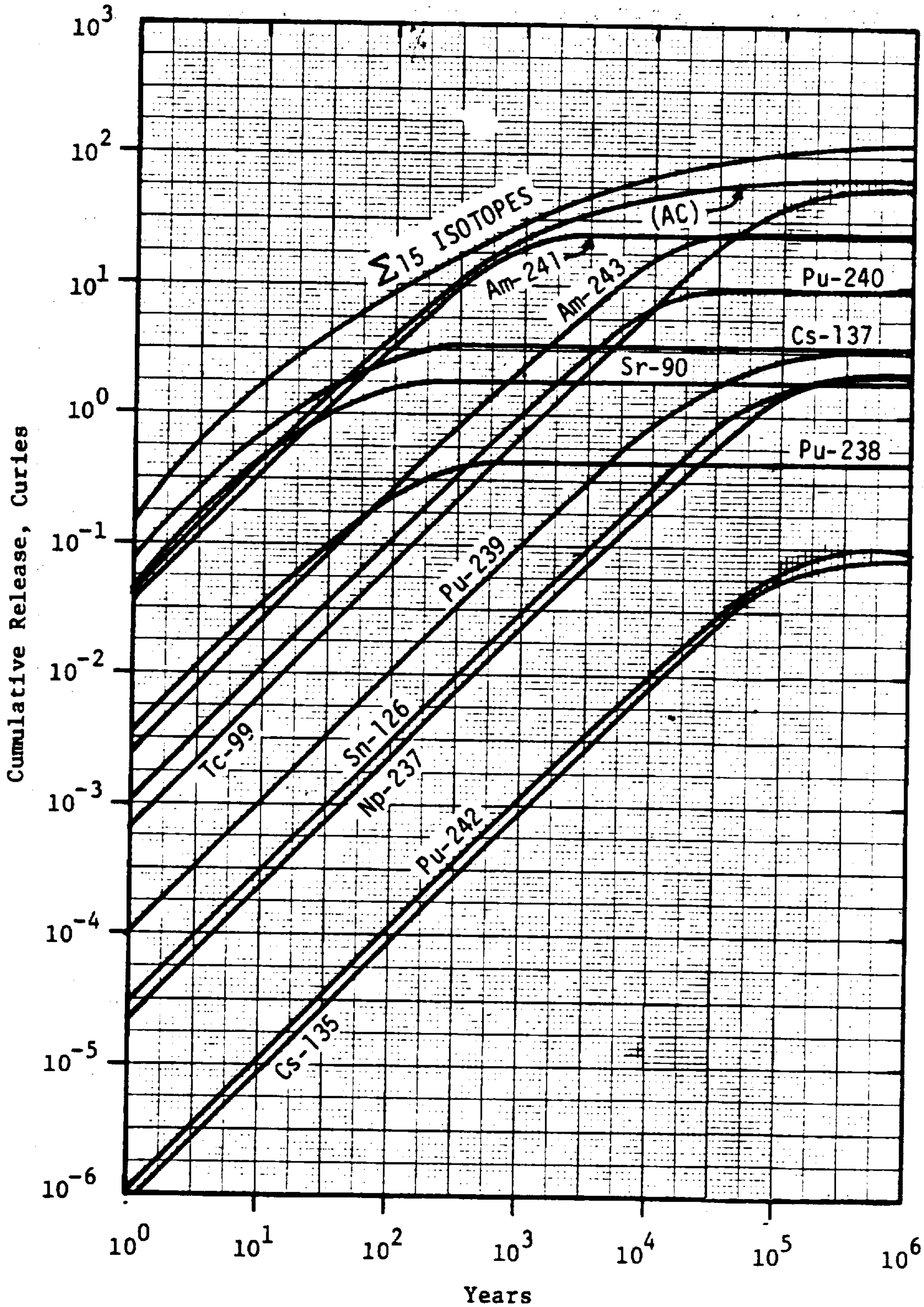

FIGURE 5-7. CUMULATIVE OCEAN RELEASES FOR ESRL ACCIDENT INVOLVING IMMEDIATE LEACHING OF REFERENCE CERMET WASTE FORM

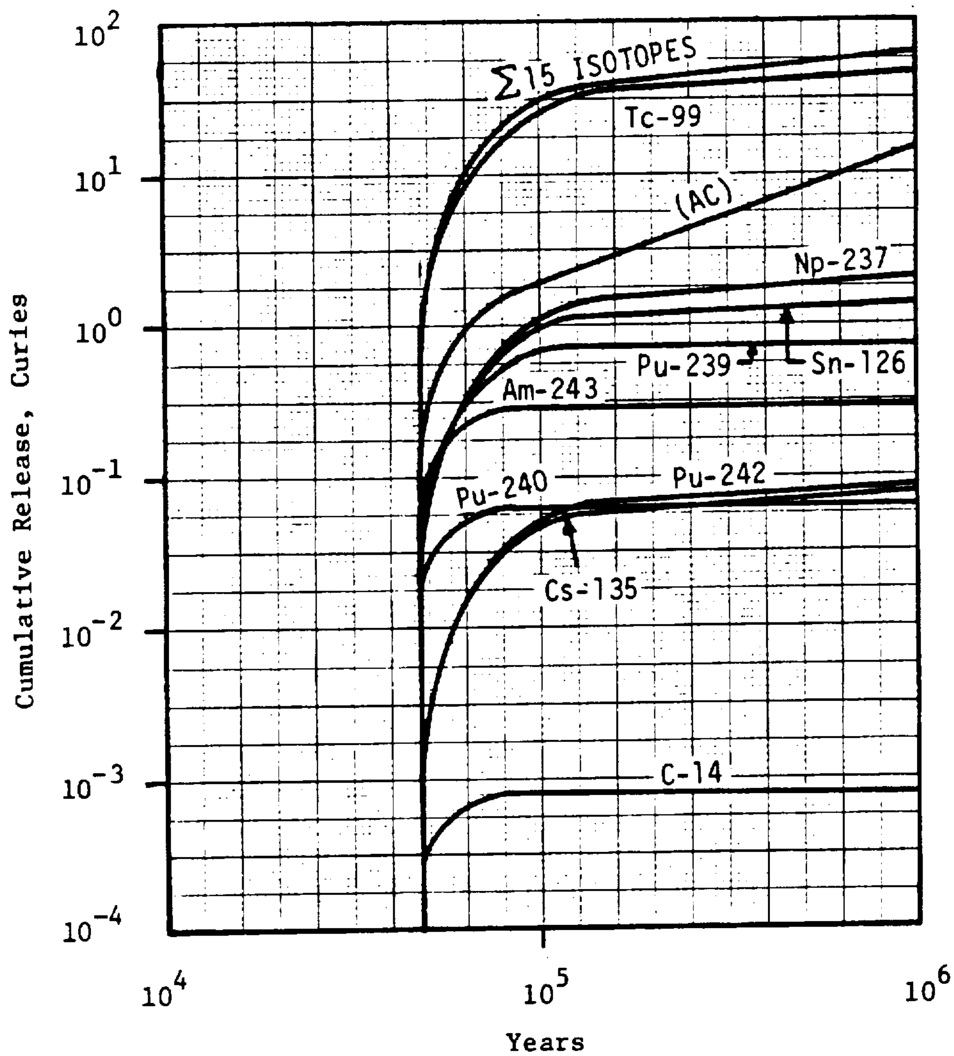

FIGURE 5-8. CUMULATIVE OCEAN RELEASES FOR ESRL ACCIDENT INVOLVING LONG-TERM CORROSION OF SHIELD FOR REFERENCE CERMET WASTE FORM

RETAC code and the reentry cases that were attempted. In Section 3.7, a hand calculation by Dr. John Lee, OSU, indicates that for stable, nose forward, normal flight, only about 1 cm of the tungsten nose tip would be lost. He has also calculated the ablation for a steel nose cone; this resulted in about 2 cm loss.

5.2.2.1 RETAC Code

The RETAC computer code includes a complex thermal response model for determining the in-depth response of a material system to an external heat flux. Furthermore, internal heat generation is provided for as a code input. The external flux variation with time can be specified in input cards (e.g., to model a launch vehicle fire environment) or be calculated by the codes trajectory subroutines (the aerodynamic flux due to a vehicle reentering the Earth's atmosphere). A detailed surface energy balance is included to account for re-radiation, conduction, and surface mass loss effects. The conductivity, specific heat, heat of fusion, heat generation and density of various internal and surface material components are also input to the code to model the complex response of the material components to the input and internal heat fluxes. Variations of the above material properties with temperature are also included where appropriate. RETAC has been used to model reentry of carbon/carbon radioisotope thermal generators (RTGs) and of nuclear waste spheres used in the standard space disposal concept (Rice et al, 1982).

5.2.2.2 Reentry Cases

Reentry cases of intrest are those where the projectile reenters in a stable flight condition at various steep angles and velocities. Also, of interest would be the steep reentry of damaged projectiles at various velocities. Our first attempt at reentry calculations involved the simulation of a non-rotating stable flight of an intact projectile with the characteristics the same as those of the Reference Concept for Mission A (nuclear waste disposal in space). The projectile was assumed to have a mass of 1915 kg (2055 kg less 140 kg sabot) and a reference cross-sectional area of about 2000 cm. Steep angles, 90° and high velocities, 20 km/s, were attempted with no success. The reentry parameters that produced some results were 60° reentry angle (from horizontal) at 10 km/s. The code cut off at about 30 km altitude, with total tungsten nose recession (to that point in the calculation) of 0.001 cm. Additional software work is required to modify the RETAC code to manage smaller time steps and code instabilities.

5.3 Environmental Impact Assessment of ESRL Concept

An in-depth environmental impact assessment of the ESRL Reference Concept was not possible under this study. However, it was possible to review the current Reference Concept to assess critical environmental impact areas to determine if there were any "show stoppers"; none were found.

The environmental impact assessment activities for the ESRL Reference Concept were broken down into four major categories. These are:

- Facilities development/construction
- Normal ESRL testing and operations
- Major accident events for Mission A (nuclear waste disposal in space)
- Major accident events from Mission B (Earth-orbital missions).

These are discussed in the sections below.

5.3.1 Facilities Development/Construction

The environmental impact for facilities development and construction are highly dependent upon the location at which the site is constructed. Without a candidate location (island) little can be said regarding specific environmental impacts. If a launch complex were placed on a remote island, the environmental impacts to the island could be significant; however, when weighed against launching from non-remote areas located in other parts of the globe, the overall impact from a a remote island-based facility to the quality of the human environment, would likely be less. Major impacts for launch site development might involve the following: the relocation of inhabitants (if present), the destruction of vegetation and wildlife habitats, the extinction of local animals species, and the possible disturbance of archeological sites. Site selection criteria for choosing the ESRL launch location could be used to minimize these effects to some degree.

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

- Launcher system
- Power plant
- Airfield
- Roads
- Buildings
- Housing.

The types of effects caused by the construction of the above-listed facilities are typical of any construction type activity in an undeveloped area. Unique aspects of the ESRL relate primarily to the construction of the launcher system. Large amounts of earth and stone would be removed and dumped in some location above ground. The construction of the two airfield runways could also pose significant environmental impact to the area. The construction of power plant roads, buildings and other housing is not expected to pose significant effects.

The question of materials usage also needs to be addressed. Table 5-4 lists some of the major specific materials that are required to construct the launcher system. Also shown in the table are the projected materials usages up to the year 2000 (Teeter and Jamieson, 1980). As one notes from the table, little impact on materials usage is predicted.

It can be seen from this table that major material requirements for development and construction of the ESRL launcher do not appear to be significant. The significant finding is that the large amounts of aluminum and copper that are believed necessary for the launcher do not significantly impact the total production, when compared to the annual consumption rates as shown in the table.

TABLE 5-4. MAJOR MATERIALS REQUIREMENTS FOR DEVELOPMENT/ CONSTRUCTION OF ESRL LAUNCHER

Material	Estimated ESRL Requirement, MT	Fraction of Annual U.S. Consumption by the Year 2000[a]	Fraction of Annual World Consumption by the Year 2000[a]
Aluminum	34,000	0.0018	0.00056
Copper	18,000	0.0033	0.00066
Iron	430,000	0.0029	0.00041
Cement[b]	800,000	0.0040	0.00056

[a]From Teeter and Jamieson, 1980.
[b]Assumed to equate to concrete one-to-one.

5.3.2 Normal ESRL Testing and Operations

This section discusses the expected environmental impacts from normal ESRL testing and operations. The Reference Concept was assumed in the evaluation, where two flights per day of nuclear waste disposal payloads and eight flights per day of Earth-orbital applications payloads are performed. Areas of concern relating to this particular impact area relate to the following:

- Sonic boom
- Power plant emissions
- Normal radiation doses to workers
- Chemical effluents
- Solid waste disposal
- Materials usage.

The following subsections discuss each of the above-mentioned concerns.

5.3.2.1 Sonic Booms

Because of the relatively large size of the projectiles, their extremely high velocity, and the assumption of some ten launches per day, there is concern for significant environmental impact from sonic booms. At the onset of the study, this area was believed to be a potential "show stopper." To calculate the magnitude of individual sonic booms, a relationship for the overpressure was derived.

The basic theory and origin of the sonic boom equation used in this assessment is based on a derivation by L. I. Sedov (1959) in his book Similarity and Dimensional Methods in Mechanics. Conceptually, the rising projectile is replaced by a line of blast wave energy which creates a shock wave of circular cross-section radiating horizontally. From consideration of the laws of conservation of mass, momentum and energy; and from a dimensional analysis of the physical phenomenon, Sedov deduces that the pressure rise across a strong cylindrical shock wave is given by:

$$\Delta p = \frac{E}{2\ (\gamma + 1)} \frac{1}{X} \qquad (1)$$

where:

Δp is the pressure rise
E is the energy per unit length of the disturbance source
γ is the ratio of specific heats, C_p/C_v
X is the radial distance from the disturbance line.

If the assumption is made that all the drag-loss energy of the projectile appears as wave drag, the total energy of the disturbance is equal to the product of the projectile drag and the vertical distance. So, the energy per unit length of the disturbance line is simply the projectile drag. If the ratio of specific heats is taken to be 1.4, the equation becomes:

$$\Delta p = .208 \frac{D}{X^2} \qquad (2)$$

But,

$$D = C_D \frac{\rho v^2 A}{2}$$, by definition of the drag coefficient

where

D = projectile drag
C_D = drag coefficient
ρ = atmospheric density
v = projectile velocity
A = cross-sectional reference area.

In terms of reference diameter, d,

$$D = \frac{\pi}{8} (C_D \rho v^2 d^2) \tag{3}$$

Substituting in the pressure-rise equation, we arrive at:

$$\Delta P = \frac{0.082 \rho v^2 C_D d^2}{X^2} \tag{4}$$

Using the relationship shown in Equation 4 above, limit distances for expected sonic boom overpressures for the Reference Concept missions are given in Table 5-5. Overpressure limits given in Table 5-5 were provided in CPIA, 1972. Sonic booms generated by Mission A are more severe than those from Mission B. Only Mission A will be discussed further. The critical distances from the launcher muzzle were calculated assuming a drag coefficient of 1.0 to represent the jettisoning process of the sabot in the early portions of the flight. Also, the diameter of the projectile was taken as the diameter of the sabot. (To truly represent a square-shaped sabot, the values for critical distances for Mission A and B should be increased by 13 percent). To discuss the overpressure limits in the table, the lethal limits means that if a person is standing at 30 m from the muzzle, that person is on the threshold of being killed. If the shockwave reflects off some structure/rock and then impacts the person death is likely. At 75 m distance from Mission A, eardrum rupture in an average human would be at the threshold. At 240 m from the launcher muzzle, window breakage would occur for typical glass. At 375 m from the launcher muzzle for Mission A, an overpressure of 0.138 N/cm^2 would be expected; this is typical for uncontrolled areas. At about 1.4 kilometers away from the launcher muzzle, the sonic boom would be approximately equivalent to the sonic boom generated by supersonic aircraft at a high altitude. Based upon the data shown in the table, one can conclude that payload designed building structures within about 100 m could survive repeated launches. People in that region should not be outdoors during launches, but should perhaps be 200 to 300 m away with ear protection. Environmental impact to local biology would likely mean that most of all animal life forms within 50 m of the launcher muzzle would be killed or forced to leave the area. Probably at distances of the order of hundreds of meters away from the launcher muzzle most animals would leave and seek other habitat. People living or working within several kilometer radius 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.

TABLE 5-5. LIMIT DISTANCES FOR EXPECTED SONIC BOOM OVERPRESSURES FOR REFERENCE CONCEPT MISSIONS

Type of Limit	Limit[a] Overpressure N/cm^2	Critical Distance from Launcher Muzzle, m[b] Mission A	Mission B
Lethal	20.7	30	15
Eardrum Rupture	3.45	75	40
Window Breakage	0.345	240	120
Typical-Uncontrolled Areas	0.138	375	190
Typical-Aircraft	0.010	1375	700

[a] From CPIA, 1972.
[b] Rounded to nearest 5 m; assumes: C_D = 1.0 (for sabot), v = 20 km/s (Mission A), v = 6.85 km/s (Mission B), d = 67 cm (Mission A), d = 100 cm (Mission B) and sea-level air density.

5.3.2.2 Power Plant Emissions

As indicated in Section 4.0, the power plant assumed for the Reference Concept is a nuclear facility. Normal emissions from nuclear reactors are not expected to pose a significant hazard to the environment. Accident risks from nuclear power plants located at the launcher site are not likely to be any different than nuclear power plants located elsewhere in the country. The total aspect of environmental impact of a power plant is probably less from nuclear than from coal or other fossil fuel power plants. It is concluded that emissions from a power plant are not a significant environmental impact for the ESRL Concept.

5.3.2.3 Normal Radiation Doses to Workers

Normal and expected radiation dose to both nuclear power plant workers and to handlers of Mission A payloads (nuclear waste disposal in space) are not expected to be any different than any nuclear operation conducted currently under guidelines provided by the federal government. This area is not a significant area of concern.

5.3.2.4 Chemical Effluents

Chemical effluents resulting from the ESRL testing and operations have yet to be identified. It is expected that various types of cleaning solvents and various propellant contaminants would be released into the biosphere, both air and water. These activities are not expected to be of any significance and are expected to be similar to those of current Space Shuttle launch activities.

5.3.2.5 Solid Wastes

Without knowing more about the ESRL operation, little can be said about the type and quantities of solid waste generated at the launcher site. It is, however, estimated that solid wastes would be expected to be similar to those of current Space Shuttle operation or industrial-type operations. No significant environmental impact is expected from the generation of solid waste produced by testing and operating an ESRL system.

5.3.2.6 Materials Usage

Materials consumed as a result of testing and operations of an ESRL system relate to all consumable materials and resources. Items include: (1) fuel rods for operating the nuclear power plant facility, (2) materials that make up the non-reusable portion of projectiles, (3) worn out components, (4) materials utilized to support transport activities of ships, aircraft, automobiles, and transporters, and (5) materials and supplies to support on-site personnel. Items of large quantity that are not considered as materials use include fresh water generated by salt water distillation, nitrogen generated by air liquefaction, and hydrogen and oxygen, as generated for the preboost system. These items are all generated on site by using excess power plant energy. Table 5-6 provides a brief comparison of ESRL materials usage to U.S. and world annual consumption in the year 2000. From the table, one can see that tungsten consumption on a yearly average, is a significant portion of the world and U.S. use. The manufacturing of ClF_3 and N_2H_4 will require significant upgrade to meet the demand. It must be pointed out that the use of these chemicals as propellants for the Earth-orbital applications mission is not critical nor is it expected that these propellants will remain as part of the Reference Concept. Therefore, tungsten represents the most critical of the materials that make up the annual operation and testing activity. Based on this, it is recommended that something else be used for the nose cone and fins for both the space disposal and the Earth-orbital mission applications.

5.3.2.7 Reentry of Hardware

During the normal ESRL testing and operational activities, the reentry of the nose cone and the payload support structure will occur for every mission. The proper landing area of these materials would allow minimal risk to the population, as well as the potential for recovery of the material from the bottom of the ocean, if economical.

TABLE 5-6. MAJOR MATERIALS REQUIREMENTS FOR ESRL TESTING AND OPERATIONS

Material	Estimated ESRL Requirement, MT	Fraction of Annual U.S. Consumption by the Year 2000[a]	Fraction of Annual World Consumption by the Year 2000[a]
Electricity, GW_e-hr	880	0.0001	0.00003
Tungsten	3420	0.15	0.043
Iron	9930	0.00007	0.00001
ClF_3[b]	2480	(c)	(c)
$-Cl_2$	950	0.00003	0.00001
$-F_2$	1530	0.0009	0.0002
N_2H_4	876	0.097[d]	(c)
He	5.8	0.0009	0.0002

(a)From Teeter and Jamieson, 1980.
(b)ClF_3 is made directly from Cl_2 and F_2, plant capacity would have to be expanded to support ESRL.
(c)Data not available in Teeter and Jamieson (1980).
(d)1963 basis, from Faith, 1965.

5.3.3 Major Accident Events for Mission A

There are many possible accidents that could occur from Mission A. Section 5.1 identifies many different types of events that could occur which could cause a release of nuclear waste material to the biosphere. Probably the two most significant accidents of global nature relate to the upper atmospheric burnup of a nuclear waste payload and the long-term corrosion of a lost payload in the ocean. Assessing the risk for an ESRL launched nuclear waste disposal in space traffic would require better concept definition before anything meaningful could be said about the risk.

For upper atmospheric burnup and dispersion, as a result of a reentry event, where the release of material occurs above 21 kilometers altitude, the worldwide dose for 1 micron sized particles can be estimated based on information in Rice et al, 1980. Assuming 250 kilograms of waste is dispersed per event, it is estimated that the world lung dose would be 4.2 million manrems; the worldwide bone dose would be 3 million manrems and the world total body dose would be 0.3 million manrems. Based upon dose factors also provided in Rice et al, 1980, this hypothetical worst-case accident scenario could result in some 30 cancer deaths throughout the world (these would not be measureable). Because the ESRL nuclear waste disposal in space mission contains only about 1/25 of that of that "standard" Shuttle-based disposal mission, the ESRL accident, should it occur, is believed to be of less consequence.

Results of the corrosion and corrosion/leaching events were previously discussed in Section 5.2.1. Again, the most that could be said for these events would be that the total expected cumulative release to the biosphere would be about 1/25 of that of a nuclear waste disposal via the Shuttle.

In a recent study by Rice, et al, 1982, preliminary estimates of cumulative release risk to the biosphere for standard Shuttle-based nuclear waste disposal in space was estimated. Figure 5-9 provides a summary of the expected release risk as a function of time for high-level waste disposal. The graph on the right hand side of the figure indicates the expected release risk in Curies of the sum of 15 isotopes. Based upon the risk level shown in the graph and the fact that ESRL payloads are likely to be harder to recover from the ocean, it is estimated that the rail launcher disposal system would have to be between 99.9 to 99.99 percent reliable, assuming an eventual 100 percent release, if not recovered from the ocean. This assumes that no other accident is possible other than than long-term corrosion and leaching in the ocean.

5.3.4 Major Accidents and Events for Mission B

The major accident events for Mission B that could pose significant hazard to the human population or to the biosphere would be the atmospheric payload breakup and reentry along with the possibility of propellant spills at the launch site. Atmospheric payload breakup and release of material into the atmosphere is believed to be no more hazardous than the current use of expendable or reusable launch vehicles where a considerable amount of toxic propellants are carried up through the atmosphere. For the Reference Concept for Earth-orbital applications, the propellants currently used are extremely toxic. The threshold limit value (TLV) for CLF_3 is 0.1 ppm. For hydrazine, the TLV is also 0.1 ppm. Because the Earth-to-space rail launcher for Earth applications has a zero degree inclination launch azimuth, the world's human population is hardly exposed to any threat because of the the overflight patterns for zero degree inclination orbits.

For toxic propellant spills at the launch site, significant care must be taken to avoid hazardous exposure to workers and the local uncontrolled human population.

5.3.5 Concluding Remarks

Based upon this preliminary environmental impact assessment for the ESRL Reference Concept, no significant environmental impact problems have been found. Sonic boom would create localized problems for animals surrounding the constructed rail launcher system, however, little effect is expected on the human populations. Another area of potential concern relates to the consumption of tungsten. It may be feasible that other metals or high-strength steels could replace the tungsten material in the nose cone and fins to reduce the overall impact on materials consumption. The initial construction of facilities is expected to create some environmental impact to the local area,

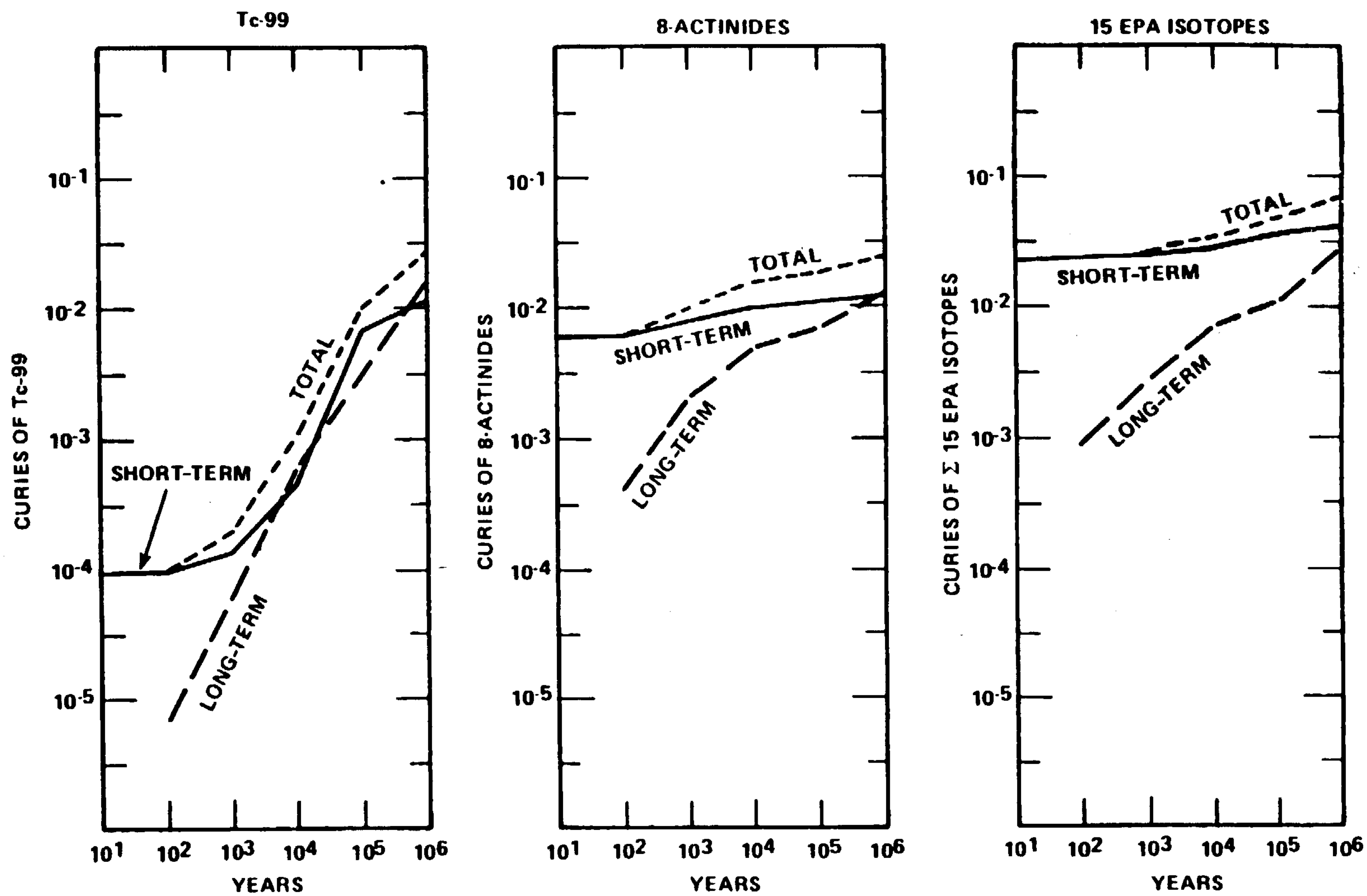

FIGURE 5-9. EXPECTED CUMULATIVE SPACE RISKS FOR HLW DISPOSAL IN SPACE FOR STANDARD SHUTTLE-BASED DISPOSAL (FROM RICE ET AL, 1982)

however, this is not expected to be significant. If one were to compare the environmental impact of the rail launcher system to that of the Space Shuttle on a per kilogram (payload) basis, it is expected that the rail launcher system may have less detrimental effects to the environment than Shuttle operations.

In conclusion, environmental impact benefits are perhaps possible by using the ESRL concept to carry out some space missions; however, this benefit should not be a driving force. The environmental impact benefits are not that significant. No "show stoppers" have been found thus far in the environmental impact evaluation. Economics appears to be the most important non-technical issue.

6.0 ESRL COST ESTIMATES

Costs for the Reference ESRL Concept (see Section 4.0) have been estimated according to the Work Breakdown Structure (WBS) shown in Tables 6-1 and 6-2. It should be noted that work breakdown structures are usually tailored toward accounting systems, rather than technical analysis of the particular system or its components. The WBS developed and used here also provides a preliminary estimate of the range of costs for the ESRL system concept. All costs are presented as 1981 dollars.

It should also be recognized that the ESRL WBS does not include the research, technology development, and design efforts needed prior to initiation of an ESRL development activity. There are two major reasons for not including these costs: (1) some costs may not be paid by the program as the research may be pursued by others; and (2) advanced research and technology development costs are highly uncertain. The research and technology development costs required prior to initiating ESRL system development are expected to be of the order of ten percent of the initial ESRL development and investment; the 90 percent confidence range on this expectation is from 5 to 25 percent. (Research and technology development requirements are discussed in Section 7.0.)

The development and operations costs for the current ESRL concept are more readily determinable, since much of the cost is concentrated in facilities which are expected to be built utilizing existing technology. Unique hardware items, such as homopolar generators (HPG's) have sufficient development history that estimates can be made by analogy to comparable hardware systems. The cost estimates provided here include: (1) systems development and construction; (2) initial flight test program; and (3) operations. A cost summary section presents an overview of the costs developed here, and provides estimates on the cost per unit mass for the space delivery missions considered.

6.1 Development and Investment Cost Estimates

The following subsections discuss how the development and investment cost estimates were assembled for the categories: (1) facilities and supporting systems and (2) the rail launcher systems. Low, expected, and high cost 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.

6.1.1 Facilities and Supporting Systems

Seven basic categories have been identified under this cost category: (1) land; (2) power plant; (3) personnel support facilities; (4) shipping docks, storage, and transportation facilities; (5) airfield and hanger; (6) industrial area; and (7) administrative/engineering buildings. These cost categories are discussed below.

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

1.0 Facilities and Supporting Systems

- 1.1 Land
- 1.2 Power Plant
- 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 Rail Launcher Systems

- 2.1 Tunnels/Shafts
  - 2.1.1 Nuclear Waste and Planetary Probe Launcher
  - 2.1.2 Earth Orbital Launches
  - 2.1.3 Elevator Shafts/Access Tunnels (including Storage/Work Facilities)
- 2.2 Launcher Tubes
  - 2.2.1 Copper Alloy (rails)
  - 2.2.2 Rail Spacers-Insulation
  - 2.2.3 Kevlar Containment
  - 2.2.4 Vacuum Container and Exterior Insulation
- 2.3 Homopolar Generators (includes hydraulic motors and hydraulic distribution) and Supporting Structures
- 2.4 Inductors and Switches (includes LN_2 distribution system)
- 2.5 Gas Injection Systems
  - 2.5.1 Preboost System
  - 2.5.2 Safety Deceleration System
- 2.6 Power Plant to Homopolar Generator Power Conversion Facility
- 2.7 Water Distillation Plant
- 2.8 Gas Handling Facilities
  - 2.8.1 Liquid Nitrogen Plant and Storage
  - 2.8.2 Vacuum System for Launcher Tube
  - 2.8.3 Water Electrolysis Plant
- 2.9 Elevator Systems and Projectile Handing Devices
- 2.10 Control Center, Controls, and Monitoring Systems
- 2.11 Tracking Systems
- 2.12 Accident Recovery Systems
  - 2.12.1 Ship
  - 2.12.2 Submersible

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

3.0 Projectiles and Mission Peculiar Equipment

3.1 Nuclear Waste Disposal Mission

3.1.1 Payload
3.1.2 Radiation Shield and Structure (Primary)
3.1.3 Nose Cone
3.1.4 Thermal Protection System
3.1.5 Instrument Package
3.1.6 Fins
3.1.7 Sabot(s)
3.1.8 Auxilliary Radiation Shields and Specialized Equipment
3.1.9 Transportation Costs

3.2 Planetary Probe Mission

3.2.1 Payload
3.2.2 Structure
3.2.3 Nose Cone
3.2.4 Thermal Protection System
3.2.5 Projectile Instrument Package
3.2.6 Fins
3.2.7 Sabot(s)
3.2.8 Auxilliary Equipment (Handling Equipment)
3.2.9 Transportation Costs

3.3 Earth Orbital Missions

3.3.1 Payload
3.3.2 Structure
3.3.3 Nose Cone
3.3.4 Thermal Protection System
3.3.5 Projectile Instrument Package
3.3.6 Fins
3.3.7 Sabot(s)
3.3.8 Auxilliary Equipment (Handling Equipment)
3.3.9 Propulsion System and Propellants Including Instrument Package
3.3.10 Transportation Costs

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)

TABLE 6-2. (Continued)

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
  - 4.3.6.1 Nuclear Waste Mission
  - 4.3.6.2 Planetary Probe Missions
  - 4.3.6.3 Earth Orbital Missions
- 4.3.7 Facility Utilities Crew
- 4.3.8 Accident Recovery Team

5.0 Development Test Program

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

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

	Low	Expected	High
Facilities and Supporting Systems			
Land	9.6	12.0	16.0
Power Plant (200 MW_e)	215.0	240.0	260.0
Personnel Support Facilities	50.0	77.0	115.0
Shipping Docks, Storage, Transport	20.0	50.0	100.0
Airfield and Hanger	26.3	56.0	100.0
Industrial Area	40.0	60.0	80.0
Administration/Engineering Buildings	5.0	5.0	5.0
Subtotal	(365.9)	(500.0)	(676.0)
Rail Launcher Systems			
Tunnels/Shafts			
- Mission A Launcher	148.0	223.0	390.0
- Mission B Launcher	148.0	222.0	390.0
- Elevator Shafts/Access Tunnels	116.0	173.0	300.0
Tubes- Copper	65.0	81.0	146.0
- Spacers	23.7	47.4	79.0
- Kevlar	127.2	259.4	389.0
- Vacuum Container	6.0	12.6	28.2
Homopolar Generators			
- Generators	1,176.0	1,499.0	1,885.0
- Support Structures	337.0	488.0	561.0
Inductors and Switches	420.0	536.0	637.0
Gas Injection Systems			
- Preboost System	100.0	125.0	150.0
- Safety Deceleration System	1.0	2.0	3.0
Power Conversion Plant	25.0	48.0	70.0
Water Distillation Plant	5.0	5.0	5.0
Gas Handling Facilities			
- LN_2 Plant and Storage	34.0	42.8	80.0
- Vacuum System for Launcher Tube	1.0	1.5	1.5
- H_2O Electrolysis Plant	0.2	0.3	0.4
Elevator Systems and Handling Devices	40.0	140.0	328.0

TABLE 6-3. (Continued)

	Low	Expected	High
Control Center, Controls, Monitoring Systems	20.0	100.0	224.0
Tracking Systems	10.0	100.0	200.0
Accident Recovery Systems			
- Ship	30.0	40.0	70.0
- Submersible	5.0	10.0	20.0
Subtotal	(2,831.1)	(4,156.0)	(5,957.1)
Total Development/Investment	3,204.0	4,656.0	6,633.1

6.1.1.1 Land

The ESRL Reference Concept proposes to use a remote island near the equator. Since there are few U.S. territories fitting this requirement, land for a base would probably have to be acquired by treaty with another country. The treaty could provide for a lump sum payment and an annual payment. If the ESRL system were to be an international facility or the host country were to receive some other benefit, such as some free use of the system, there might not be a cash payment for use of the land. However, for this cost estimate, the value of land is estimated at \$2470 per hectare (\$1000 per acre), which is in line with moderately expensive agricultural land. For cost estimating purposes, the ESRL facility is expected to occupy from 24 to 40 km^2 (15 to 25 mi^2) or 3885 to 6475 hectares (9600 to 16,000 acres). The estimated cost would then range from \$9.6 M to \$16.0 M, if there is any land cost. The expected cost is estimated at \$12 M.

6.1.1.2 Power Plant

It is assumed that a nuclear power plant would be selected. While the initial capital investment for a coal or petroleum plant is expected to be lower, the cost of fuel and shipping the fuel to a remote site for a thirty year life of the plant is expected to exceed the undiscounted cost of providing a nuclear power plant. The power requirements identified in Table 6-4 indicate that 50 MW_e would permit two Mission A launches per day (considering launch windows). Two-50 MW_e reactors will easily permit two nuclear waste launches per day, as well as a number of Earth orbital applications launches (Mission B).

A 200 MW_e coal or oil power plant, as a substitute for the nuclear plant capacity suggested here, would be expected to require approximately 5,000 MT of coal or 8,000 barrels of oil per day of full operation. If coal or oil in appropriate quantity and quality were located on or near the island, the selection of a nuclear plant might change.

An additional two or three reactors are needed as a maintenance reserve--civil reactor availability runs 66 percent due to both scheduled (about 20 percent) and unscheduled (about 14 percent) maintenance requirements. Thus, with four reactors, the basic power needed for operation of the facility would be available at least 98 percent of the time $(1-(1-0.66)^4 = 0.987)$. In addition, the usual availability of additional power will permit scheduling use of the two launchers fairly close together. If additional power is determined to be necessary, replication of the design is available at a reasonable additional cost.

Naval and other small nuclear power reactors are believed to be in the desired range of 50 to 100 MW_e. For economic reasons, civil power reactors are larger than 200 MW_e. While little information is available on the Naval designs, they have been proven in decades of reliable operation on ships and submarines and, accordingly are assumed possible for use in the ESRL

TABLE 6-4. ELECTRICAL POWER REQUIREMENTS FOR ESRL

Power Transfer Efficiencies:	Electric Generator to Hydraulic Facility:	0.98
	Electric to Hydraulic Conversion:	0.95
	Hydraulic to HPG Mechanical Conversion:	0.98
	HPG Mechanical to Inductor Electrical:	0.85
	Inductor Electrical to Rail Electrical:	0.85
	Net Efficiency (Product)	0.66

Projectile Energy Requirements: ($1/2\ mv^2$)

Mission A: $1/2 \times 2055\ kg \times (20{,}000)^2\ m^2/s^2 =$ 4.11×10^{11} Joules $\times\ 2.778 \times 10^{-7}$ kWh/Joule = 144,166 kWh = 114 MWh

Mission B: $1/2 \times 6500\ kg \times (6{,}850)^2\ m^2/s^2 =$ 1.53×10^{11} Joules $\times\ 2.778 \times 10^{-7}$ kWh/Joule = 42,363 kWh = 42.4 MWh

Electrical Work Requirements [($1/2\ mv^2$)/Efficiency]

Mission A: 114 MWh/0.66 = 173 MWh/shot
Mission B: 42.4 MWh/0.66 = 64 MWh/shot
Liquid Nitrogen @ 388 kWh/MT for 2500 MT = 971 MWh/day
Personnel: 1000 Workers x 5 kW x 24 hours = 120 MWh/day

Average Daily Power for Low Schedule (2 x Mission A + 2 x Mission B = 1094/24 h = 45.6 MW

Peak Power Requirements

Minimum Peak Power Requirement	Recharge Mission A Launcher in 4 hours + Homes 173 MWh/4 + 5 MW = 48.25 MW
Expected Peak Power Requirement for 2 x A + 8 x B	Recharge Mission A Launcher ONCE in 4 hours AND Recharge Mission B Launcher TWICE in 4 hours (173 + 2 x 64)/4 h + 5 MW = 301/4 + 5 = 80.25 MW
Potential High Volume Peak Power Requirement	Recharge Mission A AND Mission B Launchers in ONE hour 173 + 64 + 5 MW = 242 MW

Power Cost Estimates

Unit Costs, Installed:	\$1080-\$1300/kWh (Expected: \$1200/kWh)
200 MW_e, Installed:	\$215-\$260 M (Expected: \$240 M)

system. For power plant costs, a survey of civil power plants (Electrical World, 1981), gives a value of $922 per installed kilowatt for a recently completed nuclear plant. Because civil nuclear plants are typically built over a period of at least 5 to 10 years, the quoted cost has been adjusted upwards by 30 percent to reflect the effect of inflation during construction that would impact a plant being started, rather than completed, at the present time. This results in an estimate of $1200 per installed kilowatt.

Because the regulatory environment for a remote island plant is expected to be significantly different than for a typical U.S. commercial plant, this plant may be built and installed much more rapidly than has been shown in recent U.S. commercial experience. A savings of 50 percent appears to be possible based on comparisons between U.S. and Japanese nuclear power reactors. In this case, however, much of this savings may be consumed by the fact that subscale units have been selected. Thus, while the uncertainty in the ultimate cost is high, an uncertainty of 10 percent has been selected for purposes of the calculation and the low, expected, and high costs per installed kilowatt are estimated as $1080, $1200, and $1320. This leads to estimates of $215 M to $260 M as the cost range for a 200 MW_e nuclear power station, with $240 M the expected value.

6.1.1.3 Personnel Support Facilities

Personnel support facilities include such items as housing, roads, sanitation, and school buildings. It is assumed that there will be a permanent workers' community. Personnel facilities are estimated at $100,000 per worker. Since most work facilities are identified separately, these are not included in this estimate, but amenities for the worker's family have been considered. This results in estimates of $50 M, $77 M, and $115 M for estimates of approximately 495, 770, and 1145 operating personnel (see Section 6.2.2).

6.1.1.4 Shipping Docks, Storage and Transport Facilities

The cost of surface transport and storage facilities required will 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 can probably be supported by small ships, it is expected that a protected pier will be required for the construction phase. Accordingly, these facilities are estimated in the $20 M to $100 M range with a $50 M expected cost.

6.1.1.5 Airfield and Hangar

An airfield capable of handling the largest standard cargo aircraft will be required, together with facilities for refueling and aircraft maintenance. Two 3,000 meter runways (with taxiways), one full-shelter hangar and a fuel depot are envisioned. The local geography will be the major determinant of the airfield cost. Costs for construction could vary by as much as a factor of ten depending upon topography of the site. The estimates made here cover factors of one to three to reflect the cost of grading land which is easy to develop (factor of 1) to land requiring moderate contouring to achieve an acceptable grade (factor of 3). Runways are expected to cost in the range of \$3000 to \$5000 per lineal meter with minimal soil preparation, yielding an estimate of from \$18 to \$30 M for two 3000 m runways. A single hangar of 100 x 100 meters at \$450/m^2 will be \$4.5 M. A moderately sized fuel depot is expected to cost \$1 M. Other anciliary facilities are expected to cost \$1 M. Taxiways are estimated at 10 percent of the minimum cost for runways, or \$1.8 M.

The unknown factor for runway grading applies only to the runways as it is assumed that the land will be appropriate for the construction of the hangar and other facilities. Thus, the costs for the hangar, taxiways and fuel dump are estimated at \$8.3 M and the runways can range from \$18 to \$30 M to \$54 to \$90 M. The low estimate of the total is \$26.3 M. The expected estimate is \$56 M and the high estimate is \$100 M.

6.1.1.6 Industrial Area

Since the ESRL concept employs a considerable amount of moving machinery (such as homopolars, gas liquifaction compressors, etc), numerous maintenance and repair activities are anticipated. Thus, a facility which can repair and refurbish the ESRL equipment is needed. There will also be a need to store replacement hardware components in a warehouse. Because of the lack of knowledge 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 also most likely be accommodated in inexpensive buildings which can later be used to accommodate transient personnel during the operational phase. Since the initial motivation for construction of buildings would likely be the development test program and the ongoing use could be for personnel chiefly associated with applications, rather than launcher operations, an estimate of \$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 and 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 Rail Launcher Systems

6.1.2.1 Tunnels/Shafts

The costs of sinking shafts have been documented (STRAAM Engineers, 1978) for shafts up to 6.7 m in diameter, and for sinking and lining shafts up to 8.5 m in diameter by the Office of Nuclear Waste Isolation (Brown, 1980). These sources caution against extrapolating to greater diameters. All the costs quoted, however, are consistent with the cost per unit of depth being proportional to the diameter of the shaft, and approximately independent of the depth. The major sources of cost variance is due to the type of rock; basalt is approximately 1.5 times more costly to work than granite. As the location of the launcher is undetermined, the estimates made assume that a basaltic site would be selected, and that extensive lining of the shaft would not be needed. If extensive lining and equipment for water control is required, this would cost approximately 75 percent of the cost of sinking the tunnel.

Specific costs estimated for an 8.2 m diameter shaft in 1977 dollars per meter and adjusted to 1981 by the Consumer Price Index, are given below in Table 6-5 (Brown, 1980):

TABLE 6-5. COSTS ($, 1981) PER METER DEPTH FOR A 8.2 METER DIAMETER SHAFT

	Dollars					
	Salt		Granite		Basalt	
Years	Sinking	Lining	Sinking	Lining	Sinking	Lining
1977	24,850	16,060	16,530	560	24,210	1,110
1981	36,780	23,760	24,455	823	35,830	1,640

Source: Brown (1980); 1977 data modified by Consumer Price Index for inflation.

Accordingly, the following formulas have been developed for cost of shafts in 1981 dollars.

$$\text{Cost of Shaft } (\$, 1981) = (A)(24{,}455)(D/8.2)$$

where,

A = 1 for granite
A = 1.5 for basalt
D = Shaft diameter in meters.

The cost of a liner to control dust, particles, spall fragments and very minor water seepage is included. If major water control problems are likely, a full liner with a pumping system would be required. This cost is estimated as 75 percent of the tunnel cost.

$$\text{Cost of Liner (\$, 1981)} = 18,341\ (D/8.2) \qquad (2)$$

The current ESRL Reference Concept calls for shafts approximately 2040 m long for the two rail launchers, with approximately another 200 m for a preboost system, a loading platform, and some maneuvering room. In addition, there would be underground storage and work areas, as well as both personnel and freight elevators. The storage/work areas are believed to be approximated by an additional 500 m of the same diameter as the main launch shafts. The main elevator would be approximately the same length as the launcher itself. It is also anticipated that because of potentially hazardous cargo of all types, personnel and freight elevators will not occupy a common shaft. It is also anticipated that the homopolar generators (HPG's) would be installed and removed for major repairs through the launching shaft. This implies that the shaft would have a larger diameter than would be needed if the HPG's were either remote from the tube or installed from the bottom up and never needed to be removed. However, the latter is not considered an appropriate design. Because of the uncertainty in the HPG design concept, the diameter for the ESRL shafts is set at 20 m (65 feet) for costing. It is expected that the homopolar generators and inductors can be designed in such a manner that this is more than adequate for their installation and removal. Since the HPG's are expected to cost more than the shaft, it would be necessary to adjust the shaft design to accommodate them. The freight elevator shafts are expected to be 5 m (square) and personnel elevators are expected to be 3 m (square). Twenty-four service tunnels, 5 m square and 30 m long, totaling 720 m, and connecting the launcher shaft with the elevator shafts are also costed.

Accordingly, the two rail launchers plus 500 m of storage/access/work areas account for 4980 m of shafts at 20 m diameter. Basalt and no water problems are assumed for the nominal case, resulting in an estimate of $445 M for the main shafts. One freight elevator/service tunnel for each launcher plus 12 access tunnels of 30 m account for 4840 m of 5 m square tunnel and one passenger elevator/personnel access shaft for each launcher plus 12 access tunnels of 30 m account for 4840 m of 3 m square tunnels. The nominal case cost for the 5 m tunnels is $108 M, and for the 3 m tunnels, $65 M, for a total of $173 M. If water problems are encountered, an additional $334 M will be required for the rail launchers and $130 M for the elevators. Thus the nominal estimate for all the tunnels is $618 M (basalt, no water problems), the low estimate is $412 M (granite, no water problems) and the high estimate is $1080 million (basalt, water liner needed). This is summarized below in Table 6-6.

TABLE 6-6. COST ESTIMATES FOR SHAFTS AND TUNNELS ($, M, 1981)

Shaft and Tunnel Element	Low Estimate	Expected Estimate	High Estimate
Mission A Launcher	148	223	390
Mission B Launcher	148	222	390
Elevator Shafts/Access Tunnels	116	173	300
Totals	$412	$618	$1080

6.1.2.2 Launcher Tubes

The launcher tube in the ESRL Reference Concept has a square inner bore and an overall outer configuration that is circular (see Section 4.0, Figure 4-7). The bore heights used for cost analysis, 67 cm for the Mission A tube and 100 cm for Mission B. The launcher tube consists of a pair of rectangular copper-zirconium alloy (AMZIRC) rails, assumed to be 25 cm thick and 80 cm (Mission A) and 120 cm wide (Mission B). These are held in place by insulating spacers (and inductor contact leads) to fill out a circular tube (radius values assumed here are 1.0 m and 1.5 m, respectively). These spacers are confined by a Kevlar fiber winding estimated to be 5 to 10 cm thick. To provide vacuum containment, a sheath of aluminum, about 1 cm thick would probably be used.

Because the tube would be surrounded by homopolar generators, the cost of the supporting structure is considered with the generators and not the launch tube. It has been assumed here that active cooling of the rails is not needed. If this is not the case, and cooling with either water or liquid nitrogen is later considered necessary, the cost impact from associated changes in rail design, materials and fabrication is believed to be a relatively small problem.

6.1.2.2.1 Copper Alloy Rails

Based on the square bore design with rails assumed to be 25 cm thick and 80 cm or 120 cm wide, and an AMZIRC density of 8.96 g/cc, a Mission A rail would have a mass per unit length of 1792 kg/m, while the rail for Mission B would have a mass per unit length of 2688 kg/m. The rail pairs would have masses of 3.584 MT/m (Mission A) and 5.376 MT/m (Mission B). AMZIRC is estimated to be 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 (AMM, 1982), the rail pairs would cost $1.78 per kg and have costs per meter of $6,380 and $9,570, respectively for the two mission tubes. For the two launchers 2040 m long, the masses are 7,311 MT and 10,967 MT, the materials costs are \$13.0 M and \$19.5 M, with total masses of 18,278 MT and total materials costs of \$32.5 M.

It is expected that the rails will be cast, heat treated, and 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 the resultant expected cost is \$81 M. The low estimate is formed by selecting the labor factor of 2 and is \$65 M; the high estimate is formed by assuming a labor factor of 3 and a 50 percent increase in materials price or \$146 M.

6.1.2.2.2 Rail Spacers/Insulation

Fiber-reinforced rail spacers and insulating spacers, as shown in Figure 4-7, have been selected for the Reference Concept because they represent tested railgun technology. Their proportional size (as shown in Figure 4-7), however, has been reduced because of the significant cost impact on other subsystems. It is believed that the spacer and insulator size can be made smaller and still accommodate the voltages and mechanical stresses. The railguns developed in this country frequently employ Ferribestos, a material used in brake shoes as insulative spacers. Because of human health concerns relating to asbestos, it is likely that it would not be used for a large ESRL system. It is expected that a substitute would be found for asbestos and that this substance will cost less than \$1.00 per kg, installed. Even at this price, however, the amount required is sufficiently large that it is very likely that an alternative tube design such as that proposed by R. Hawke (see Section 3.4) would be selected to reduce the spacer/insulator as well as other costs while maintaining the same bore size. The engineering investigation to determine the best choice, however, was not possible within this study.

The cross-sectional area of the spacers for the Mission A launcher is 2.29 m^2 while that for the Mission B launcher is 5.77 m^2, representing radii of 1.0 and 1.5 m respectively with areas subtracted for the bores and rails. At spacer-insulator densities in the range from 3 to 5 g/cc (3 to 5 MT/m^3), the mass-per-unit-length ranges are respectively 6.87 to 11.45 MT/m and 16.38 to 27.3 MT/m. At an expected cost of \$1000/MT, installed, the spacers for 2040 m tubes are estimated to cost from \$14 M to \$22.3 M for the Mission A launcher and from \$33.4 M to \$55.7 M for the Mission B launcher. This design uses large quantities of materials which, even if inexpensive, drive requirements having significant cost impacts, such as the Kevlar winding. For this reason, it is likely that more advanced design such as that proposed by Hawke can be achieved. Therefore, the low estimate for both launchers is set at half of the expected cost calculated, or \$23.7 M. The expected and high estimates are then \$47.4 M and \$79.0 M.

6.1.2.2.3 Kevlar Containment

To hold the rails and spacer/insulator material in place against the pressures developed during launching, the rails and spacer/insulation material 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. Kevlar is made from two components, yarn and epoxy resin. The yarn is currently being sold in quantity at from $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 $28.20/kg of composite. Direct costs of labor to fabricate are given by DuPont personnel as being equal to material costs. Since the winding will have to be penetrated by inductor to rail conductors, it is most likely that a complex buildup pattern will be selected and a machine will be used to make the winding. For this type of operation, a direct labor factor of 2 times the material cost is appropriate and is used for the expected cost estimate. The lower cost estimate is formed by assuming that a 5 cm thickness will provide sufficient containment, and a labor factor of 2 is used; the expected cost estimate is formed using a 10 cm thickness and a labor factor of 2; the upper estimate uses a 10 cm thickness and a labor factor of 3. For these estimates, an insulator of 10 cm thickness is assumed to occupy the circular cross section from 1.0 to 1.1 m for the Mission A launcher and from 1.5 to 1.6 m for the Mission B launcher, both of which are 2040 m long. The estimates are given in Table 6-7.

TABLE 6-7. KEVLAR CONTAINMENT COST ESTIMATES

	Mission A Launcher Volume, m^3	Mission B Launcher Volume, m^3	Total Mass, MT	Materials Cost @ $28.20/kg	Labor Factor	Total Cost ($, M)
Low Estimate	656.9	977.3	2255.2	$ 63.6 M	2	$127.2
Expected Estimate	1345.8	1986.7	4598.9	$129.6 M	2	$259.4
High Estimate	1345.8	1986.7	4598.9	$129.6 M	3	$289.0

6.1.2.2.4 Vacuum Container and Exterior Insulation

An exterior container to provide some mechanical support, but to be used chiefly as a vacuum seal, is likely to be needed. The vacuum container is assumed to be aluminum, 1 to 3 cm thick; an inexpensive plastic covering, providing electrical insulation and assumed to cost about the same as aluminum, wou'd also be about 1 to 3 cm thick and have half of aluminum's density (1.35 g/cc vs. 2.7 g/cc). A nominal direct labor factor of 2 is used, and a labor factor of 3 together with a 50 percent increase in materials costs is used for the high estimate. Ingot aluminum currently costs $1.68 per kg (AMM, 1982) and plastics are available in the same price range. A uniform materials price of $2.20 per kg was used for both shields, at 2, 4, and 6 cm thickness. The Mission A launcher container cross-sectional area extends from a 1.05 or 1.10 m radius and the Mission B launcher cross-sectional area extends from a 1.55 or 1.6 m radius. Each launcher tube is 2040 m long. The cost estimates are given in Table 6-8.

TABLE 6-8. VACUUM CONTAINER AND EXTERIOR INSULATION COST ESTIMATE

	Thickness, cm	Mission A Launcher Volume, m^3	Mission B Launcher Volume, m^3	Total Mass, MT	Mat. Cost @ $2.20/kg	Labor Factor	Total Cost ($, M)
Low Estimate	2	271.7	399.9	1360.0	$3.0 M	2	$ 6.0 M
Nominal Estimate	4	574.2	830.5	2844.8	$6.3 M	2	$12.6 M
High Estimate	6	574.2	830.5	2844.8	$9.4 M*	3	$28.2 M

*$2.20/kg x 1.5 = $3.30/kg.

6.1.2.3 Homopolar Generators (HPG's)

The homopolar generators (HPG's) are estimated to be the largest source of uncertainty in the mechanical design of the ESRL. 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. The two launchers will require a total of 20,400 HPG's, each capable of storing a maximum operating energy level of about 56 MJ. Additional spares will be required, but there is no experience to indicate a reasonable level of spares. A nominal spare level of 10 percent is very high if the machines are as reliable as is needed for a system of such high loading. Five percent is considered to be reasonable and is used as the expected level, while 3 percent is used for a low estimate. This results in high, expected, and low estimates of the HPG population of 22,440, 21,420, and 21,012. To avoid heat buildup in the launcher shafts due to conversion of electrical energy into mechanical energy in the HPG's, it is likely that some form of conversion external to the launcher tube area will be desired. Hydraulic conversion was selected for this concept because the hydraulic fluid can carry away the waste heat. Similarly the electrical energy reconverted from the HPG mechanical energy will also result in heat which will need to be conducted away from the launcher, and depending upon the thermal design, a parallel cooling fluid system may be needed.

The preliminary estimates of the size of each 56 MJ HPG are 1.8 m in diameter, 1.5 m long, and a mass of about 10 MT. Between the HPG and the launcher tube is the inductor. The size of these devices is such that they would have to be arranged in a circle around the launcher tube. The tube and electrical devices will also need to be supported by massive steel support structures (see Figure 4-7).

Considering cost information on existing homopolar generators and possible production runs on the order of 10,000, R. Marshall indicates that the HPG's should cost between $1,000/MJ and $1,500/MJ, thus, a 56 MJ HPG should cost between $56,000 and $84,000, with an expected cost of $70,000. Based on a 56 MJ HPG weighing 10 MT, this is about $7.00/kg (range $5.60/kg--$8.40/kg), and consistent with automobile costs; ($7.00/kg for a small automobile). Based on the assumption that the HPG is a massive device with few moving parts, and that the production rate for the major procurement is reasonable, it is plausible that the costs could even be lower than $4.50/kg ($2.00 per pound). The major raw material, iron, costs about $0.50/kg in mill forms (AMM, 1982). Based on a maximum rim speed of 300 m/s, the brush contact speed believed to be reasonably achievable with acceptable life, the 1.8 m diameter implies a rotational velocity of about 50 revolutions per second or 3000 revolutions per minute. Accordingly, the bearings would be well within current technology. The major technical uncertainty with respect to this design is the achievable brush contact speed; 300 m/s is believed achievable, but will require some development effort. A lower brush contact speed of 220 m/s is considered current technology (W. F. Weldon, CEM-UT, a personal communication). For a fixed brush speed, HPG energy capacity scales linearly with mass. Thus, if a more conservative design were ultimately needed, the major requirement impacting cost is the need to use more iron, one of the least expensive materials available, in the rotor. Accordingly, the low, expected, and high estimates for HPG's and their supporting structures have been estimated as presented in Table 6-9.

TABLE 6-9. COST ESTIMATES FOR HPG's AND SUPPORT STRUCTURES

LOW ESTIMATE: \$1000 per MJ (\$5.60/kg) for 56 MJ per HPG for 21,012 units (3 percent spares) or \$1,180 M. The mass of the 20,400 HPG's to be installed will be 224,400 MT which is believed to require supporting structures of about the same mass at a cost of \$1.50/kg, installed, representing a labor factor of three. The structures cost is then \$337 M.

EXPECTED ESTIMATE: \$1,250 per MJ (\$7.00/kg) for 56 MJ per HPG for 21,420 units (5 percent spares) or \$1,500 M. The supporting structure is costed at \$2.00/kg, installed, representing a labor factor of four, and a cost of \$448 M.

HIGH ESTIMATE: \$1500 per MJ (\$8.40/kg) for a 56 MJ HPG for 22,440 units (10 percent spares) or \$1,885 M. The supporting structure is costed at \$2.50/kg, installed, a labor factor of five, and a structures cost of \$561 M.

6.1.2.4 Inductors and Switches

Preliminary calculations by Marshall (see Section 3.2 for discussion) indicate that, for the Reference Concept, the inductors must store approximately 48 MJ of energy at a current of 4 MA to achieve the reasonably 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, inductor 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 (see Section 3.2 for discussion). 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 most of the LN_2 used to cool it. Foamed insulation currently has problems with cracking and separation upon repeated cryogenic cycles, and research is being conducted in this area for application to reusable space vehicles. Thus, it is reasonable to expect that foamed insulation will be approprate at the time of implementation. Contained foam (preformed) insulation will always be available as a back-up technology. The major problems foreseen are electrical switching controls (low maintenance is an assumed requirement) and plumbing and venting for liquid and gaseous nitrogen. Switching, however, represents the major technology problem (see Section 3.2.4.3). Accordingly, the inductors are costed with a labor factor of 10 times the raw material price to reflect the uncertainty of the technology and to provide an allowance for plumbing complexity in the nitrogen distribution system. The current price for aluminum ingots is \$1.67/kg (AMM, 1982). 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 \$2,000 to \$3,000. 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. Thus, for 21,012, 21,420, and 22,440 inductor units, installed, the costs are estimated as \$420 M, \$536 M, and \$673 M.

6.1.2.5 Gas Injection Systems

Gas injection systems include the preboost system and the safety deceleration system. The cost estimates for these are given below.

6.1.2.5.1 Preboost System

A design for the preboost system is not developed, only a preliminary concept has been suggested (see Section 4.3.1.5); it is assumed that a hydrogen/oxygen driven piston system with a square steel barrel attached to the end of the rail launcher tube (approximately 200 meters long) would be used to accelerate the projectile at about 260 g's. The piston would drive a mixture of nitrogen and hydrogen gas which would in turn accelerate the saboted projectile. The cost of the propellants, given the availability of gas liquefication and water electrolysis plants, 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 two barrels can be accomplished for \$100 to \$150 M with an expected cost of \$125 M. The cost for only one barrel, however, would probably result in a reduction in these totals of only \$10 to \$20 M.

The reason for the high cost of a conceptually simple system is the need to inject a very large quantity of gas at very high pressure in a very short time (0.4 sec). Hydrogen and oxygen high-pressure rocket engine pumps, about the size of those on the Space Shuttle Main Engine (SSME), will be required to inject the liquids. These will have an operating time of only a few seconds per shot, so that service life will be very long (10 to 20 years), based on very 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 can reduce costs. Thus, while an SSME currently costs on the order of \$20 to \$30 M, many components, such as nozzles, engine mounts, etc., will not be needed. Thus, it is reasonable to expect that hardware components adapted to this task, including spares, can be purchased at the same time for both launchers at about the cost for one SSME at the present time. The design effort, however, will be significant and accounts for most of the costs estimated.

6.1.2.5.2 Safety Deceleration System

To prevent misfired projectiles from achieving velocities which would make their safe recovery difficult, it is considered likely that some type of gas injection system would be used to slow down projectiles when problems are detected early in the acceleration phase. If the problems are detected after more than the first 100 msec, however, the injection of gas might cause more of a hazard (launcher distortion) than it would prevent. Thus, the first decelerator system would have the ability to stop adding energy to the launcher for early misfires.

The second decelerator system would be used to cushion the fallback of a projectile on the launcher breech. The system envisioned would include gas injection ports, with rupturable diaphrams along the tube. These would permit high pressure nitrogen to enter the launcher tube and cause the projectile to decelerate before it impacts the breech. It could also retard the forward motion if the misfire occurred very early in the launch.

These two deceleration methods also would have a minimal cost implication and are included at a range of cost estimates of $1, $2, and $3 M.

6.1.2.6 Power Conversion Plant (Electrical to Mechanical)

To avoid heat buildup in the launcher shaft, and because motoring the homopolar generators would reduce brush life, conversion of electric to hydraulic power is envisioned to occur in a facility near the launcher tubes. It is also expected that hydraulic motors would save space in the launcher shaft, as well as being somewhat lower in cost than electric motors. While it is possible to transmit power in the form of steam from the reactors at the power plant, the transmission flexibility of electrical power suggests that it would be better to convert alternating (or direct) current from the nuclear power plant at a separate facility to drive the HPG's. This facility would be a pumping station with the pumps being driven by electrical motors. The conversion efficiency would be at least 95 percent, with the hydraulic fluid carrying away the energy loss from the reconversion to mechanical energy at the HPG's.

Other sources of heat in the launcher tube are expected to be conducted away by the residual heat capacity of the nitrogen used to cool the inductors. The thermal design balance is expected to be a complex problem which can be addressed only in detailed design studies. The goal in this assessment is to select low-cost options which do not complicate this problem.

Since it is desirable to be able to charge both launchers as rapidly as possible, conversion of the full power from all of the four reactors might be desirable. The calculations of Table 6-4, however, indicate that the peak power requirement for two Mission A plus eight Mission B launches is 80.25 MW, so the power conversion plant is sized at 100 Mw. It would be designed with several parallel units.

Since electric to high-pressure hydraulic conversions of this magnitude do not appear to have been undertaken previously, no good analogy is available to draw upon. It is expected, however, that the costs would run at about one-half those of coal-fired electrical plants, on a per-installed-kilowatt basis. These cost in the range of \$250/kW to \$700/kW. Accordingly, the low, expected, and high estimates for 100 MW are \$25 M, \$48 M, and \$70 M.

6.1.2.7 Water Distilling Plant

The 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 sea water distilling plant will probably be needed. The plant is sized at 2,000,000 liters per day, representing 400 liters per person per day for 5,000 people. This is expected to have reserve capacity for a crew of 1,000 with 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 \$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.

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 ESRL launcher tubes; and (3) hydrogen and oxygen storage for the preboost system. These are discussed in the following three sections.

6.1.2.8.1 Liquid Nitrogen Plant and Storage

To provide acceptable masses for the inductors, it will be necessary to drop the resistance of their conductive material 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 and, therefore, the costs are too uncertain to make reasonable cost estimates in the foreseeable future. The major uncertainties in selecting LN_2 cooling are the requirements for LN_2 due to insulative losses in the inductors and their plumbing and to the efficiency of transmitting the electrical power from the HPG through the inductor to the rails. This would involve both thermal and electrical losses, both 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, and an assumed 15 percent of input energy as a combined thermal and electrical inefficiency causing LN_2 boiloff, the requirement for LN_2 is estimated (see Table 6-10) at 1286 MT per day for both launchers at two shots per day per launcher. Six additional Earth Orbital launches would require a total of 2331 MT. To provide an additional margin, 2500 MT per day is costed. From information provided by J. Cost, Air Products Company, a plant providing 325 MT per day would cost \$4 M, and would scale upward by a 0.6 power law on cost, resulting in a cost estimate of \$13.6 M. The 0.6 power law is believed to be somewhat optimistic; if a 0.7 power is used, the plant cost would be \$16.7 M. Under the worst-case assumption that multiple independent units would be required, their cost would be \$32.0 M.

The plant will require 41 MW of electrical power for full-scale production. See Table 6-10 for calculations of the cooling and related power requirements.

Storage tanks needed for three days of reserves (8,700 m^3 at a density of 804 kg/m^3) are expected to represent an investment of about half that needed for the plant. The transport and venting lines are expected to represent an investment equivalent to that of the plant. Accordingly, if the low, expected, and high costs of the plant are \$13.6 M, \$16.7 M, and \$32.0 M, the LN_2 system cost estimates are then \$34.0 M, \$42.8 M, and \$80.0 M.

6.1.2.8.2 Vacuum System for Launcher Tube

At a length of 2,240 meters, a cross section of 1 meter or less, and an air density of 1.3 kg/m^3, the evacuation of the launcher tube to approximately 1/100 (7.6 mm Hg) atmosphere will require the removal of less than 1,300 kg of air for the Mission A tube (3,000 kg for the Mission B tube). 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 five pumps (two per launcher and one spare) 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 for both launcher tubes.

6.1.2.8.3 Water Electrolysis Plant

To provide hydrogen and oxygen for the preboost system, a water electrolysis plant is proposed. 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 liquefy the gases are provided within the estimate for the liquid nitrogen plant. In addition to these elements, there will 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 6-10. LIQUID NITROGEN AND ASSOCIATED POWER REQUIREMENTS

DETERMINING ASSUMPTION: ENERGY LOSSES AND THERMAL GAINS RESULTING IN LN_2 BOILING EQUAL 15 PERCENT OF ENERGY PER SHOT

Energy Requirements: Mission A: 173 MWh/Shot x 0.15 = 25,950 kWh/shot
Mission B: 64 MWh/shot x 0.15 = 9,640 kWh/shot

Heat of Vaporization of LN_2: 47.6 kcal/kg - 0.05534 kWh/kg
(860.1 kcal/kWh)

LN_2 Requirements: Mission A: 25,950 kWh/0.05534 kWh/kg = 468.9 MT
Mission B: 9,640 kWh/0.05534 kWh/kg = 174.1 MT

LN_2 and Related Energy Requirements

(LN_2, 388.3 kWh/MT power requirement)[a]

Launches	LN_2 in MT	Energy in MWh
Mission A	469	182.1
Mission B	174	67.6
2 x A + 2 x B	1286	499.3
2 x A + 4 x B	1634	634.5
2 x A + 6 x B	1982	769.7
2 x A + 8 x B	2331	904.9
Additional Margin	2500	970.8

(a)Telephone conversation with J. Cost, Air Products Company, Allentown, PA, January, 1982.

6.1.2.9 Elevator Systems and Handling Devices

Projectile handling devices, as contrasted to elevators of all types, are expected to be a very small portion of the materials handling requirement. Specific projectile and HPG handling equipment is expected to cost on the order of $15 M and this is added to each of the three estimates for elevators. The size of the homopolar generators and inductors, and the necessity to replace and service them, is envisioned as the major cost driver in materials handling equipment. The homopolar generators are envisioned as being arranged in columns around the launcher tube. The current design has structure between each two columns of homopolar generators. The ESRL Reference Concept calls for six elevators in hexagonal configuration (Figure 4-7).

In addition, there will be two independent freight elevators and, for safety, two independent passenger elevators. There will also be 180 m of access tunnels and 180 m of access elevators, all requiring some form of carriage. There are indications (personal communication with Mr. Minelt, Otis Elevator Company, March, 1982) that light elevators can cost as low as $500/m and the heavy elevators can be as low as $1250/m, resulting in a low estimate for all elevators and handling systems of $40 M. It is possible that the requirements were miscommunicated as being commercial/light industrial and more capable systems may be needed. For the expected and high cost estimates, higher costs are used since it may be desirable to load the elevator systems heavily by multiple cars. Thus, while the passenger elevators need not support heavy loads and can be furnished at about $1000/m, the freight elevators and launcher tube elevators must be able to carry at least 15 MT, representing the combined weight of a HPG/inductor plus personnel and handling equipment. Based on the current design, it would also be desirable to be able to transport a double load. These are expected to cost in the range of $2000/m to $5,000/m. Thus, there are 4,840 m of passenger elevators at $1000/m, or $4.8 M. The freight elevators, however, constitute 14 x 4480 m or 62,720 m at $2,000/m to $5,000/m. The expected and high estimates for these are formed by costs of $2,000/m and $5,000/m for the stated lengths, or $125 M and $313 M. The low, expected, and high estimates for elevators and projectile handling equipment are $40 M, $140 M, and $328 M.

6.1.2.10 Control Center, Controls and Monitoring Systems

A preliminary system design, as well as a specification of the control requirements, is needed before accurate control costs can be given. It is reasonable, however, to estimate that homopolar generators and inductors can probably be monitored and controlled by signals from the master control center for a relatively low cost. A tentative estimate of $1,000 to $10,000 per homopolar generator/inductor set is suggested. Since these are calculated to be from 20,400 to 22,440 homopolar generators in the two launchers, the control cost could range from $20 M to $224 M. The expected cost of $100 M has been selected.

6.1.2.11 Tracking Systems

A specialized tracking system is required to monitor the trajectory of launched projectiles. The adoption of a military tactical radar station is assumed. Costs in the neighborhood of \$10 M are suggested, but because tracking and communications requirements tend to grow, the expected and high-range estimates are set at \$100 M and \$200 M to reflect the establishment of a sophisticated tracking station for use with Mission A and B profiles.

6.1.2.12 Accident Recovery Systems

For recovery of nuclear waste disposal payloads, a ship with a submersible vessel capable of operating in deep water is envisioned. It is expected that most non-nuclear payloads would be abandoned if they abort and reenter in water. Nuclear waste payloads are expected to have survivable radio/acoustic transponders so that they can be located without sophisticated equipment. The recovery ship, which must be of a design which can withstand severe weather and manipulate a submersible, is costed at \$40 M and the submersible, assumed to operate by television and remote control, is costed at \$10 M. At present no other tasks are foreseen for these vessels. Given their expected infrequent use, it might be appropriate to commission an oceanographic research vessel with this recovery task as its emergency duty.

6.2 ESRL Operations Cost Estimates

The costs to operate an ESRL facility have two components: the recurring costs associated with each projectile fired, and the annual costs for personnel and supplies. Projectile costs are summarized in Table 6-11; Personnel and supply costs are summarized in Table 6-12.

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. Because the initial projectiles must be ready for testing at the conclusion of ESRL construction, 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 check out payloads, this would be a simple procedure. It is assumed that payloads would largely be manufactured, assembled, and checked out on the mainland. Only tasks such as loading of propellants and other fluids, initiation of guidance systems, and verification of status would normally be undertaken at the facility. Any repairs would be the responsibility of the mission program.

6.2.1 Projectiles and Mission Peculiar Equipment

The costs for projectiles are highly uncertain due to technological advances needed to achieve and demonstrate the capabilities required. In addition, the annual quantities required (thousands) are not large enough to indicate that major savings through mass production (in the manner of automobiles) can be achieved. The phenomenon of learning, nevertheless, would occur. The major problem is that aerospace quality materials must be developed, engineered, and have designs modified to meet changing requirements without any indication that the designs will be amortized over a very large number of launches. In a situation like this, the material costs tend to be a small fraction of the total costs, which are dominated by labor and overhead. The next three major sections discuss the costs for Mission A and Mission B projectiles and mission peculiar equipment.

6.2.1.1 Nuclear Waste Disposal Mission

6.2.1.1.1 Payload

The payload would be fabricated as a cermet waste form in a waste processing plant which separates the recyclable fuel from the material which is selected for space disposal. Since one of the products of this plant would be fuel which has economic value, it is likely that only the direct costs associated with manufacture of the waste form would be charged to the payload. Thus, payload costs are not expected to greatly exceed the cost of the projectile. Because payload costs are usually separated from launch costs for purposes of analysis, the payload manufacturing costs are not considered here.

TABLE 6-11. COST ESTIMATES FOR ESRL PROJECTILES ($, K, 1981)

WBS Element	Mission A						Mission B		
	Nuclear Waste Disposal Projectile			Planetary Probe Projectile			Earth-orbital Projectile		
	Low	Expected	High	Low	Expected	High	Low	Expected	High
Payload	--	--	--	--	--	--	--	--	--
Structure/ Radiation Shield	(10.0)(a)	(10.0)(a)	(100.0)(a)	50.0	100.0	200.0	10.0	10.0	100.0
Nose Cone(b)	30.0	50.0	80.0	40.0	80.0	160.0	40.0	185.0	250.0
Thermal Protection System	5.0	10.0	20.0	10.0	20.0	40.0	5.0	10.0	20.0
Instrument Package	1.0	2.0	2.0	1.0	2.0	2.0	(c)	(c)	(c)
Fins	1.0	5.0	10.0	1.0	5.0	10.0	1.0	5.0	10.0
Sabots	1.0	5.0	200.0	1.0	5.0	200.0	1.0	5.0	200.0
Handling and Equipment/Shields	0.1	1.0	10.0	0.1	0.1	0.1	1.0	2.0	5.0
Propulsion System and Propellants	--	--	--	--	--	--	25.0	100.0	200.0
Transportation	1.0	1.0	10.0	1.5	1.5	1.5	0.5	1.0	1.0
Totals (d)	39.1	74.0	332.0	104.6	213.6	613.6	83.5	318.0	786.0

(a)Radiation shield, a payload requirement, also serves as the primary structure. The cost is displayed for completeness, but not included in total for NWD.

(b)For tungsten nose cones. If steel can be used, the nose cone cost estimates would be $6.0, $10.0, $15.0; $25.0, $65.0, $140.0; $10.0, $15.0, $20.0.

(c)This instrument package would be part of the astrionics and is costed as part of the propulsion system.

(d)For tungsten nose cones. If steel can be used, the totals would be $15.1, $34.0, $267.0; $89.6, $198.6, $593.6; $53.5, $148.0, $556.0.

TABLE 6-12. PERSONNEL AND SUPPLY COST ESTIMATES FOR ESRL OPERATIONS

WBS Category	Number of People Required			Estimated Costs ($, M, 1981)		
	Low	Expected	High	Low	Expected	High
4.1 Management and Support						
4.1.1 Management	20	30	40			
4.1.2 Engineering	50	100	200			
4.1.3 Facility Support	50	75	100			
4.2 Power Plant Operations						
4.2.1 Crew	100	125	150			
4.2.2 Supplies ($, M)				10	10	10
4.3 Technical Personnel and Supplies						
4.3.1 Control Center Crew	20	40	60			
4.3.2 Launcher Equipment Crew	50	75	100			
4.3.3 Equipment Refurbishment Crew	50	75	100			
4.3.4 Power Conversion Facility Crew	10	15	20			
4.3.5 LN_2 Plant/Evacuation System Crew	10	15	20			
4.3.6 Projectile/Payload Operations Crew						
- Nuclear Waste Mission	30	40	50			
- Planetary Probe Missions	5	10	15			
- Earth Orbital Missions	30	40	50			
4.3.7 Facility Utilities Crew	50	100	200			
4.3.8 Accident Recovery Team	20	30	40			
4.3.9 Technical Supplies ($, M)				10	10	15
Personnel Total	495	770	1145			
Annual Personnel Cost at $50 K per Year ($, M)				25	38	57
Total Annual Cost				45	58	82

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6.2.1.1.2 Radiation Shield and Structure (Primary)

The radiation shield and primary container for the waste form is described in Section 4.0. The shield would be welded shut after the waste form and container is inserted. While high standards of quality control would be needed, the shield should be relatively inexpensive to manufacture. A reusable auxiliary shield would provide both cooling and radiation protection during transportation (see Section 4.0). Based on a production rate of 400-to-1000 per year, it is expected that the radiation shield, if made from steel, would cost on the order of $10,000 per unit, including all services. Exotic material use could run the cost up by an order of magnitude or more to $100,000 per unit. Thus, the low and expected cost estimates for the radiation shield are $10,000 and the high estimate is $100,000. Because the radiation shield is also required for other space disposal methods where it is considered part of the payload, this cost is not added to the projectile cost total. This is done to preserve compability for later comparisons of transport costs among space disposal options.

6.2.1.1.3 Nose Cone

The nose cone envisioned is primarily ablative and is needed for ascent and possible reentry. Much technical work is needed on concepts involving metal ablators. If nose cone concepts are reasonable, their fabrication should be relatively inexpensive. Because of the use of exotic materials (tungsten) and the relatively low production rates, the thermal protection system is expected to cost about $100,000 per unit, unless production rates can greatly exceed 1000 per year. If demand for similar nose cone systems permits total production rates of over a thousand per year, or more, it may be possible to achieve costs in the range of $25,000 to $50,000 per unit. The materials cost for 440 kg of tungsten (at $33/kg--AMM, 1982) is $14,520. Labor factors of 2 to 5 are also reasonable for production rates in the hundreds and thousands per year. The low estimate is set at $30,000 with a labor factor of two times materials; the expected estimate is $50,000 (3.5 x materials); and the high estimate is $80,000 (5 x materials).

6.2.1.1.4 Thermal Protection System (Sidebody)

The sidebody thermal protection system (TPS) consists of 25 kg of carbon-carbon composite. At the period when these projectiles could be flying, it is reasonable to expect that these composites, or their equivalents, can be available for about the same materials price per kilogram as tungsten ($33/kg). This would result in a materials cost of $825 for the 25 kg of composite. The major uncertainty, however, is the appropriate labor factor for production in the quantities indicated for this program (thousands per year) and the rate of progress in learning how to manufacture articles using them. Conceivably, the total cost for the TPS for the Mission A projectiles could cost less than $1,000, but this represents more progress in materials cost and labor reduction than is considered likely. Accordingly, to reflect uncertainties, labor factors of about 5, 10, and 20 are used to produce cost estimates of $5, $10, and $20 thousand per unit.

6.2.1.1.5 Instrument Package

The instrument package for the nuclear waste disposal mission is a radio beacon to provide cooperative tracking. If the projectile reenters in the ocean, a salt-water activated pinger would be activated to provide underwater location detection. The instrument package could also activate drag devices to slow down the projectile in the event of a misfire.

High-acceleration survivable electronics and electromechanical actuators are already available in the 10,000 g range and are expected to be readily adaptable to this application. Based upon multiyear procurements concentrated in a single year, it is expected that the instrument package could be procured for less than $1000 per unit.

6.2.1.1.6 Fins

The use of fins is required for aerodynamic stability of the projectile and they must survive for two or three seconds in a very severe environment. Their costs are estimated to be from $1000 to $10,000 per unit for production of 400 to 1000 units per year. The materials cost of 10 kg of tungsten at $33/kg is $330.

6.2.1.1.7 Sabot

The sabot positions the projectile during launch and aids in distributing the acceleration forces. An unresolved question is whether the sabot can be made of relatively low-strength material which can deform to accommodate local stresses or whether it must be made of high-strength materials. If low-strength (and, therefore, inexpensive) materials can be used, the cost per launch could be low--less than $1,000--if some common plastics are acceptable. It is expected that 140 kg of high-strength materials can be made into a sabot for less than $5000 ($35/kg, including fabrication) and this is selected as the expected case. If ultra high-strength materials are required, their costs, and the cost of testing them, can become a major fraction of the projectile cost. To reflect these uncertainties, a high cost estimate of $200,000 is selected.

6.2.1.1.8 Auxiliary Radiation Shields and Specialized Equipment

For transport and handling, a shipping cask providing shielding and active cooling is needed. This would be reusable. Depending on the design requirements the cask represents an investment of hundreds of thousands to millions of dollars, if a multiple projectile shipping cask is selected. Lacking a detailed design, it is estimated that a specialized equipment cost per launch of $1000 can be achieved, but depending upon the design, this could range from $100 to $10,000 per launch.

6.2.1.1.9 Transportation Costs

Based on shipping multiple casks per trip and cooling by ducted or compressed air returned to the atmosphere, shipping expenses could be relatively inexpensive. Transportation by ship at $1000 per projectile is believed to be reasonable. This is used as the low and expected estimates. The high estimate is $10,000.

6.2.1.2 Planetary Probe Mission

The planetary probe missions would use the nuclear waste launcher and be carried in a projectile, perhaps using many of the same components as the nuclear waste projectile. It is expected that after the projectile leaves the atmosphere, at least some portion of the exterior structure would be discarded to expose instruments. This would require some mechanism to remove the structure. The planetary probe spacecraft would then operate in a manner similar to current spacecraft with on-board propulsion to provide attitude control and velocity adjustments. The major difference would be the possible retention of some of the heat shield and/or related structure to provide protection during aerobreaking or planetary atmosphere entry maneuvers. Because the structure will not be serving as a nuclear radiation shield, it can be thinner so as to provide a higher volume for instrumentation.

Because costs usually depend upon both the launch rate and the detailed requirements for each mission, the supporting program cost is highly uncertain. A generalized design may work for most missions, providing variations in volume needed by the instrumentation. Any additional volume increase would typically be accomplished by lengthening the projectile.

Thus, if the technology were available from the nuclear waste disposal projectile, it is expected that most costs would be similar, but impacted by much lower production rates. The production rate effect on cost would be most prominent in the structure and thermal protection systems, and due to low amortization of engineering efforts.

6.2.1.2.1 Payload

Not costed; great variation is likely. Most payload checkout expenses are usually charged to the mission and not the launch.

6.2.1.2.2 Structure

The modified nuclear waste structure would have a low production rate, and would be mechanically more complex as the structure must reliably separate from the payload. Radio or timed command of explosive bolts is envisioned. The low, expected, and high cost estimates for the structure are $50,000, $100,000, and $200,000.

6.2.1.2.3 Thermal Protection System Including Nose Cone

The changed structure would probably require modifications to the thermal protection system. Again, these modifications will require engineering effort which will be amortized over relatively few launches. The low, nominal, and high estimates for the TPS are \$50,000, \$100,000, and \$200,000.

6.2.1.2.4 Projectile Instrument Package

Assumed costs identical to nuclear waste disposal mission.

6.2.1.2.5 Fins

Assumed costs identical to nuclear waste disposal mission.

6.2.1.2.6 Sabot(s)

Assumed costs identical to nuclear waste disposal mission.

6.2.1.2.7 Auxiliary Equipment (Handling Equipment)

The handling equipment will be less complex and most of the auxiliary equipment would be reusable. An expected value of \$100 is used for expendables and amortization of light industrial handling equipment. None of these estimates contains an allowance for the scientific payload checkout, as these costs are normally a part of the scientific mission and not part of the launch services. Facility space for the checkout would be required.

6.2.1.2.8 Transportation

These payloads would normally be air transported from the U.S. already installed in the projectile. With transport packaging, the mass could be 1 to 3 MT tons and be flown in as part of normal resupply and personnel transport. A cost of \$1.00 per kilogram for 1.5 MT, or \$1500 is used.

6.2.1.3 Earth Orbital Missions

The Earth orbital projectile will be similar in exterior design to that of the nuclear waste projectile, but would be larger. The gross masses are 6.5 MT for the Earth orbital projectile versus 2 MT for the nuclear waste projectile. The Earth orbital projectile must carry propulsion, guidance, navigation, and attitude control systems to enable it to achieve a stable orbit. A storable propulsion system is currently envisioned. These propellants can also be used to supply attitude control during the orbital burns and after the final orbit has been achieved. Given the current Reference Concept definition, the on-board propellant mass is 1150 kg for a 500 km circular orbit, and less for higher altitudes.

To achieve a reasonably attractive cost for this propulsion system, it is necessary to hypothesize advances in several technologies as well as a high production rate (for example, 2000 or more per year). With computer-assisted manufacturing and the assumption that high-g components, once developed, are not significantly more expensive than normal components, it is believed possible to produce large numbers of propulsion and GN&C units for an average cost on the order of $100,000 per unit. This is about one-tenth the cost of comparably sized low-g units produced today at a production rate of 5 to 10 units per year.

A higher production rate might permit a cost on the order of an automobile ($10,000 per unit). Extrapolation to this low a cost is possible, but unlikely. Therefore, a lower limit of $25,000 per propulsion system is used. An upper limit of $200,000 per unit, however, is believed to be appropriate under the assumption of a moderately low launch rate.

The need for the propulsion system also implies the need for prelaunch servicing and checkout. Based on the previous assumptions, it is believed that this can be accomplished with a few man-days of effort, in contrast to man-months at the present time. This checkout effort applies only to the propulsion system, and not to a complex payload.

It is noted that the propulsion system costs indicated strongly imply that use of the ESRL for delivery of bulk materials will be attractive only if a low cost propulsion system is achievable.

6.2.1.3.1 Payload

Not costed.

6.2.1.3.2 Structure

Assumed costs identical to nuclear waste disposal mission.

6.2.1.3.3 Nose Cone

The Reference Concept uses tungsten which costs $33/kg. Application of the same costs and labor factors to the projectile results in low, expected, and high cost estimates of $75,000, $185,000, and $260,000.

6.2.1.3.4 Thermal Protection System

Assumed costs identical to the nuclear waste disposal mission.

6.2.1.3.5 Fins

Assumed costs identical to the nuclear waste disposal mission.

6.2.1.3.6 Sabot

Assumed costs identical to the nuclear waste disposal mission.

6.2.1.3.7 Auxiliary Handling Equipment

The propulsion system and GN&C system will require checkout, which is expected to require a few man-days. Low, expected, and high cost estimates are $1000, $2000, and $5000 for time and materials.

6.2.1.3.8 Propulsion System Including Instrument Package

See discussion under 6.2.1.3. Estimates are $25,000, $100,000, and $200,000. Reuse is possible, but not assumed here. (Reuse of propulsion systems could reduce the costs of ESRL Earth orbital missions considerably.) These estimates include all instrumentation and propellants. There is considerable uncertainty about the cost of propellants in the quantities assumed for ESRL use. New production facilities will be needed, and the price of energy would be the determining factor. The cost per kg for propellants (ClF_3 and N_2H_4) or their major ingredients (Cl_2 and F_2) ranges from $0.18/kg for chlorine to $30/kg for propellant-grade hydrazines. At prices of $6/kg, the 1150 kg of propellant envisioned for the projectile would cost $6900.

6.2.1.3.9 Transportation

The 6.5 MT projectile can be transported either by ship, very inexpensively (e.g., $500), or by air at a much higher cost (e.g., $7000). If the projectile is used to transport bulk material to a space station, there would be an additional cost to recover the projectile in orbit. If the space station's orbit and phase in that orbit can be matched, the recovery cost could be very small and offset by the opportunity to reuse the propulsion system. If an orbit transfer vehicle must be dispatched to retrieve the projectile, the additional cost could be significant.

6.2.2 Operations

Operating personnel and operations support would be located in the continental U.S., as well as at a remote site, assumed to be within ± 10 degrees of the equator. In addition to technicians for both the launcher and for some payload support which must be done on site, an ongoing engineering effort would be expected to improve the launchers during their lifetime. The level of effort in the engineering cannot be precisely forecast at this time.

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 all others would be located on site. The personnel identified work on the launcher and their ability to do much more than insert the payload in the launcher is limited. It is expected that complex payloads would be built and checked out before being flown 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 495 to 1145, with an expected estimate of about 770 people. Of these, 100 to 150 are needed to operate the power plant. The power plant crew size was estimated from Electrical World (1981). The personnel and cost estimates on a per MWe basis given by this reference were doubled to arrive at the 100 to 150 estimate for the ESRL power plant.

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 of at least two years. If one-shot brushes were used and it took an average of one man-hour per generator to replace them, the launcher equipment support crew would need at least 1250 people to support one launch per day.

The annual purchase of supplies to support the launcher facility is estimated to be equal to that needed to support the nuclear power plant (Electrical World, 1981), or about $10 M for each. There is insufficient information to justify a highly specific level for supplies for the installation; this level was selected as being a reasonable assumption; it is also noted that at this level, the materials cost has approximately the same magnitude as manpower costs.

The accident recovery team, while four percent of the total, is for people who are needed infrequently. While they would be partially occupied with ongoing training, they would also be available for other tasks.

The personnel estimates, as well as supply estimates, are given in Table 6-12. A value of $50,000 per man-year is used in estimating the cost of the staffing. This includes an allowance for remote-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 low, expected, and high estimates of annual expenses of $45, $58, and $82 M.

6.3 Development Test Program

Details of a development test program are difficult to predict 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 rail segments on the mainland to verify the performance of a railgun segment and most other subsystems. The projectile would be accelerated at 10,000 g's to give confidence 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 one launcher would be complete, and more than 50 percent of the investment in the other launcher would also be complete. The development test would concentrate on assuring that the controls operate correctly and that the hypothesized 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 rocket-propelled reentry tests and subscale electromagnetic launchers, there would still be a need to verify fullscale designs. If the fullscale tests are successful, the development tests would be expected to last about one year. If they are unsuccessful, rework of either the launcher or the projectiles would be required and non-productive costs would mount. For this reason, several alternative projectile designs would probably be undertaken, so that the probability of success would be higher than if only a single design were undertaken. The first component of the development test, that of full-sized segment(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 crew operations costs.

These considerations are taken into account in the development test program which includes:

- Testing of an all-up launcher segment on the mainland
- 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. These costs are summarized in Table 6-13.

TABLE 6-13. DEVELOPMENT TEST PROGRAM COST ESTIMATES
($, M, 1981)

	Low	Expected	High
Launcher Segment Tests	75	100	150
Development of Projectiles	40	60	80
Transient Housing	5	5	10
Two Years of Operations	90	116	164
Totals	$210 M	$281 M	$404 M

6.4 ESRL Cost Summary

This section assembles the various cost estimates to provide an overview of the system costs and their implications for costs per launch and cost per unit mass of payload. This section also discusses sensitivities of launch costs to design parameters. The costs given represent costs for transport only; they do not include costs for payloads. Also excluded are the costs of an ongoing technology support program which would provide the designs used in any follow-on development, whether at the same site to upgrade the launchers or to provide the technology or designs for a greater capability at another site.

The costs developed in detail for the two launcher system are summarized in Table 6-14. Research and design efforts are estimated uniformly at 10 percent of the development/investment cost estimates. A reasonable estimate of the range on these costs in from 5 to 25 percent; the actual amount would depend upon the success of basic research, hardware development, and system design efforts. The effective lifetime of the initial investment in the ESRL facilities is estimated at 30 years. From these assumptions and a maximum mission rate of two launches per day from the nuclear waste launcher (Mission A) and eight per day from the Earth orbital launcher (Mission B), programmatic costs are developed. These include annual transport costs, the transport cost per mission and the effects of lower use rates on the Mission B launcher. The 30-year average cost per launch for the nuclear waste disposal missions ranges from \$85 K to \$425 K, while that for the Earth orbital mission ranges from \$130 K to \$879 K. The expected costs are \$139 K and \$383 K. The projectile costs are a dominating factor and less expensive design is clearly desirable. The expected startup cost is \$5.4 B and for 30 years of operation at ten launches per day, the expected total transport cost would be \$36.6 B.

To provide an estimate of the costs for only one of the launchers, it is necessary to estimate the costs for each launcher separately. Low, expected, and high estimates of the facilities and rail launcher systems are presented in Table 6-15. The majority of the facilities are allocated to the Mission A launcher in this table, but are reallocated in Table 6-16, which presents expected costs only, for three different configurations (Mission A launcher only, Mission B only, and both Missions A and B). If only the Mission A launcher were built, the expected unit launch cost would increase from \$139 K to \$237 K. This is due to the fact that this launcher is used only twice a day and the costs are not as well ammortized as they would be at a launch rate of ten per day.

The effect of variation in launch rates on unit launch costs is shown in Table 6-17. The cummulative 30 year total low, expected, and high launch cost estimates and the resultant average unit costs are presented for the two launcher system. Two Mission A launches per day is assumed to be the minimum launch rate. Only for the low estimate is there any significant variation in the unit launch cost with launch rate. This is due to the very high unit costs determined for the projectiles.

TABLE 6-14. ESRL COST ESTIMATE SUMMARY--TWO LAUNCHER CONCEPT ($, M, 1981)

Cost Category	Missions A and B Low	Expected	High
Research and Design (10% of Development/Investment)	320	466	663
Development/Investment	3,204	4,656	6,633
Development Test Program	210	281	404
Total Investment	3,734	5,403	7,700
Annual Operating Expenses	45	58	82
Projectile Unit Costs			
- Nuclear Waste Mission	0.039	0.074	0.332
- Planetary Probe Mission	0.105	0.214	0.614
- Earth Orbital-With Propulsion	0.083	0.318	0.786
Annualized Costs--10 Launches Per Day			
- 30 Year Amortization of Total Investment	124.5	180.1	256.7
- Annual Operating Expenses	45.0	58.0	82.0
- 730 Nuclear Waste Projectiles	28.5	54.0	242.4
- 2920 Earth Orbital Projectiles	242.4	928.6	2,295.1
Annual Totals	440.4	1,220.7	2,876.2
Annualized Unit Launch Cost -10 Launches Per Day			
(Projectile plus Share of Annualized Costs - 20/80)			
- Nuclear Waste Mission	0.085	0.139	0.425
- Earth Orbital Mission	0.130	0.383	0.879
Annualized Cost per kg of Payload	-----------	Dollars per Kilogram	----------
- Nuclear Waste Mission (250 kg)	340	556	1,700
- Earth Orbital Mission (650 kg)	200	589	1,352

TABLE 6-15. DEVELOPMENT/INVESTMENT COST-DIVISION BETWEEN LAUNCHERS ($, M, 1981)

	Mission A			Mission B		
	Low	Expected	High	Low	Expected	High
Facilities and Supporting System						
Land	9.6	12.0	16.0	--	--	--
Power Plant	108.0	120.0	130.0	108.0	120.0	130.0
Personnel Support Facilities	33.0	52.3	75.0	17.0	24.7	40.0
Shipping, Docks, Storage, Transp.	20.0	50.0	100.0	--	--	--
Airfield and Hangar	26.3	56.0	100.0	--	--	--
Industrial Area	40.0	60.0	80.0	--	--	--
Administration/Engineering Building	5.0	5.0	5.0	--	--	--
Subtotal	(241.9)	(355.3)	(506.0)	(125.0)	(144.7)	(170.0)
Rail Launcher Systems						
Tunnels/Shafts	206.0	309.0	540.0	206.0	309.0	540.0
Tubes - Copper	21.6	27.0	48.7	43.4	54.0	97.3
- Spacers	7.7	15.8	26.3	16.0	31.6	52.7
- Kevlar	42.4	86.5	129.7	84.8	172/9	259.3
- Container	2.0	4.2	9.4	4.0	8.4	18.8
Homopolar Generators	588.0	749.5	942.5	588.0	749.5	942.5
- Support Structure	168.5	244.0	280.5	168.5	244.0	280.5
Inductors and Switches	210.0	268.0	318.5	210.0	268.0	318.5
Gas Injection Systems						
- Preboost System	90.0	100.0	120.0	10.0	25.0	30.0
- Deceleration System	0.5	1.0	1.5	0.5	1.0	1.5
Power Conversion Plant	12.5	24.0	35.0	12.5	24.0	35.0
Water Distillation Plant	5.0	5.0	5.0	--	--	--
Gas Handling Facilities						
- LN_2 Plant and Storage	17.0	21.4	40.0	17.0	21.4	40.0
- Vacuum System for Tube	0.5	0.7	0.7	0.5	0.8	0.
- H_2O Electrolysis Plant	0.2	0.3	0.4	--	--	--

TABLE 6-15. (Continued)

	Mission A			Mission B		
	Low	Expected	High	Low	Expected	High
Elevator Systems, Etc.	20.0	70.0	164.0	20.0	70.0	164.0
Control Center, Controls	11.0	50.0	112.0	9.0	50.0	124.0
Tracking Systems	10.0	100.0	200.0	--	--	--
Accident Recovery System						
- Ship	30.0	40.0	70.0	--	--	--
- Submersible	5.0	10.0	20.0	--	--	--
Subtotal	(1447.9)	(2126.4)	(3065.2)	(1515.2)	(2029.6)	(2904.9)
Total	1689.3	2481.7	3571.2	1640.2	2174.3	3074.9

TABLE 6-16. EXPECTED COST ESTIMATE SUMMARY--ONE AND TWO LAUNCHER CONCEPTS ($, M, 1981)

	Mission A Only	Mission B Only	Missions A and B
Research and Design	450	450	466
Development and Investment	2480	2635	4656
Development Test Program	225	235	281
Total Investment	3157	3320	5403
Annual Operating Expenses	40	45	58
Projectile Unit Costs			
- Nuclear Waste Mission	0.039	--	0.039
- Earth Orbital, With Propulsion	--	0.318	0.318
Annualized Costs	2 Launches	8 Launches	10 Launches
- 30 Year Amortization of Investment	105.2	110.7	180.1
- Annual Operating Expense	40.0	45.0	58.0
- 730 Nuclear Waste Projectiles	54.0	—	54.0
- 2920 Earth Orbital Projectiles	--	928.6	928.6
	199.2	1084.3	1220.7
Annualized Unit Launch Cost			
- Nuclear Waste Mission (2)	0.273	--	0.139
- Earth Orbital Mission (8)	--	0.371	0.383
Annualized Cost per kg of Payload	------------Dollars per Kilogram---------		
- Nuclear Waste Mission (250 kg)	1092	--	556
- Earth Orbital Mission (650 kg)	--	571	589

TABLE 6-17. EFFECTS OF LAUNCH RATE ON UNIT LAUNCH COST--2 LAUNCHER CONCEPT
($, M, 1981)

	30 Years Cumulative and Unit Costs					
	Low		Expected		High	
	Total	Unit	Total	Unit	Total	Unit
Research and Design	320	--	466	--	663	--
Development-Investment	3,204	--	4,656	--	6,633	--
Development Test Program	210	--	281	--	404	--
Total Startup Costs	3,734	--	5,403	--	7,700	--
30 Years-Annual Costs	1,350	--	1,740	--	2,460	--
	5,084		7,143		10,160	
2 x Mission A per Day	5,983	0.271	8,764	0.400	17,430	0.796
1 x Mission B	6,847	0.208	12,245	0.372	26,037	0.793
2 x Mission B	7,756	0.177	15,727	0.359	34,644	0.791
3 x Mission B	8,664	0.158	19,210	0.350	43,251	0.789
4 x Mission B	9,573	0.145	22,692	0.345	51,857	0.789
5 x Mission B	10,482	0.136	26,147	0.341	60,464	0.788
6 x Mission B	11,391	0.130	29,656	0.339	69,070	0.788
7 x Mission B	12,300	0.125	33,138	0.336	77,677	0.788
8 x Mission B	13,209	0.120	36,620	0.334	86,284	0.787
Mission B Projectile Unit Cost		0.083		0.318		0.786

The projectile unit cost is very sensitive to the technology assumptions. The nose cone is considered to be made of tungsten, which is expensive and difficult to fabricate. Tungsten was selected here because of the extreme thermal environment and lack of knowledge of the response of materials to this environment. Since that time, preliminary calculations indicate that steel might work. It is hoped some less expensive material can be shown to be adequate. This would result in significant savings in total costs as the tungsten nose cone represents 67 percent of the expected cost for the Mission A projectile and 58 percent of the Mission B projectile. If the nose cone cost is lowered, the next most significant cost is that of the propulsion system for the Mission B projectile. Here manufacturing technology is important, as rocket propulsion systems are now made at rates of tens per year and would be required in the thousands per year. The uncertainty in their ultimate cost is reflected in the order of magnitude range in the estimated costs.

For the launchers, the major cost sensitivity uncovered is that of the launch tube diameter. The outside diameter of the tube should be kept as small as possible for two reasons: the material needed to confine the tube itself (Kevlar) is expensive, and a small tube would also permit a smaller shaft. Since the cost of the shaft is proportional to the volume of rock removed, the diameter needs to be as narrow as possible consistant with other design requirements. The costs of excavation are second only to those for the homopolar generators, and clearly give impetus to design efforts to keep the HPG's as small as possible, or to consider other technologies which may be more volumetrically efficient.

7.0 TECHNOLOGY ASSESSMENT

The objective of this task was to assess the status of current railgun technology as it applies to an Earth-to-Space Rail Launcher concept, 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. Basic railgun technology input was provided by the three railgun consultants, namely: Dr. Richard Marshall, University of Texas at Austin; Dr. John Barber, IAP Research, Inc., Dayton, Ohio; and Mr. Ron Hawke, Lawrence Livermore National Laboratory, California. 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 work. The second subsection provides a suggested plan for NASA to follow, if it desires to develop railgun technology for propulsion applications in the space program.

7.1 Technology Evaluation

The technology evaluation activity centered around the review of many railgun documents that are available in the literature (see references in Appendix A). Also, based upon the numerous discussions with railgun experts throughout the country, and based upon the current Reference Concept Definition, as given in Section 4.0 of this report, and information contained in Section 3.0 of this report, some basic technological areas have been identified for supporting research and technology (SR&T) for Earth-to-Space Rail Launcher systems. It should be noted that on-orbit launcher propulsion also matches many of these identified technology areas.

The primary areas which require technology development (as identified in the study) are given as follows:

- The experimental demonstration of the distributed energy store concept with switching and control
- Development of a low drag, survivable, and flight-stable projectile concept, including technology work on materials
- Testing and development of sabot concepts and survivable materials
- Experimental work related to the early and rapid separation of the sabot in the atmosphere
- The study of friction problems in high-speed railguns and the determination of bore tolerances
- Experimental work related to the possible use of a solid/plasma armature
- Technology work related to the use of round bore railguns

- Technology work related to the improvement of high-speed HPG concepts with improved brush materials

- Propulsion systems and instruments that are designed to survive 2500 to 10,000 g's

- Demonstrations of preboost and deceleration system concepts

- Design work related to the payload support structure and the hardware interface with the rail launcher tube itself.

The following paragraphs further discuss each of the above named technological need areas that were identified in this study.

7.1.1 Energy Distribution, Switching, and Control

From a performance standpoint, distributed energy store rail launch systems are the most desirable for launching material into space. Because of their desirability, distributed energy stores have been selected for the Reference Concept in this study. However, considerable technology work is required to verify the concept of distributed energy and the associated switching. At the University of Texas, Center for Electromechanics, research work is currently under way to verify the concept of a simple distributed energy store system for a small railgun. Work of this type is considered essential for the development of the technology. Technology related to segmented railguns does not exist. Basically, this concept involves the end to end placement of multiple railgun systems. The switching for this concept is also complex. Switching and control of current into the rail is very critical to the feasibility of these concepts. To our knowledge, little work has been done in this area, and to justify continued work on distributed energy store railgun systems, the switching area should take priority.

7.1.2 Earth-to-Space Rail Launched Projectiles

The ultimate design of a ESRL projectile depends on aerodynamic drag, aerodynamic stability, aerodynamic heating of surfaces, and the launch and flight stress on the body. Work done during this study indicates that all four of the above are critical to the concept of launching material into space.

Aspects of drag 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 traverses the atmosphere. Preliminary assessment conducted during this study indicates that the drag coefficient will be dominated by Newtonian drag. Other contributions to the drag, such as skin friction may prove to be important. The skin friction drag component would likely be dominated by the type of surface material used on the nose cone and on the sidebody of the projectile. In the current

Reference Concept developed in this study, it was assumed that a drag coefficient of 0.1 was achievable with slender, spherically blunted cones. Experimental verification of low drag bodies under these flight conditions is most desirable to verify the ESRL concept. Based on some experimental work by Daniel and Milton (1980) it appears that low drag bodies are possible, but extensive experimental work is necessary.

The aerodynamic features of an ESRL projectile relate directly to the aerodynamic stability as the projectile leaves the muzzle of the launcher tube. 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 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. One possible solution to the stability problem could be that one uses a round bore railgun with a round projectile and sabot. In this configuration the projectile could be spun up at the breech prior to launch. This would very much enhance aerodynamic stability during the launch phase and could eliminate the need for 3-axis control on-orbit. A significant amount of theoretical and experimental work is required in this area before a definitive assessment can be given on the issue of projectile stability.

Aerodynamic heating is also a very critical aspect of the ESRL concept. Initial assessment indicates that because the projectile flies so rapidly through the atmosphere, 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. The current Reference Concept assumes that tungsten is used for the nose cone. If, after additional study, it is found that the mass loss on the tungsten nose tip is not significant and steel performs well, then it is recommended that high-strength steel materials be used for the nose tip. It is very important that the drag characteristics for the projectile not be modified to any significant degree during the early portion of the flight by a change in shape of the nose tip. It has been assumed that the heating on the side of the projectile could be accommodated by a carbon/carbon material. Small scale testing in a experimental railgun would significantly aid the development of projectile concepts by being able to actually test the conditions that are of concern.

The fourth major area of concern in design of the projectile relates to the ability of the projectile to withstand the stresses, both aerodynamic and launch, that will be experienced by the various components in the projectile. A finite element structural analysis is required to establish confidence in conceptual designs for the projectiles. For the nuclear waste disposal in space mission, the survivability of the radiation shield section, which comprises the bulk of the mass in the projectile, is critical to the performance of the mission. For the planetary probe mission, it is important that the instrumented payload be able to survive the high acceleration levels. Detailed response analysis for possible candidate instrument packages is required to verify that this mission could be accomplished. For the Earth orbital missions, under lower acceleration levels (1100 g's), structural analysis for the projectile, including the propulsion system and various candidate payloads is also needed. Existing expertise in the area of high acceleration gun-launched projectiles with smart warheads would perhaps be able to contribute significantly to this evaluation.

7.1.3 Sabots

The sabot used to allow the projectile to be accelerated in the ESRL system is critical to the feasibility of the concept. Technology work is needed to establish a data base on both square and round shaped sabots. Sabots for use in the Earth-to-Space Rail Launcher system would undoubtedly have to exhibit very high-strength and non-conducting characteristics. Additional research and testing is necessary to establish the survivability of the sabot in the acceleration phase and the ability of the sabot to be jettisoned quickly in the atmosphere without imparting significant pitching moments to the projectile after breach. Various analytical and experimental tests are believed necessary to evaluate designs and materials. Various experts in sabot technology should be tasked to aid in the development of these concepts.

7.1.4 Friction and Bore/Sabot Tolerances

Another critical technology area that needs to be investigated prior to development of an Earth-to-Space Rail Launcher system, is the evaluation of: (1) sabot/projectile friction during the launch phase, and (2) the tolerances that are required to avoid sabot 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 sabot at the time of launch. Experiments could be conducted in existing railgun facilities. The problem of rail movement as a result of continued firings of the launcher is an important aspect related to the bore tolerances. This aspect determines the reusability of the rails after numerous firings.

7.1.5 Solid/Plasma Armature

It may be desirable to use a solid/plasma armature to reduce the voltage loss in the armature during the firing, as suggested by Marshall in Section 3.2. Therefore, experimental work related to the possible use of a solid/plasma armature is recommended to be undertaken.

7.1.6 Launcher Tube Bore

The design of a bore and its supporting structure is a key element in determining the reusability of the rail launcher tube. The selection of materials and support structures is critical to maintaining rail position and bore tolerances. Experimental work is required to establish the technical feasibility of the round bore railgun concept. This would allow the capability for spin stabilizing the projectile for the flight. The round bore concept also allows the possible re-machining of the bore and use of a larger sabot size as the gun is utilized over time. Lawrence Livermore National Laboratory has conducted some preliminary experimental work on round bore railguns. Work to date indicates that the round bore concept is feasible and looks attractive.

7.1.7 Technology Improvements on Brush Materials for HPGs

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. In discussions with Bill Weldon at the University of Texas, Center for Electromechanics, 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. Currently, 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 perhaps 300 m/s using advanced materials, but still experience a great deal of erosion. A big issue in the Earth-to-Space Rail Launcher system would be the replacement rate of brush material. It is desirable to operate the HPGs at high speeds and have brushes which will allow minimal maintenance over long periods of time. This is 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.

7.1.8 Propulsion Systems and Instruments

The survivability of propulsion systems and instruments on board projectiles is obviously critical to the feasibility of the ESRL Earth orbital missions. Additional studies are required to optimize the propulsion system for an Earth-to-space rail launched projectiles. Both solid and liquid propulsion systems should be considered in the analysis, and the choice of the propellants should be optimized. Detailed design analysis for a propulsion system's structural integrity under high accelerations (in the range of 1000 to 2500 g's) needs to be accomplished. The propulsion design concepts need to be coordinated carefully with the projectile overall design. Also, work is needed to evaluate the survivability of computers and sensors and other instruments that are required to carry out the mission. It is recommended that the first activity be a detailed design study of the propulsion options available for the Earth orbital mission application.

7.1.9 Preboost and Deceleration Systems

Experimental demonstration of preboost and deceleration system concepts can probably be done at a minimal cost using existing railgun systems. This area of work is not considered critical to the concept in that a high degree of confidence exists that a preboost system can be built that would work. The basic technology is available from the light gas gun work that has been accomplished over the years. Deceleration systems are not necessarily critical to the overall feasibility of the Earth-to-Space Rail Launcher system. It is felt that these systems could be designed and be made to operate without significant degree of effort.

7.1.10 Rail Launcher Structural Support and Hardware Interface

Preliminary design work related to the structural support and hardware interface of a rail launcher tube is required. Aspects of the launcher tube recoil and alignment need emphasis. A preliminary system study on hardware interfaces would help in improving cost estimates for the system.

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. Figure 7-1 provides an overall implementation schedule for our recommendation. Table 7-1 provides our estimates of funding requirements for this recommendation in 1982 dollars.

Three major areas of activity have been categorized; (1) ESRL experimental research, (2) ESRL system studies, and (3) special studies. The philosophy in developing the schedule for ESRL SR&T was based upon first developing, on an experimental basis, a sound data base upon which to project growth in railgun systems. It is also important to evaluate other possible concepts for Earth-to-space accelerator launch systems; this would be conducted early on in the activity to provide a change of direction, if necessary, in the program. Detailed system design studies would occur in the third year of activity and would peak in the fourth year. The fifth year would be used to integrate the results of the entire effort to provide a preliminary environmental impact assessment and allow for a recommendation for termination or a continuance of the program.

It is anticipated that further detailed analysis and design efforts would identify additional areas of needed technology. This preliminary feasibility assessment has indicated major technological needs, but additional efforts should identify new needs. It is anticipated that basic research on railguns by various other government agencies and industrial organizations will continue to contribute to the advancement of the ESRL concept. It is also possible that other agencies in the federal government might join with NASA and contribute to the overall funding requirements for an ESRL launcher system. Such agencies include: the Department of Defense, in particular, the United States Air Force Space Division, and Rocket Propulsion Laboratory, and the U.S. Army; and the Department of Energy (in support of its fusion program).

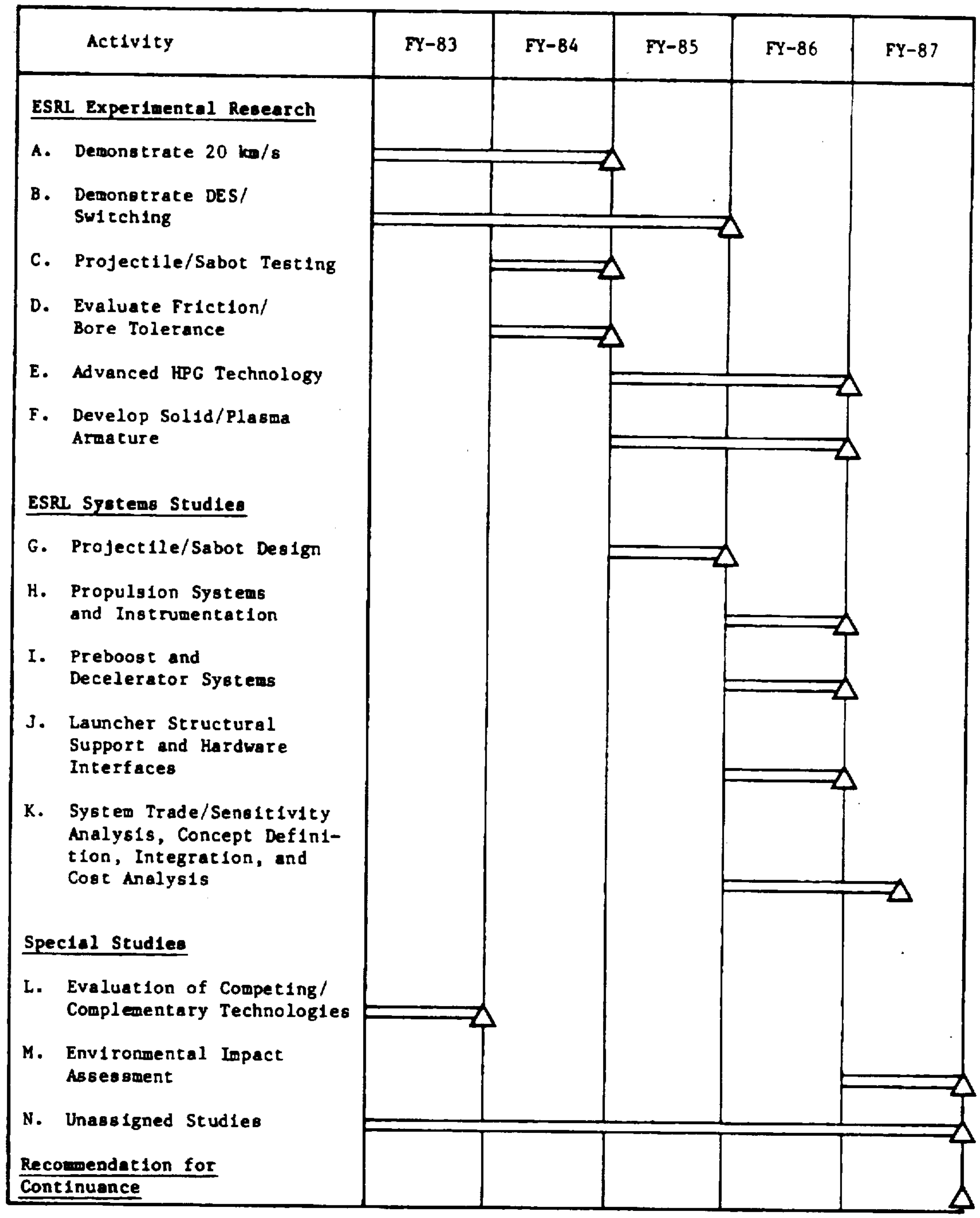

FIGURE 7-1. RECOMMENDED SCHEDULE FOR ESRL SUPPORTING RESEARCH AND TECHNOLOGY ACTIVITIES

TABLE 7-1. ESTIMATED FUNDING REQUIREMENTS (1982, K$) FOR RECOMMENDED ESRL SUPPORTING RESEARCH AND TECHNOLOGY ACTIVITIES

Activity	FY-83	FY-84	FY-85	FY-86	FY-87	Total
ESRL Experimental Research						
A. Demonstrate 20 km/s	200	200	--	--	--	400
B. Demonstrate DES/ Switching	250	250	150	--	--	650
C. Projectile/Sabot Testing	--	50	--	--	--	50
D. Evaluate Friction/ Bore Tolerance	--	50	--	--	--	50
E. Advanced HPG Technology	--	--	--	100	100	200
F. Develop Solid/Plasma Armature	--	--	--	50	25	75
Subtotal	(450)	(550)	(150)	(150)	(125)	(1425)
ESRL Systems Studies						
G. Projectile/Sabot Design	--	--	200	--	--	200
H. Propulsion Systems and Instrumentation	--	--	--	150	--	150
I. Preboost and Decelerator Systems	--	--	--	50	--	50
J. Launcher Structural Support and Hardware Interfaces	--	--	--	250	--	250
K. System Trade/Sensitivity Analysis, Concept Definition, Integration, and Cost Analysis	--	--	--	200	100	300
Subtotal	(--)	(--)	(200)	(650)	(100)	(950)
Special Studies						
L. Evaluation of Competing/ Complementary Technologies	125	--	--	--	--	125
M. Environmental Impact Assessment	--	--	--	--	100	100
N. Unassigned Studies	50	50	50	50	50	250
Subtotal	(175)	(50)	(50)	(50)	(150)	(475)
Total SR&T	625	600	400	850	375	2850

8.0 APPLICATIONS AND BENEFITS ASSESSMENT

The objective of the applications and benefit assessment task was to identify possible significant benefits of an Earth-to-Space Rail Launcher (ESRL) system and to provide a preliminary economic analysis. An additional objective was to identify and assess possible applications of electromagnetic rail launchers with respect to the payload and launch requirements, safety, environmental impact, and economic analysis of the applications.

This section is composed of three subsections. Section 8.1 presents the candidate ESRL applications that were identified. Section 8.2 discusses the applications individually in terms of the justification for including the application, the requirements of the application, and its assessment. Section 8.3 describes the economic assessment conducted for the Reference ESRL components.

8.1 Identification of Possible Applications

A list of peaceful candidate applications of an Earth-to-Space Rail Launcher (ESRL) system was identified in support of the preliminary requirements. The eight identified peaceful applications and brief discussions of each are listed below.

Nuclear waste disposal in space (NWDS)--This is the Reference Concept for the study. High-level nuclear waste would be placed in a solar system escape trajectory for permanent disposal. A minimum velocity of 16.7 km/sec (without atmospheric losses) is necessary for this mission. The simple design and economics of a reusable launcher system make this an attractive nuclear waste disposal option.

Earth-orbital applications--The rail launcher system could be used to deliver supplies to space stations, to launch materials for use in space, and to launch satellites. This use would require additional propulsion to reach orbit. The system could be used to resupply space stations with items such as propellants, food, and spare parts. It is possible that an orbit transfer system might be required to move the payload to the space station. Another use would be to launch materials for fabrication or materials processing in space. The ESRL system could also be an economical alternative to the Space Shuttle launch of small satellites.

Atmospheric research--The rail launcher system could be used as an atmospheric research tool. Possible uses are chemical release experiments in the upper atmosphere, general sounding rocket type applications, and reentry studies.

Interstellar/planetary probes--Scientific payloads could be launched for interstellar and/or planetary exploration purposes. Either direct or indirect (gravity-assisted) trajectories are possible. Indirect trajectories could involve the use of midcourse-correction propulsion systems.

Chemical rocket boost system--A rail launcher could be used to give a "low acceleration" initial Δv to a chemical system. This could assist chemical systems attain higher performance. The rail launcher could also be used as a boost system for an advanced horizontal space vehicle.

Toxic chemical disposal in space--Disposal of toxic chemicals by means of a rail launcher is a possible beneficial application. The mission envisioned is similar to that of nuclear waste disposal.

Hybrid rail launcher and laser propulsion--An ESRL projectile containing a laser propulsion system would be launched. An on-orbit laser beam could be focused onto a collection window to heat hydrogen for a propulsion maneuver needed to prevent reentry.

Lunar-gravity-assisted launch of payloads--A moon-swingby trajectory after a rail launch could place the projectile into a stable orbit without a chemical stage. This could allow delivery of material to orbit without the use of an additional propulsion system.

8.2 Applications Assessment

After the eight candidate ESRL applications were identified in Section 8.1, they were evaluated. This section presents the results of the preliminary evaluation. These evaluations were based upon several issues including:

- Description of application
- Payload
- Launch requirements
- Additional propulsion requirements
- Safety
- Environmental impact
- Economics.

A preliminary study described mission scenarios for each application. Alternatives (both conventional and prospective) were identified and compared. A first consideration in the assessment was the payload to be launched. Payload requirements include volume, mass, additional propulsion systems needs, and any other mission-specific requirements. Payload characteristics were estimated, and these were used to produce a traffic model for the Reference ESRL Concept. This traffic model was then used to assess the possible demand for an ESRL system. Input to the traffic model included past and current space station studies, nuclear waste disposal studies, and NASA launch vehicle traffic models. Payload scenarios were developed for a thirty-year period (2020-2050) for Mission A (primarily nuclear waste disposal) and for Mission B (Earth orbital applications). Two versions of the traffic models are shown for each case: the number of launches required each year by payload type, and the mass of payload to be launched per year. For several of the payloads, different levels of activity are given which indicate estimates for maximum and minimum launch activity.

The preliminary traffic model for Mission A is shown in Tables 8-1 and 8-2. The nuclear waste available in both cermet and PbI_2 forms has been determined in previous studies (Rice et al, 1982). The mass of waste form in each ESRL projectile is 250 kg for the cermet and 600 kg for the iodine. The difference in mass is due to the radiation shielding requirements for cermet. Two launches per day with various levels of activity are assumed. High, medium, and low launch activity correspond to seven, six, and five work days per week, respectively. Secondary missions for this launcher system would be planetary probes. The demand for this type of mission was estimated from past traffic models and future plans, given the availability of an ESRL system. Ranges of space probe activity were estimated at one to four launches per year, with a 600 kg payload.

Some Mission B applications depend upon a space station community. The U.S. orbital presence for the years 2020-2050 was based upon past and present space station studies which were extrapolated for the later years. The number of space station personnel was divided into civilian (LEO), military (LEO), and geosynchronous orbit (GEO) personnel, and is tabulated in Table 8-3.

Mission B payloads which appear in the traffic model include space station resupply articles, LEO satellites, propellants, and materials for microgravity processing. Space station resupply items are food, propellants for LEO to GEO crew rotation, and spare parts. The Mission B traffic model is summarized in Tables 8-4 and 8-5. The payload numbers are taken from the discussion of the corresponding application in this section, while the number of launches per year is derived from the total mass available per year and an estimated payload density (which corresponds to a maximum payload mass per projectile from Table 3-2).

Table 8-6 summarizes the expected daily launch rates as a function of year. This table provides a preliminary estimate for the number of launches per day that one might expect from an all-up ESRL facility providing support for nuclear waste disposal, planetary probes, and Earth orbital missions.

The eight applications were also evaluated regarding launch requirements. These requirements include projectile trajectory, launch velocities and angles, and acceleration limits. The application determines the projectile trajectory, which in turn, determines the launch velocity. Descriptions of the launch velocity requirements for LEO and solar system escape missions are contained in Section 3.1.3. Acceleration limits were estimated depending upon the sensitivity of the payload and any on-board equipment.

TABLE 8-1. NUMBER OF ESRL MISSION A LAUNCHES PER YEAR

Year	Iodine for Disposal	High-Level Nuclear Waste for Disposal			Planetary Probes	
		High	Medium	Low	High	Low
1	3	180[a]	153[a]	128[a]	4	1
2	2	363[a]	311[a]	259[a]	4	1
3	2	546[a]	467[a]	390[a]	4	1
4	2	728	624	520	4	1
5	3	727	623	519	4	1
6	2	728	624	520	4	1
7	2	728	624	520	4	1
8	3	727	623	519	4	1
9	3	727	623	519	4	1
10	3	727	623	519	4	1
11	3	727	623	519	4	1
12	4	726	622	518	4	1
13	3	727	623	519	4	1
14	3	727	623	519	4	1
15	4	726	622	518	4	1
16	4	726	622	518	4	1
17	3	727	623	519	4	1
18	3	727	623	519	4	1
19	4	726	622	518	4	1
20	3	727	623	519	4	1
21	4	726	622	518	4	1
22	3	727	623	519	4	1
23	4	726	622	518	4	1
24	3	727	623	519	4	1
25	4	726	622	518	4	1
26	3	727	623	519	4	1
27	4	726	622	518	4	1
28	4	726	622	518	4	1
29	3	727	623	519	4	1
30	4	726	622	518	4	1
Totals	95	20,711	17,745	14,783	120	30

[a] ESRL "phase-in" period.

TABLE 8-2. MASS (MT) OF MISSION A PAYLOADS LAUNCHED PER YEAR

Year	Iodine for Disposal	High-Level Nuclear Waste for Disposal			Planetary Probes	
		High	Medium	Low	High	Low
1	1.8	45.00[a]	38.25[a]	32.00[a]	2.4	0.6
2	1.2	90.75[a]	77.75[a]	64.75[a]	2.4	0.6
3	1.2	136.50[a]	116.75[a]	97.50[a]	2.4	0.6
4	1.2	182.00	156.00	130.00	2.4	0.6
5	1.8	181.75	155.75	129.75	2.4	0.6
6	1.2	182.00	156.00	130.00	2.4	0.6
7	1.2	182.00	156.00	130.00	2.4	0.6
8	1.8	181.75	155.75	129.75	2.4	0.6
9	1.8	181.75	155.75	129.75	2.4	0.6
10	1.8	181.75	155.75	129.75	2.4	0.6
11	1.8	181.75	155.75	129.75	2.4	0.6
12	2.4	181.50	155.50	129.50	2.4	0.6
13	1.8	181.75	155.75	129.75	2.4	0.6
14	1.8	181.75	155.75	129.75	2.4	0.6
15	2.4	181.50	155.50	129.50	2.4	0.6
16	2.4	181.50	155.50	129.50	2.4	0.6
17	1.8	181.75	155.75	129.75	2.4	0.6
18	1.8	181.75	155.75	129.75	2.4	0.6
19	2.4	181.50	155.50	129.50	2.4	0.6
20	1.8	181.75	155.75	129.75	2.4	0.6
21	2.4	181.50	155.50	129.50	2.4	0.6
22	1.8	181.75	155.75	129.75	2.4	0.6
23	2.4	181.50	155.50	129.50	2.4	0.6
24	1.8	181.75	155.75	129.75	2.4	0.6
25	2.4	181.50	155.50	129.50	2.4	0.6
26	1.8	181.75	155.75	129.75	2.4	0.6
27	2.4	181.50	155.50	129.50	2.4	0.6
28	2.4	181.50	155.50	129.50	2.4	0.6
29	1.8	181.75	155.75	129.75	2.4	0.6
30	2.4	181.50	155.50	129.50	2.4	0.6
Totals	57.0	5177.75	4436.25	3695.75	72.0	18.0

[a] ESRL "phase-in" period.

TABLE 8-3. PROJECTED SPACE STATION PERSONNEL (Year 1=2020)

Year	Civilian LEO	Military LEO	GEO	Total
1990	12	--	--	12
1995	12	12	--	24
2000	25	25	5	55
2010	50	50	10	110
2020	100	100	20	220
1	100	100	20	220
2	100	100	20	220
3	100	100	20	220
4	150	110	20	280
5	150	110	20	280
6	150	110	20	280
7	200	110	20	330
8	200	120	20	340
9	200	120	20	340
10	250	120	20	390
11	250	120	20	390
12	250	120	20	390
13	300	130	20	450
14	300	130	20	450
15	300	130	20	450
16	300	130	20	450
17	350	130	20	500
18	350	130	20	500
19	350	140	20	510
20	350	140	20	510
21	400	140	20	560
22	400	140	20	560
23	400	140	20	560
24	400	140	20	560
25	450	140	20	610
26	450	150	20	620
27	450	150	20	620
28	450	150	20	620
29	450	150	20	620
30	500	150	20	670

TABLE 8-4. NUMBER OF ESRL MISSION B LAUNCHES PER YEAR

Year	Materials for Space Processing (ρ = 2.7 g/cc)	Food for Space Station (ρ = 1 g/cc)	Spares for Space Station (ρ = 1.5 g/cc)	Propellants for LEO → GEO Crew Rotation (ρ = 1g/cc)	LEO Satellites	Propellants for transport of LSS (ρ = 1 g/cc)
1	365	336	50	830	3	516
2	365	336	50	830	3	516
3	365	336	50	830	3	516
4	365	427	64	830	3	516
5	365	427	64	830	3	516
6	365	427	64	830	3	516
7	365	503	75	830	3	516
8	365	519	78	830	3	516
9	365	519	78	830	3	516
10	365	595	89	830	3	516
11	365	595	89	830	3	516
12	365	595	89	830	3	516
13	365	686	103	830	3	516
14	365	686	103	830	3	516
15	365	686	103	830	3	516
16	365	686	103	830	3	516
17	365	762	114	830	3	516
18	365	762	114	830	3	516
19	365	778	116	830	3	516
20	365	778	116	830	3	516
21	365	854	128	830	3	516
22	365	854	128	830	3	516
23	365	854	128	830	3	516
24	365	854	128	830	3	516
25	365	930	139	830	3	516
26	365	945	141	830	3	516
27	365	945	141	830	3	516
28	365	945	141	830	3	516
29	365	945	141	830	3	516
30	365	1,021	153	830	3	516
Totals	10,950	20,586	3,080	24,900	90	15,480

TABLE 8-5. MASS (MT) OF MISSION B PAYLOADS LAUNCHED PER YEAR

Year	Materials for Space Processing	Food for Space Station	Spares for Space Station	Propellants for LEO → GEO Crew Rotation	LEO Satellites	Propellants for Transport of LSS
1	234	105.6	22.0	261.2	1.4	162.8
2	234	105.6	22.0	261.2	1.4	162.8
3	234	105.6	22.0	261.2	1.4	162.8
4	234	134.4	28.0	261.2	1.4	162.8
5	234	134.4	28.0	261.2	1.4	162.8
6	234	134.4	28.0	261.2	1.4	162.8
7	234	158.4	33.0	261.2	1.4	162.8
8	234	163.2	34.0	261.2	1.4	162.8
9	234	163.2	34.0	261.2	1.4	162.8
10	234	187.2	39.0	261.2	1.4	162.8
11	234	187.2	39.0	261.2	1.4	162.8
12	234	187.2	39.0	261.2	1.4	162.8
13	234	216.0	45.0	261.2	1.4	162.8
14	234	216.0	45.0	261.2	1.4	162.8
15	234	216.0	45.0	261.2	1.4	162.8
16	234	216.0	45.0	261.2	1.4	162.8
17	234	240.0	50.0	261.2	1.4	162.8
18	234	240.0	50.0	261.2	1.4	162.8
19	234	244.8	51.0	261.2	1.4	162.8
20	234	244.8	51.0	261.2	1.4	162.8
21	234	268.8	56.0	261.2	1.4	162.8
22	234	268.8	56.0	261.2	1.4	162.8
23	234	268.8	56.0	261.2	1.4	162.8
24	234	268.8	56.0	261.2	1.4	162.8
25	234	292.8	61.0	261.2	1.4	162.8
26	234	297.6	62.0	261.2	1.4	162.8
27	234	297.6	62.0	261.2	1.4	162.8
28	234	297.6	62.0	261.2	1.4	162.8
29	234	297.6	62.0	261.2	1.4	162.8
30	234	321.6	67.0	261.2	1.4	162.8
Totals	7020	6480.0	1350.0	7836.0	42.0	4884.0

TABLE 8-6. SUMMARY LAUNCH TRAFFIC MODEL FOR REFERENCE ESRL CONCEPT

Year	Mission A*	Mission B	Total
1	0.5	5.8	6.3
2	1.0	5.8	6.8
3	1.5	5.8	7.3
4	2.0	6.0	8.0
5	2.0	6.0	8.0
6	2.0	6.0	8.0
7	2.0	6.3	8.3
8	2.0	6.3	8.3
9	2.0	6.3	8.3
10	2.0	6.6	8.6
11	2.0	6.6	8.6
12	2.0	6.6	8.6
13	2.0	6.9	8.9
14	2.0	6.9	8.9
15	2.0	6.9	8.9
16	2.0	6.9	8.9
17	2.0	7.1	9.1
18	2.0	7.1	9.1
19	2.0	7.1	9.1
20	2.0	7.1	9.1
21	2.0	7.4	9.4
22	2.0	7.4	9.4
23	2.0	7.4	9.4
24	2.0	7.4	9.4
25	2.0	7.6	9.6
26	2.0	7.7	9.7
27	2.0	7.7	9.7
28	2.0	7.7	9.7
29	2.0	7.7	9.7
30	2.0	7.9	9.9
Totals	57.0	206.0	263.0
30-year Daily Average	1.9	6.9	8.8

*Assumes 7-day work week.

Another issue to be considered in the assessment of the various applications is the requirement of an on-board propulsion system and the associated payload mass penalty for carrying the system. Any ESRL application which requires a velocity increment (besides the initial rail launcher boost) to change the projectile trajectory, such as an insertion into Earth orbit, will necessitate an additional propulsion system. Section 3.1.5.2 describes the procedure for estimating the size of the propulation system.

Other analyses necessary for examination of the defined ESRL applications are the safety and environmental impact assessments. Section 5.0 discusses the safety and environmental impact issues. The preliminary assessment indicates that for all ESRL launches the sonic boom at the remote launch site is not a problem to human health and safety, assuming reasonable safe distances are maintained. Repeated sonic booms could, however, pose a problem to the local ecosystems. Accidents involving toxic materials such as nuclear waste, ClF_3, N_2H_4, or other toxic materials pose only a localized impact if they land on remote land or the ocean. The payloads and propulsion systems are relatively small, and little significant environmental impact is expected from a launch accident. Worker impacts due to exposure to radiation or fumes from toxic propellants could pose significant risks. However, these risks would be minimized by proper radiation material handling procedures and proper propellant handling procedures. Another mitigation aspect would be to accept a lower performance by selecting a non-toxic propellant system, e.g. RP-1/LOX over N_2H_4/ClF_3.

Finally, an economic assessment was conducted. The cost information was derived from available concept information, based upon physical principles, and estimated costs for materials. The assessment of the candidate applications was made upon the basis of whether or not the application could fully support a large capital investment. An application which does not fully justify the development of the system, but which would be pursued if another agent developed it, was deemed a marginal application.

There is a variety of nearly equivalent criteria used in evaluations and assessing the economic value of any project before there is a firm commitment to proceed with development of the project. Governmental programs may have a benefit which is difficult to reduce to precise dollar terms (e.g. the benefit of research results), so the criterion for selecting between alternatives is the lowest total cost to accomplish the given objective. For programs where the benefits are sufficiently tangible that they can be related to dollar values, the criterion for selecting between alternatives is the highest discounted benefit-to-cost ratio, where all the serious contending alternatives will have ratios greater than one. Because money must usually be invested in alternatives well before benefits are received, both of these criteria are ususally discounted, i.e., adjusted for the implicit interest the investment would earn if placed in a secure paying investment (e.g. Treasury Bonds). Discounting is an additional burden on new concepts and is intended to make decision makers consider all the implications of selecting new ways of accomplishing tasks over existing ways. The use of discounting in making these decisions is thus an additional method of testing whether the new ways are significantly better than existing ways or those with lower-development costs. The applications were evaluated and then rated on a five point scale

(0 to 4). The numerical rating forces an additional subjective evaluation to the terms "marginally cost-beneficial". Some applications probably would not be considered unless the technology is available, but would then be highly advantageous. Other applications were either so uncertain or have other alternative methods of accomplishment that if the technology were available today, it might not be pursued because it would have the same or slightly higher costs as other means of accomplishing the same task. The use of the rating scale thus assigns a subjective opinion of where the application lies in this range. Figure 8-1 summarizes the initial ratings.

The nuclear waste disposal mission was ranked a 4, the highest. This was because it did offer the permanent disposal of nearly all the civilian high-level nuclear waste. On the other hand, toxic chemical disposal was ranked anywhere from a 0 to 1; it was felt that the energy used to launch a vehicle could be better spent neutralizing the chemicals here on Earth. Atmospheric research is rated between a 1 and a 2; the sounding rockets currently are readily available and they are relatively inexpensive. Small satellite launcher was rated a 3 because of the expected phase out of small launcher systems (e.g. Delta, Scout) and there is a need, but the demand is expected to be relatively low. Launch of materials for space use was rated between 3 and 4. It was felt that this mission perhaps would not justify the ESRL development by itself, but if the ESRL were developed for another application, such as nuclear waste disposal, it would be very advantageous. This mission would require an on-orbit propulsion to be included in the projectile. Chemical rocket assist is the subsonic horizontal launch that was rated somewhere between from 0 and 2. It was felt that the other alternatives would be more cost-beneficial, such as an subsonic airplane or a rocket sled. The hybrid rail launcher laser propulsion system was rated between a 2 and a 3. It seemed to be a good idea; however, there were two technologies that did need to be developed. The lunar gravity assist was rated between a 1 and a 3; it is expected that it would be a complicated system/approach. To accomplish the mission, considerable accuracies in the velocity and direction would be required. The interstellar and planetary probe concept is rated between a 3 and a 4; it appeared to have high merit; however, the high acceleration instrumentation would need to be developed.

8.2.1 Nuclear Waste Disposal in Space

An application of an ESRL system which appears to be very promising is disposal of high-level nuclear wastes in space. Vertical launches at a solar system escape velocity timed with the rotation of the Earth (6 hour launch window), would provide a permanent disposal of the waste. Only high-level waste is being considered for this evaluation. Additional waste mass, including transuranic (TRU) and radioactive gases could be considered in follow-on efforts. It may be possible to totally eliminate the need for mined geologic repositories.

The current U.S. Department of Energy (DOE) plan (U.S. DOE, 1980) for disposal of high-level and TRU radioactive waste is to place it in mined geologic repositories beneath the Earth's surface. Space disposal holds the promise of lower long-term risks than indicated in the DOE plan (Rice et al,

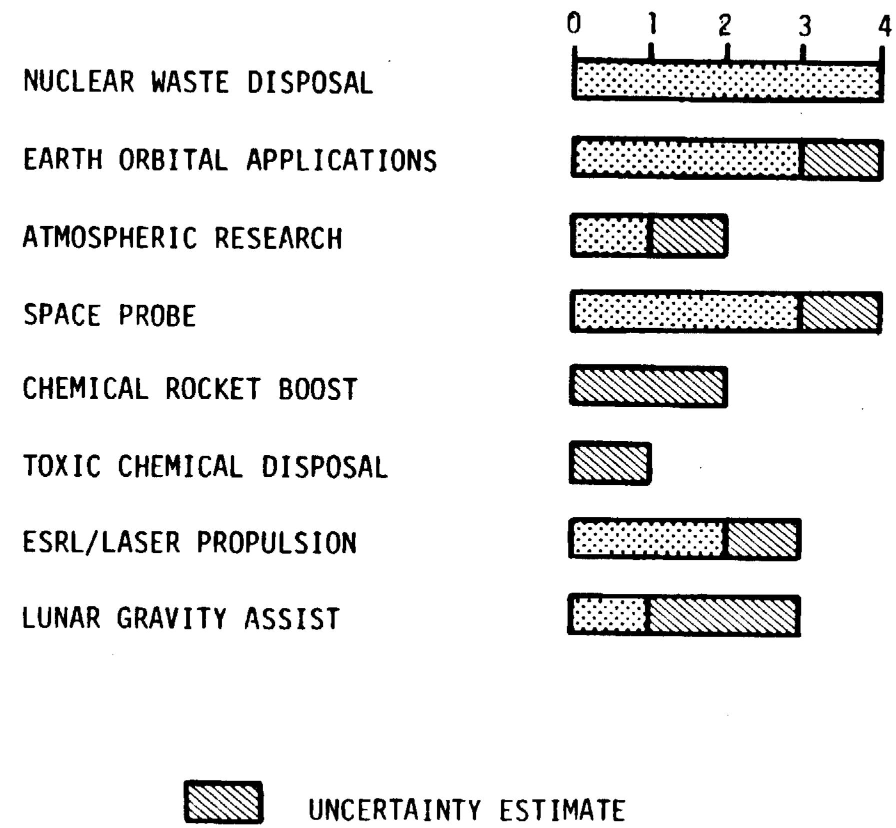

FIGURE 8-1. PRELIMINARY ECONOMIC ASSESSMENT OF ESRL APPLICATIONS

1982). Also, the reduction of "perceived" risks may be possible via space disposal.

For the "standard" space disposal concept (see Figure 8-2 for overall view), one Uprated Space Shuttle and one Shuttle Derived Vehicle (SDV) would be readied for launch for a given disposal mission. Pad C, which is to be constructed at KSC Launch Complex 39, would be used to launch the nuclear-payload carrying Uprated Space Shuttle. Existing launch pads A or B would be used for the SDV launch.

The SDV would be launched first to place the orbit transfer system/solar orbit insertion stage (OTV/SOIS) in a 370 km circular orbit inclined 38 degrees to the equator. The SDV propulsion and avionics module would reenter and be recovered for reuse. Approximately four hours after SDV launch, the Uprated Space Shuttle, with two spherical waste packages (shielded to 1 rem/hour at 1 meter), would be launched to rendezvous with the orbiting OTV/SOIS. The Shuttle Orbiter would approach the OTV/SOIS using its vernier thrusters. There would be a soft docking, at which point the Orbiter's attitude control would be shut down. Several hours later a transfer of the payload to the OTV/SOIS in the cargo bay of the Orbiter would occur. The Orbiter and OTV/SOIS would then separate and the Orbiter would back off from the OTV/SOIS payload. After the OTV delivers the nuclear waste payload and SOIS to the desired trajectory and returns to a low Earth orbit, the Orbiter would rendezvous with the OTV and return it to the launch site to be refurbished for use on a later mission.

When the OTV/SOIS/waste payload system has passed final systems checkouts, the OTV propulsive burn would place the SOIS and its attached waste payload on the proper Earth escape trajectory. Control of the propulsive burn from low Earth orbit would be from the aft deck payload control station on the Orbiter, with backup provided by a ground control station. After the burn is complete, the SOIS/waste payload is then released. In approximately 165 days the payload and the cryogenic LOX/LH_2 propellant SOIS will travel to its perihelion at 0.85 A.U. about the Sun. [One astronomical unit (A.U.) is equal to the average distance from the Earth to the Sun.] The SOIS will place the payload in its final space disposal destination by reducing the aphelion from 1.0 to 0.85 A.U. To aid in obtaining the desired orbital lifetimes, this orbit will be inclined to the Earth's orbital plane by 1 degree.

Recovery burns using the remaining OTV propellant and aerobraking would return the OTV to low-Earth orbit for rendezvous with the Shuttle Orbiter for subsequent recovery, refurbishment, and reuse of the OTV on a later mission.

8.2.1.1 Requirements

The major mission requirements that have evolved during the study are summarized from Sections 2.0 and 4.0:

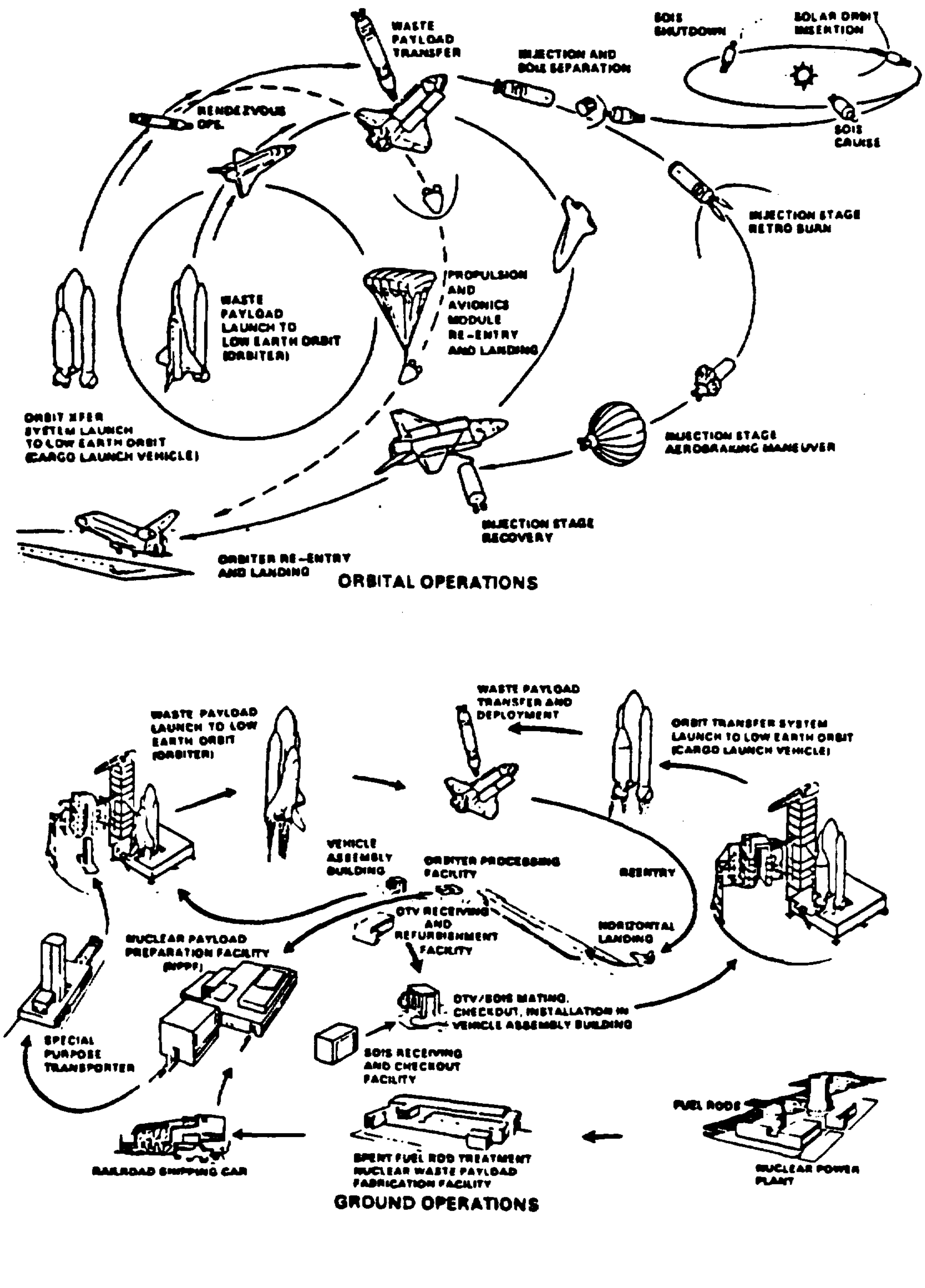

FIGURE 8-2. OVERALL VIEW OF "STANDARD" SPACE DISPOSAL MISSION

- Equatorial launch site
- Acceleration limit = 10,000 g's
- Launch velocity for solar system escape = 20 km/s
- Elevation angle = 90 degrees from horizontal
- 500 kg waste form to be launched per day to keep pace with expected waste generation
- Six-hour launch window.

The payload contains high level wastes from U.S. domestic power plants. The waste form is made up of fission products and actinides, including 0.5 percent plutonium and 0.1 percent uranium, with 95 percent of the cesium and strontium removed. It is assumed that the waste has been aged approximately 50 years.

The requirement to launch 500 kg waste form per day necessitates two launches per day. This is due to ESRL restraints on payload mass (2055 kg) which leaves 250 kg for the cermet payload.

8.2.1.2 Assessment

Using an ESRL system to launch high-level nuclear wastes out of the solar system has, at first glance, several advantages over other concepts. It offers permanent disposal with virtually no long-term risks. Its single shot method uses no upper stages, so there are no propulsion system reliability problems. It offers a quick and potentially low risk method of disposing of the domestic high-level nuclear waste.

With the disposal of nuclear waste in space via the ESRL concept, it is possible to reduce the calculated and perceived release risk (of other concepts) to the biosphere of radioactive material. Rough estimates of release risk based upon data in Rice et al, 1982, indicate that a mission reliability of 99.9 to 99.99 percent may be required to match the risk of standard space disposal. The perceived risk benefit could be the more important factor (see Rice et al, 1982). The use of the ESRL concept would remove the fear of an on-pad Shuttle-type catastrophy near a population center (area surrounding KSC, Florida). The 19,000 ESRL launches equivalent to 1500 Shuttle vehicle launches to dispose of the nuclear waste would reduce total energy, materials consumption, and chemical pollutant releases to the atmosphere. Given that one upper atmospheric burnup/upper atmospheric dispersion occurs for each scenario, the radiological health risk would be considerably lower for the ESRL system, because of the smaller payloads involved.

The costs for disposal in space using existing or near-term launch vehicles are sufficiently high that space disposal is currently being considered for only selected isotopes of especially high biological hazard. For the adaptation of current technology (i.e., Uprated Space Shuttle), we have estimated that the recurring launch vehicle transportation costs would be approximately \$20,000/kg (1981 \$). For an equivalent 100,000 MTHM mined repository, disposal costs are estimated at \$50 to 100 B (personal communication with R. E. Best, Office of NWTS Program Integration, Columbus, Ohio).

For a repository partially complemented via standard space disposal (disposal of HLW) the estimated cost is \$40 to 80 B (Best, 1982), with the cost for space disposal additional. Based upon the above data, estimates of ESRL costs, and other data found in Best (1982), it would not be cheaper to complement the mined geologic repository (MGR) with space disposal. From a cost standpoint it might make sense to dispose of TRU wastes along with the HLW (or fuel rods) in space to eliminate the MGR, and hence save a significant investment (\$50-100 B/100,000 MTHM repository). This would not be possible with the standard space disposal concept (costs are too high).

It is thus obvious that if space disposal of high-level wastes is desired; alternatives can support significant development and recurring expenditures and still be justified. For this reason, ESRL launch of nuclear waste materials is an application which can fully support envisioned research and development costs toward an operational rail launcher system. An ESRL disposal system would have lower recurring costs and possibly have lower development expenditures than those for new highly-advanced launch vehicle systems.

8.2.2 Earth Orbital Missions

An ESRL system could be used to perform a variety of Earth orbital missions. These missions would include satellite launches, space station resupply, and materials delivery to low Earth orbit (LEO). Other possibilities are launches to geosynchronous orbit, but these are not studied here.

The ESRL could be used as a launcher for small satellites into Earth orbit. Satellite mass limits to LEO for these vehicles currently range from 270 kg (Scout-class) to approximately 2140 kg (Delta-class). For the ESRL application, satellite masses would likely be in the range of approximately 250-500 kg. The payload would consist of the satellite and a liquid propulsion system necessary to inject the satellite into orbit. The satellites envisioned would support Earth observation and scientific missions.

Because of the high accelerations involved in an ESRL launch, space station resupply items are limited to those of a bulk nature. For example, no delicate instruments would be launched. However, items such as propellants, food, or some spare parts could be launched this way to support an orbiting space station.

An additional mission which may be attractive for a rail launcher system is delivery of materials to a GEO-based fabrication center. Should the Solar Power Satellite (SPS) concept be revived again, this would be an excellent launch medium for structural materials. Previous studies (U.S. DOE and NASA, 1978) indicate an SPS deployment rate of 2 per year. The mass of each SPS (Silicon option) was determined to be 50,980 MT. Assuming that half of the mass is transportable by ESRL, an annual ESRL launch rate of 50,980 MT is indicated. A payload density of 2.5 (the density of silicon is 2.33 and of aluminum is 2.7) was used as a preliminary estimate to find the number of ESRL launches required (based on the current Reference Concept), and results in

80,284 launches per year. This gives an average daily launch rate of 220 launches per day. This mission may prove attractive, however, it is not considered further in this assessment.

8.2.2.1 Requirements

The major mission requirements for Earth orbital missions are summarized from Sections 2.0 and 4.0 and are listed below:

- Muzzle launch velocity for LEO = 5-10 km/s
- Additional propulsion system required
- Acceleration limit = 2500 g's
- 90 degrees azimuth
- Elevation angle = 20 degrees from horizontal
- Maximum projectile mass = 6500 kg (~650 kg payload).

Velocity requirements for orbital launches are on the order of 6 to 7 km/s for LEO missions and 12 to 13 km/s to reach geosynchronous altitudes (see Section 3.1.3.2 for details). Additional propulsion systems are needed to give the projectiles the velocity increment, Δv, necesary for insertion into the desired orbits. For LEO missions, a Δv of approximately 2.1 km/s is required and a Δv of approximately 1.7 km/s is required for geosynchronous missions.

The actual payload mass will vary depending upon the density of the payload material. Figure 3-18 illustrates the relationship between payload density and mass. To estimate a traffic model (Tables 8-4 and 8-5), densities had to be assigned to each payload type. The density of the food was assumed to be close to that of water (~1.0 g/cc). Spares and satellites were assumed to be somewhat heavier (~1.5 g/cc). Structural materials and materials for space manufacturing were assumed to have densities similar to that of aluminum (~2.7 g/cc).

The number of launches per year is function of the available payload material and its density. The amount of food to be launched for space station support depends solely upon the number of personnel in space. From the MDAC Space Station studies (1970), approximately 480 kg of food is required per year for each person. Spares to be launched by ESRL were estimated at 100 kg/person. This figure is approximately one-third of the number used in the MDAC Space Station reports (3800 kg/yr/12 persons). Many of the spares mentioned were considered unsuitable for ESRL launch, such as filters. The estimate assumes compact spare parts. Propellants would be launched to supply chemical vehicles to be used in LEO to GEO crew rotations (assumed a 90-day rotation) and for transport of large space structures (LSS). Propellants would be launched as water to be transformed to O_2 and H_2 on orbit. Water has an O_2/H_2 ratio of 8:1, while most propellant systems have ratios of 6:1. Therefore, the additional oxygen yielded in an electrolysis procedure in orbit could be used in the space station life support system or as an attitude control jet gas. From Kunz (1980), a low-thrust modular chemical system to transport LSS requires five propulsion modules for LEO to GEO transfer of

60,500 kg LSS with 25,300 kg of propellant per module. Demand for LSS transport was estimated at one trip per year to geosynchronous orbit using a low-thrust chemical system.

Materials processing in space has the potential for yielding a number of new and improved products. A list of some potential products is given below (Wuenscher, 1972):

- glasses
- alloys and intermetallic compounds
- particle dispersed metallic composites
- whisker dispersed metallic composites
- density-controlled metallic composites
- crystals
- improved material configurations
- bio and chemical compounds (from antibiotics to polymers).

Raw material supply requirements were estimated at 161,600 kg/yr for processing tungsten-nickel high-temperature eutectics alone (Bloom, 1977). This corresponds to an ESRL launch frequency of 250 per year. It is not difficult to imagine other applications which would raise the supply requirements much higher. For the ESRL traffic model, a launch rate of one per day was assumed. Satellites launches to LEO were estimated at three launches per year. The types considered were Earth observations and various scientific satellites.

To save costly amounts of orbit transfer system propellants for resupply missions, it would be highly desirable to place the payload as near the space station as possible. This placement requires matching the orbital planes and phasing. Satisfaction of these two requirements can be obtained by planning conditions such that alignment will occur, by using large amounts of propellants, or by spending time in an intermediate drift state until alignment is possible with an economical expenditure of propellants.

For most rendezvous missions, the last two options are not feasible, since long drift times are required unless the misalignment is small and large amounts of propellant are generally not available. Therefore, preplanned alignment is the only practical option. Orbital plane alignment requires launch to occur nearly in the plane of the space station. Phasing should be fixed when the plane passes over the launch site. This requirement then reduces to a condition on closed ground tracks. The condition that a circular orbit have a closed ground track depends upon its altitude and inclination. Ground tracks may be closed each plane alignment ("one-day" closed ground tracks), every other plane alignment ("two-day" closed ground tracks), etc. Note that the time between alignments in general is not "one day", but more like 23.5 hours (sun synchronous orbits are the exception). Figures 8-3 and 8-4 show the altitude-inclination relation for one and two-day closed ground tracks respectively. Figures 8-5 and 8-6 illustrate the one day closed ground track for a 500 km orbit at 28.5 degree and 55 degree inclinations.

With a space station in LEO at 0-degree inclination, the period of orbit is approximately 1.6 hours. At 0-degree inclination the ground tracks

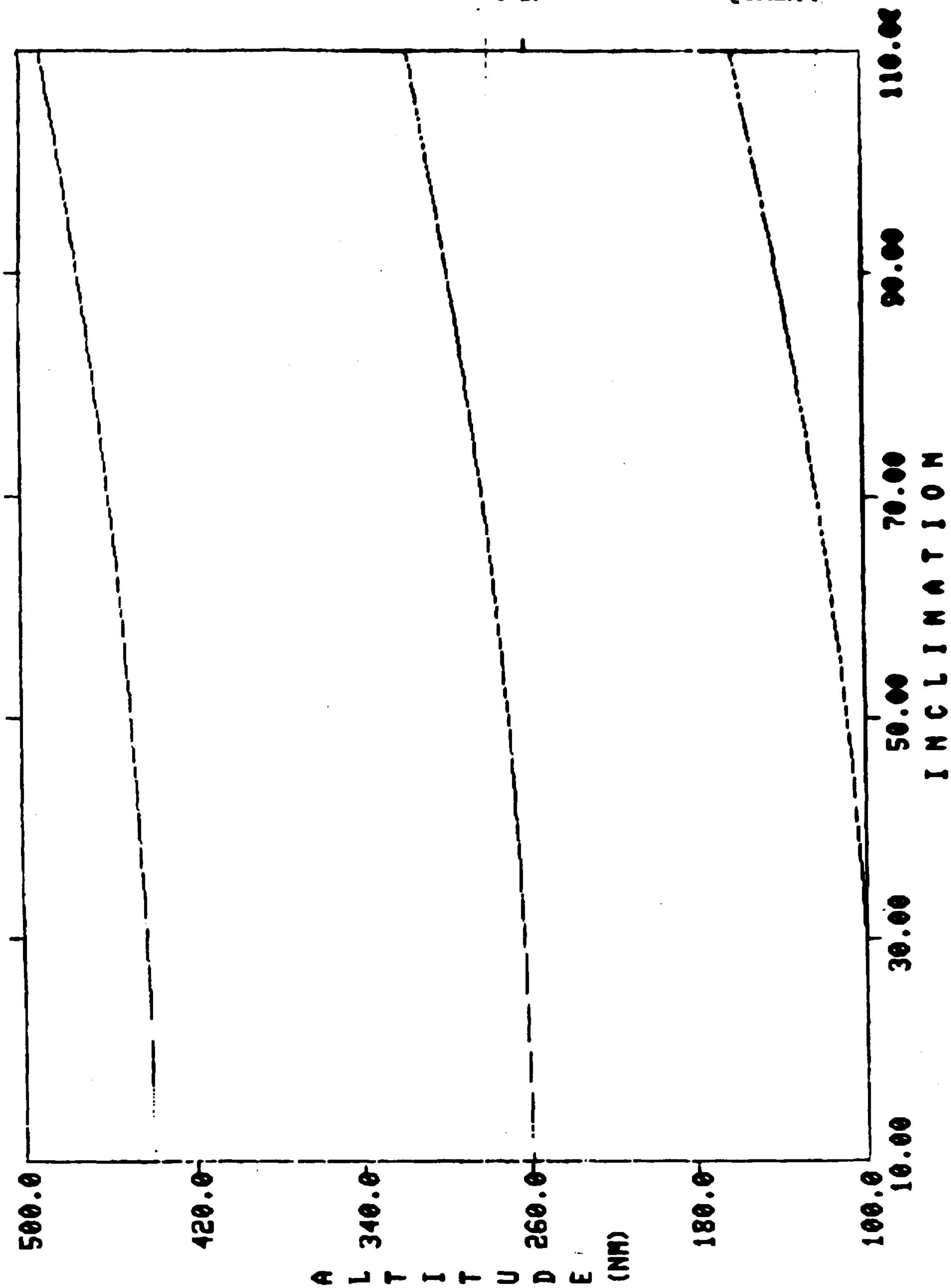

FIGURE 8-3. ALTITUDE-INCLINATION RELATION FOR ONE-DAY CLOSED GROUNDTRACKS

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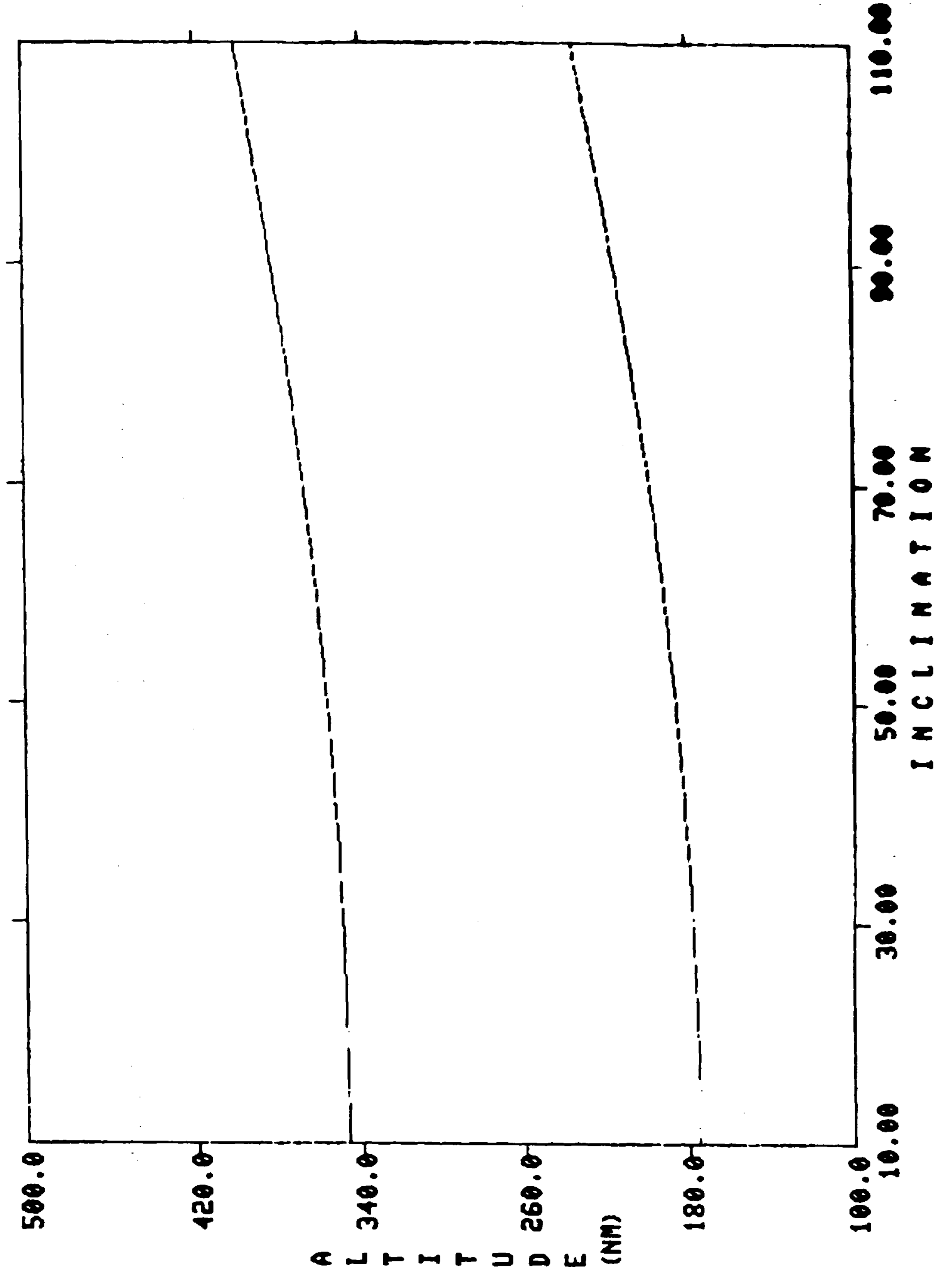


FIGURE 8-4. ALTITUDE-INCLINATION RELATION FOR TWO-DAY CLOSED GROUNDTRACKS

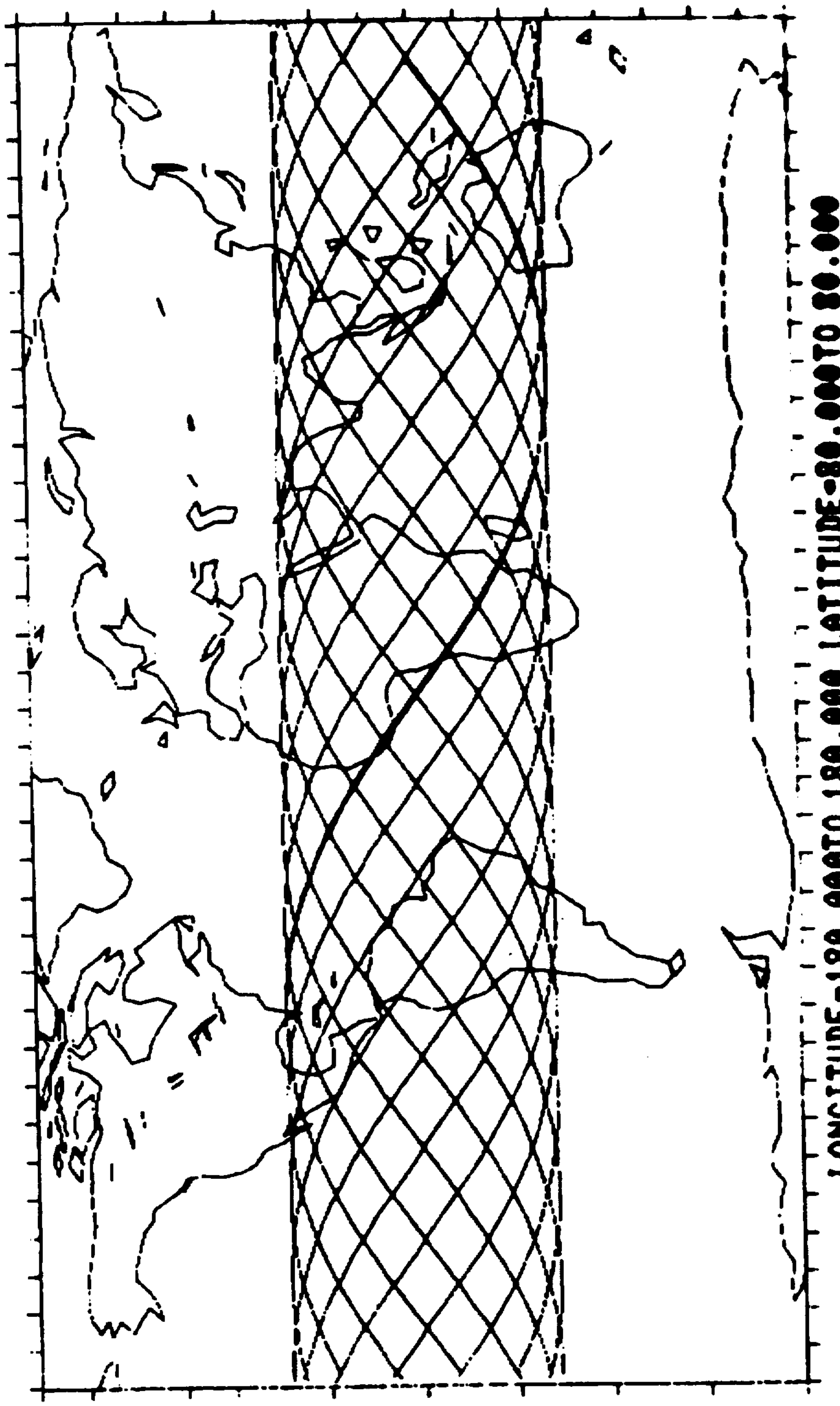
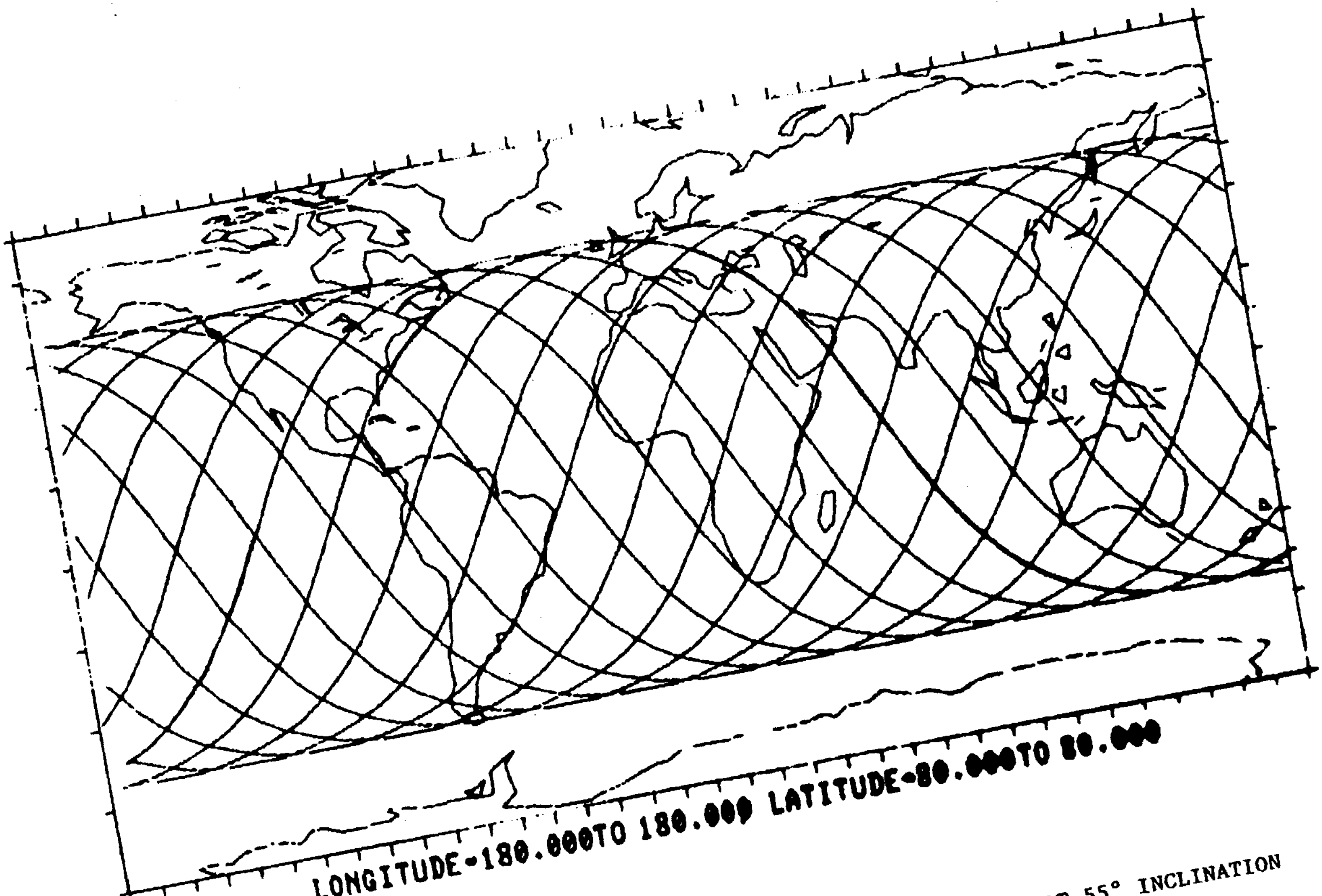


FIGURE 8-5. ONE-DAY CLOSED GROUNDTRACK FOR A 500 KM ORBIT AT 28.5° INCLINATION



FOR A 500 KM ORBIT AT 55° INCLINATION

are closed each revolution, so a launch opportunity occurs at each pass. The current Reference Concept allows up to approximately eight launches per day, although more could be accommodated by increasing the onsite power capacity.

8.2.2.2 Assessment

The Earth orbital application was rated potentially cost-beneficial (3 to 4 in Figure 8-1), which means that the application would most likely be pursued, perhaps even if the technology was not already under development for another mission application. Because of the potential for economic benefit, the Earth orbital applications (Mission B) became part of the Reference Concept.

The launching of materials for space station resupply and materials processing items would typically be limited to bulk materials such as propellants (water), structural materials, spares, processing materials, and food and water. The Shuttle and a cryogenic upper stage concept (when needed) would require expenditures on the order of $1000 to $2000/kg to deliver payloads to LEO and on the order of $5000 to 10,000/kg for geosynchronous orbit. The advantage of the conventional alternatives is that they can deliver all types of cargo, including delicate instruments and personnel, as well as bulk commodities. An ESRL development would only be feasible if large bulk masses were to be delivered to GEO.

Currently, satellites in the mass range of up to 1000 kg are launched into LEO with either the Scout or the Delta. The fully burdened cost of these vehicles at the presently low Scout launch rate of 1 to 2 per year is in the range of $7 to 8 million per launch ($25,900 kg), while Delta with a LEO launch rate of approximately 2-3 per year costs roughly $27 million per launch ($12,600 kg). This relatively high recurring cost suggests that this ESRL application would be economically desirable if rail-launcher recurring costs were low, and especially so if there was some flexibility in launch azimuth (but the Reference Concept does not show it). For infrequent individual satellite launches to LEO only, using a dedicated ESRL, the costs would be higher than those of conventional means. Other uses are necessary to provide cost benefits.

The use of an ESRL would reduce the chemical pollutants released to the lower and upper atmosphere, if the Shuttle launch rate was reduced. Less energy and materials would be consumed with a lower launch vehicle activity. The risks posed by the use of toxic propellants would not be expected to be greater than those posed by propulsion systems accommodated by the Shuttle. However, the nose cones and payload support structures of projectiles would reenter and burn up after each flight. Care would need to be taken to avoid reentry/impact risk to the public. Currently, for a launch of the coast of South America, reentry impact is expected to occur over the Pacific Ocean. No significant mission peculiar safety or environmental problems are apparent at this time.

Mission B (LEO applications) has become part of the Reference Concept and is combined with the nuclear waste disposal mission which would by itself fully support the development and construction of the ESRL system. Space station resupply and materials would be the primary payloads of the Mission B launcher tube, with several satellites per year as well. Use of the same power plant and support facilities is an attractive feature, with nuclear waste launches occuring a few hours on either side of dawn and the LEO missions occurring as often as every 1.5 hr scattered throughout the rest of the day.

8.2.3 Atmospheric Research

Upper atmosphere research is currently conducted largely using sounding rockets. The NASA sounding rocket program has supported meteorology, astronomy, physics, and planetary atmosphere studies. These vehicles have also flight tested equipment intended for later use on satellites. A rail launcher system could also perform the activities of this program.

Sounding rockets are available in a variety of sizes and payload capabilities. The sizes range from 3 m (Arcas) to 16 m (Aerobee), and reflect the number of stages available. The maximum payload capability is 1000 kg to 350 km altitude on the Aries sounding rocket, while the peak altitude available is 1000 km using the Terrier-Malemute rocket with a 60 kg payload. The Aerobee rockets are currently being recovered after launch and refurbished for later use.

There are many launch sites used by NASA in their sounding rocket program. The sites that are used most are Wallops Flight Center (Virginia), White Sands Missile Range (New Mexico), and Churchill Research Range (Canada). The highest launch rate, 175 launches, occurred in 1968. Recent figures indicate that between fifty and sixty sounding rockets are launched each year.

8.2.3.1 Requirements

Using the methods of Section 3.1.3, ESRL launch velocity requirements range from 3-5 km/s depending upon the altitude desired. An easily transportable system would be highly desired as indicated by the number of launch sites for the NASA program alone.

A nearly vertical launch is desired, since any other angle increases the projectile range and makes recovery more difficult.

8.2.3.2 Assessment

The ESRL system necessary to support the atmospheric reasearch application appears to be technically feasible, and, in fact resembles the Mission B launcher of the previously-identified Reference Concept (Section 4.0) except for the launch angle. The Mission A tube could be used if vertical launches are desired.

Environmental benefits here are not significant because of the expected few number of launches. However, small benefit is possible, again, due to reduced chemical pollutant releases (of ESRL as compared to the sounding rocket program activity). Recovery of the ESRL projectiles would be possible as is the case for the sounding rockets in the current program.

The atmospheric research application was deemed to be marginally cost-beneficial by itself alone. When "piggybacked" with other missions it looks attractive. Atmospheric research has been conducted mainly by sounding rocket launches. While the researchers prefer to work from a fixed base, these rockets are readily transportable and have been launched from every continent and from remote locations such as the Canadian Arctic to obtain information on the geographic variations in the upper atmosphere. Sounding rockets come in large variety and cost from $100 to $1 million per launch depending upon size, number of stages, and guidance requirements. It is sometimes possible to recover and refurbish them economically. Because of the relatively low launch rates and costs, the adaptation of railgun technology for this application alone is believed to be economically marginal. If the technology is proven by others, and it is possible to construct an economically transportable railgun, preferably with flexibility in launch azimuth, it is likely to be desired for repetitive launches. Multiple soundings could be made from the same location during the day perhaps, to obtain knowledge of diurnal variations in atmospheric characteristics, and payloads could probably be recovered and reused. However, the existance of relatively low-cost sounding rockets is believed to prevent this application from becoming the justification for a dedicated development of railgun systems suited for only this purpose.

With launch velocities in the range from 3 to 5 km/s, this sounding-rocket-type missions could be launched from the Mission A or Mission B tubes (see Reference Concept). However, since the launcher tubes are fixed (20 degree elevation, 90 azimuth and vertical) there would not be much demand for the ESRL system (maybe ten launches per year).

8.2.4 Deep Space Probes

A rail launcher could be suited for launching several types of deep space probes including:

- Deep interplanetary
- Planetary fly-by
- Solar
- Interstellar.

The rail launcher is not as well suited for planetary probes which orbit or land on the planet, but it could be used for these missions as well.

Interplanetary probes generally conduct fields-and-particles experiments. Because these types of experiments have no particular target, there is no need for midcourse or terminal guidance. This presents an attractive application for a direct ESRL launch.

A launch velocity of approximately 13-18 km/s is needed for a planetary fly-by mission, depending upon the target planet (Koelle, 1961). Representative launch velocities are shown below:

Target	Launch Velocity
Mars probe	13.15 km/s
Venus probe	13.0 km/s
Mercury probe	15.1 km/s

The fly by probe requires a midcourse correction capability, generally in the 10-500 m/s range. The planetary fly-by mission nominally includes a TV camera to send pictures back to Earth, which may limit launch accelerations. High-g instruments and sensors would likely have to be developed. Technology developed for gun-launched laser systems and instrumentation would be appropriate for this application.

Solar probes are designed to operate very near the Sun. High launch velocities between 19 and 34 km/s are required, but no on-board propulsion system would be necessary. Another type of probe requiring high launch velocities is the interstellar probe which requires solar system escape velocities to venture beyond the outer planets. For probes designed to take interstellar samples, no midcourse or terminal guidance system would be required, however, attitude control propulsion would probably be needed to maintain antenna pointing for information transfer. If a projectile were launched at about 30 km/sec (at the right time) it would take roughly 40,000 years for it to reach the nearest star.

8.2.4.1 Requirements

From the previous discussion, velocity requirements for most probes range between 13 and 20 km/s, depending upon the type of mission desired. Midcourse guidance may be desired, but the propulsion systems would be small in comparison to those needed for orbital insertion.

Masses of probes generally range from 500 kg to 1000 kg. A rail launcher system capable of launching these masses would be required.

Because of on board instrumentation, accelerations should be limited. A limit of 10,000 g's is thought to be acceptable.

8.2.4.2 Assessment

With medium to high launch velocities and similar payload masses, an ESRL system for the space probe and nuclear waste disposal missions are technologically similar. In fact, the ESRL Reference Concept, although designed for the nuclear waste mission, would launch probes per demand (estimated at one to four launches per year). For the Reference Concept a nominal payload mass of 650 kg was used. This figure is higher than the waste payload since the heavy shielding is not required.

Environmental benefits are not significant/measurable because of the expected few number of launches.

The probe mission was deemed marginally cost-beneficial. This mission, when carried out with an already developed ESRL system would be highly advantageous because of the substantial costs for the presently-used launch vehicles (both booster and upper stages). However, the immediate benefits of research are generally intangible and launch vehicles currently are available for accomplishing these missions. These facts, despite the benefits in launch cost reductions, resulted in the marginal rating.

8.2.5 Chemical Rocket Boost

Another application considered was the use of an ESRL system to launch a chemical rocket system by providing a portion of the initial velocity under low-g conditions. This would have the ESRL be the "first stage" of the rocket. The U.S. Air Force Rocket Propulsion Laboratory has recently funded a study to evaluate this concept. At this writing, no information is available on this feasiblity study. Additionally, a boost might be given to a larger vehicle such as an advanced Space Shuttle. The initial boost would be on the order of several hundred meters per second, similar to other concepts under consideration such as using an SSME-based rocket sled for the same purpose (Bissell, 1981).

8.2.5.1 Requirements

As opposed to the other applications, the acceleration for a Shuttle boost would need to be approximately 3 g's since this is a manned application. However, because of the lower launch velocities (approximately 200 m/s), the length of the rail launcher required is approximately 700 m and would most likely be horizontal, but could be vertical. The requirements are summarized as:

- Launch velocity = 200 m/s
- Acceleration limit = 3 g's
- Launch mass = 2000 MT.

Two concepts are possible: (1) the railgun launcher drives a piston which is attached to the vehicle; (2) the vehicle is saboted in the large rectangular bore. The first concept appears to be most practical. The size of this system would be on par with the Reference Concept described in Section 4 and costed in Section 6.

8.2.5.2 Assessment

The advanced Shuttle boost is an additional extension of railgun technology beyond other options studied. Environmental impact benefits are possible for this application. If a nonpropulsive coast period were possible, the environmental effects, at the immediate ground area, due to rocket thrusting (launch noise, rocket effluents) would be reduced.

The advanced Shuttle boost concept is essentially an electromagnetic catapult, and as such would be in competition with other catapult concepts (e.g. hydraulic, rocket or electromechanical). Because of the size of the Shuttle, these concepts would exhibit large costs which are believed to have approximately the same order of magnitude as those costs for the design, development, and first unit of a subsonic aircraft which would accomplish the same task. The aircraft could also give the Shuttle (or its derivatives) additional kilometers in altitude, which could not be obtained from catapults unless built on a mountainside. These concepts all would significantly reduce the Shuttle booster (current solid rockets or replacement) weight and cost requirements. However, since none of the alternatives are well defined, judgement of the most economically attractive concept cannot be made at this time. If the technology for an ESRL system of this size were available, the costs would be sufficiently known that these judgements may be made. Accordingly, the applications of using an ESRL to boost an advanced Space Shuttle was rated as marginal, with a high uncertainty as to its competitiveness with other methods.

8.2.6 Toxic Chemical Disposal in Space

The many problems of toxic chemicals are widely publicized. A rail launcher offers a means of permanent disposal by launching at solar system escape velocities. There are two basic methods of dealing with chemical hazards: disposal and treatment. Disposal methods deposit the toxic chemicals on or into land or water. With disposal, the toxic constituents may be released into the biosphere. Disposal methods include land fills and surface impoundment. Treatment of toxic chemicals may be by physical, chemical, or biological methods. The purposes of treatment include detoxification, neutralization, and volume reduction of the hazardous chemicals. Some common treatment methods are incineration, acid neutralization, and cyanide reduction.

Since there are many toxic chemicals, candidates for space disposal should be limited to those which pose a serious threat and which are highly resistent to decomposition in soils. Examples of possible candidates are the polychlorinated biphenyls (PCBs) and nerve gas (really a biological hazard). These are so stable that they persist for years and are recycled in the food chain.

8.2.6.1 Requirements

Because of the nature of the toxic chemical disposal mission (solar system escape), the launch requirements are virtually identical to the nuclear waste disposal mission (Sections 8.2.1.1, 2.0 and 3.0):

- Launch velocity = 20 km/s
- Acceleration limit = 10,000 g

- Launch site on equator
- Vertical launch
- Several launches per day.

The requirements for the chemical container may be slightly less stringent than those for the nuclear waste. The container should be designed to ensure no discharge prior to and during the flight, and it should be able to withstand the acceleration intact with a large safety factor.

8.2.6.2 Assessment

The ESRL system required for space disposal of highly toxic chemicals is identical to that of Mission A, nuclear waste disposal in space. This application is not rated as highly, however, because of the nature of the payload, not the mission.

This need might be best served by destroying the chemical species of concern using the energy that would have been spent in launching it, either conventionally or with the ESRL system. Impacts would be similar to those related to the nuclear waste disposal mission, except for possible catastrophic accidents, one would have toxic chemical release as opposed to radioactive material release.

Economically, toxic chemical disposal in space is perhaps, at best, a marginal application of railgun technology. In contrast to high-level nuclear waste disposal (Section 8.2.1) with extremely long half-lives so that neutralization requires time or distance, it is difficult to envision a toxic chemical which could not be neutralized with application of the energy needed for launch into space. If a rail launcher existed, it might be argued that it would be simpler to launch small amounts of chemicals into space than to accept the hazards of processing; however, the hazards of launch container loading are approximately the same as those of reaction vessel loading. These same arguments apply to biological hazards as well, but not to the high-level nuclear wastes.

8.2.7 Hybrid ESRL/Laser Propulsion

The combination ESRL/laser system is another form of boost system. The rail launcher would boost a payload containing a hydrogen propulsion system. When the payload reaches orbital altitude, an orbiting laser would direct its beam onto a collection window to heat the hydrogen. This physical reaction would propel the payload to orbit, preventing reentry.

8.2.7.1 Requirements

Initial launch velocities of 6-8 km/s are required to reach orbital altitudes and perhaps provide enough time for the system to provide the needed Δv. For a simple orbiting laser, launch opportunities should be available at least every 1.5 hours with a comparatively large launch window. A launch at 20 degrees from the horizontal would be preferred.

8.2.7.2 Assessment

In this application, not only must the rail launcher technology be developed, but on-orbit laser system as well. State-of-the-art laser technology should be sufficient to carry out the application. Lasers are currently used to induce and control chemical/physical reactions. Advantages of using lasers for this purpose include: energy delivery is remote; energy can be supplied on orbit by a solar energy collector with storage; and the projectile would not have to carry an oxidizer. Orbiting laser platform technology must be developed in conjunction with this concept. It is, however, a field which is currently getting much attention.

The safety and environmental aspects of this application are similar to others already mentioned. Misguided laser beams present the most significant hazard.

The economics of the hybrid ESRL/laser system are largely determined by the costs of alternative propulsion systems. Costs for conventional rocket launches currently range from approximately $8 million per launch for Scout up to $60 million per launch for more capable systems. The ESRL/laser concept has the potential to be fully supported by recurring cost savings over launch vehicles, but the development costs (both ESRL and laser) are expected to be high and uncertain. High development costs could make the application less desirable than launch vehicles. This hybrid propulsion system is thus rated as marginally cost-beneficial. If the system were included with the Reference Concept mission scenario, it could be cost-effective.

8.2.8 Lunar-Gravity-Assisted Launch

The lunar swingby launch is a concept designed to use the moon's gravitational force to alter the ballistic trajectory of an ESRL-launched payload preventing reentry, and at the same time eliminate the need for an additional propulsion system on-board the projectile.

8.2.8.1 Requirements

The ESRL launch would require great directional and velocity accuracy, as this is a precision maneuver. Midcourse guidance would be needed to reach the proper location in the moon's sphere of activity. The velocity requirements are nearly as high as those for Earth escape (ideally 11.2 km/s).

The payload characteristics depend upon the type of mission, but masses are generally believed to be in excess of 1000 kg.

8.2.8.2 Assessment

The geometrical constraints on the launch are severe and need further study before firm conclusions about this application can be drawn.

Environmental impact assessment indicates that this mission would be similar to the planetary probe launch, but at lower velocities. Less potential for severe accidents (little toxic propellant on-board) exists. The environmental benefit would be to reduce the quantities of chemicals released to the atmosphere due to Shuttle-type launches needed to support the missions in question.

Economically, this application is rated marginal at best. To launch payloads in this manner solely to eliminate onboard propulsion systems is costly in terms of the orbit transfer system propellants needed to transport the payload to the desired orbit. Due to the launch uncertainties, it is considered only marginally cost-beneficial.

8.3 Economic Assessment

A brief economic assessment of the current Reference Concept (see Section 4.0) has been conducted and is reported here. The purpose of this assessment is to provide an appreciation for the range of ESRL costs in relation to the costs of existing and near-term methods for Earth-to-space propulsion.

Two economic comparisons are made: (1) nuclear waste disposal in space to be accomplished by the ESRL (Reference Concept) or by advanced Space Transportation System (advanced STS) components; and (2) delivery of bulk supplies to Low Earth Orbit (Bulk to LEO) by the ESRL or by three STS Vehicles: (a) the Current Space Shuttle, (b) a liquid rocket boosted (LRB) Uprated Space Shuttle, and (c) an unmanned Shuttle Derivitive Vehicle (SDV--Orbiter replaced by SSME engine pod and cargo shroud). Because if the great differences between the alternative methods of accomplishing the same task, the basis for comparison of the economic benefit is the cost per kilogram for transporting the same mass during a 30 year period. (This period was selected because it is the expected operational life of the ESRL system and it reduces the impact of development costs on the fully burdened unit costs, in a manner fair to all alternatives.)

The results of these comparisons are then used, together with other information developed in Section 6.0 to provide a graph of the effect of varying launch rates on transportation costs. The independent variable of dollars (1981 $) per kilogram and the dependent variable of mass transported per day were selected as the common figures of merit for the two very different methods of transportation.

The information is presented in three subsections: Section 8.3.1 presents information on the existing and near-term space transport vehicles and establishes the mass transport equivalence between this mode and the ESRL.

Section 8.3.2 presents the actual costs used and the results of the dollars per kilogram computations for the specific cases selected. Section 8.3.3 presents the effect of varying mass delivery rates on the costs in terms of dollars per kilogram and presents the conclusions drawn from this analysis.

8.3.1 Conventional Space Transport and ESRL Equivalence

The current Shuttle/STS is well along in the flight test phase and nearing its operational phase at the time of this writing and its ultimate capabilities and costs are yet to be determined. In addition, there are a variety of proposals to upgrade the existing STS systems as well as adapt STS components to provide an unmanned cargo vehicle (the SDV). To proivde a self-consistent set of estimates, information provided by Frank Williams (Martin-Marietta) and Mike Van Hook (MSFC) was used and is presented in Table 8-7 together with some of the information used to make later adjustments for comparability. These adjustments relate to conversion to 1981 dollars, growth in expected STS costs be compatible with current NASA estimates, and the mass delivery capability. The Boeing data consider gross payload to 370 km orbit and the ESRL Reference Concept calls for delivery to a 500 km orbit. The 500 km orbit is selected because it is currently viewed as being the most likely to be used for long-term space facilities. The gross payload estimates shown in Table 8-7 are for a 500 km orbit. A load factor of 75 percent is then applied for all types of equipment needed to contain, manipulate, and deliver the bulk payload.

The equivalence between conventional space transportation and the ESRL is established in Table 8-8, on the basis of equal mass flown during a 30-year period. Table 8-8 presents results for both the nuclear waste disposal mission (Mission A) and the transport of materials to LEO (Mission B). To assist in the interpretation of this table, it must be noted that the current Space Shuttle was not a prime candidate to perform the high-level nuclear waste disposal in space mission (Rice et al, 1982). For "conventional" space disposal, the uprated Space Shuttle, the SDV, and two upper stages are required to achieve a heliocentric disposal orbit at 0.85 AU (see Reference Concept description in Rice et al, 1982). These two launches (see Section 8.2.1) dispose of 6.3 MT of cermet HLW, or 3.15 MT per lower stage booster launch. All the Shuttle vehicles can perform delivery of bulk to LEO.

8.3.2 Comparative Transport Costs

The transport costs used in the dollars per kilogram calculations are displayed in Table 8-9. The unit costs shown include estimated annual expenditures and are given under four conditions: (1) the unit cost with development allocated over the stated number of launches; (2) the unit costs without development; (3) the unit costs for the ESRL under the assumption that a steel (Fe) nose cone can replace the tungsten (W) nose cone proposed for the Reference Concept projectile with pro-rated development costs; and (4) the unit costs for the ESRL with steel nose cones without development. The costs for the conventional vehicles are repeated in the last two categories. The effect of fully allocating the development cost can be seen to be a relatively minor increase for both alternatives. There is, however, a significant cost

TABLE 8-7. CONVENTIONAL SPACE TRANSPORT MASS AND COST DATA

Space System	Recurring Cost	Development Cost	Mass Delivery Capability,[a] MT		
			Load	Load Factor	Net Payload
Current Shuttle	\$41.7 M (Martin)[b]	~ \$10 B (1981)	25.00	0.75	18.75
Uprated Shuttle	\$36.1 M (Martin)	\$2.473 B (Martin)	41.00	0.75	30.75
Shuttle Derived Vehicle (SDV)	\$48.4 M (Martin)	\$1.5 B (Martin)	74.00	0.75	55.50
Nuclear Waste Disposal Mission OTV and SOIS	\$10 M (Estimate)	\$2.0 B (Estimate)	--	--	6.3

(a)To 500 km circular orbit for Earth Orbit Mission; to 0.85 A.U. Solar Orbit for nuclear waste disposal in space.

(b)1981 current Shuttle cost estimated by Battelle at \$58 M in 1981 dollars. Information provided by Martin-Marietta is in 1980 dollars and was provided by Frank Williams of MMC and Mike Van Hook of MSFC. The information is part of the work, but not reported in, the Phase II Final Review, "Technology Requirements Study, Shuttle Derived Vehicles", April 26, 1982.

TABLE 8-8. ESRL-STS EQUIVALENCE MODEL, EQUAL MASS FOR 30 YEAR TRAFFIC MODEL (2020-2050)

Space Transport System	NUCLEAR WASTE DISPOSAL(a)			EARTH ORBITAL APPLICATIONS(b)		
	Mass/Launch, MT	Total Launches	Launch Rate	Mass/Launch, MT	Total Launches	Launch Rate
ESRL	0.250	21,900	2/day	0.650	87,600	8/day
Current Shuttle	--	--	--	18.75	3,037	101/year
Uprated Shuttle	3.15	1,738	58/year	30.75	1,852	62/year
Shuttle Derived Vehicle (SDV)				55.5	1,025	34/year

(a) Total cermet HLW payload mass of 5,475 MT.
(b) Total bulk materials mass of 56,940 MT to 500 km, 0° inclination orbit to support space station and space operations.

TABLE 8-9. COMPARATIVE TRANSPORTATION COSTS ($, M, 1981)

Space Transport System	Development Cost	Unit Cost with[(a)] Development	Unit Cost[(a)] No Development	Unit Cost[(a)] Fe Replaces W With Development	Unit Cost[(a)] Fe Replaces W No Development
ESRL Low	3,734				
Mission A (2/day)		0.085	0.051	0.062	0.027
Mission B (8 day)		0.130	0.095	0.100	0.066
ESRL High	7,700				
Mission A (2/day)		0.425	0.354	0.360	0.289
Mission B (8/day)		0.879	0.808	0.649	0.578
ESRL Espected	5,403				
Mission A (2/day)		0.139	0.089	0.099	0.050
Mission B (8/day)		0.383	0.344	0.213	0.164
Current Shuttle (3037 Launches)	10,000	61	58	61	58
Uprated Shuttle (2721 Launches)	3,000	51	50	51	50
Shuttle-Derived Vehicle (SDV) (1894 Launches)	2,000	68	67	68	67
Nuclear Waste Stages (1738)	2,000	21	20	21	20
Nuclear Waste Mission (869)	7,000	140	137	140	137

(a)Unit costs include annual expenditures.

reduction if steel can be used in place of tungsten in the ESRL projectile nose cones.

The estimates of Shuttle/STS costs given in Table 8-7 have been adjusted to reflect the substantial inflation since the base date for Shuttle pricing of 1975. The $58 M (1981) estimate reflects not only inflation but also the additional expenses associated in going to a 500 km circular orbit over the lower Shuttle reference orbit (see earlier discussion and Table). The lower estimates for the Uprated Shuttle and the SDV reflect the expected payload increases and costs believed to be achievable (Williams, Van Hook, 1982).

The comparison of the two modes of transport on the basis of transportation costs per kilogram ($/kg, 1981) is made in Table 8-10. The costs per kilogram are presented for both nuclear waste disposal and bulk materials to LEO missions (Missions A and B), with and without amortization of development expenditures, and with tungsten or steel nose cones for the ESRL projectiles. The substitution of steel for tungsten in the nose cones clearly is to be desired from the standpoint of cost savings as it has a much greater impact than the total development expenditures.

It should also be noted that the cost comparisons are made without any consideration of ESRL projectile reuse or the value of the projectile as a source of materials for use in orbital applications. The question of reuse of the Mission B projectile, by returning it on a down trip on the equivalent of the Shuttle Orbiter, should be examined in later system studies. If the projectile can be manufactured in the low end of the cost range, only the rocket engine may have sufficient value to justify reuse, and the rest of the projectile would have value only as scrap. If the projectile can only be manufactured at the high end of the cost range, reuse of the projectile or major parts thereof would be highly advantageous. At the high end of the Mission B projectile's cost range, the propulsion system represents over 80 percent of the cost of transportation. Even with the additional expenses associated with reuse, a reduction of 50 percent in the dollars per kilogram estimate would be reasonable. However, unless it appears that an inexpensive projectile would also be easily reusable, the dollars per kilogram calculations of Table 8-10 suggest that the goal might be a low-unit-cost projectile which is used once.

From the data in Table 8-10, the ESRL nuclear waste disposal in space mission appears a clear winner over disposal by conventional space transportation. The factor of advantage to ESRL would range from 11 to 78 with an expected value of 35 if a tungsten nose cone is required and 48 if a steel nose cone were used.

For bulk materials to LEO missions, ESRL produces lower unit transport costs than either the current Space Shuttle or the Uprated Space Shuttle in all cases. If the highest ESRL costs were realized, the SDV, however, would yield a 30 percent cost per kilogram advantage, and then only if the projectile could not be reused. The expected ESRL advantage in transport costs over the SDV is 1.5 for a tungsten nose cone and 2.8 if a steel nose cone can be used. The expected advantage to the ESRL in comparison with the current and Uprated Shuttles are 5 and 3, respectively.

TABLE 8-10. ESRL-STS TRAFFIC MODEL COST COMPARISON ($/kg, 1981--Transport Only)

Space Transport System	Number of Launches	With Development		No Development		With Development and Fe Replaces W		No Development and and Fe Replaces W	
		Mission A	Mission B	Mission A	Mission B	Mission A	Mission B	Mission A	Mission B
ESRL-Low	109,500	340	200	200	150	250	150	110	100
ESRL-High	109,500	1,700	1,350	1,420	1,240	1,440	1,000	1,160	890
ESRL-Expected	109,500	560	590	360	510	400	330	200	250
Current Shuttle	3,037	--	3,250	--	3,100	--	3,250	--	3,100
Uprated Shuttle	2,721	--	1,660	--	1,630	--	1,660	--	1,630
Shuttle Derived Vehicle (SDV)	1,894	--	1,225	--	1,200	--	1,225	--	1,200
Nuclear[(a)] Waste Mission	869 x 2	22,200	--	21,700	--	22,200	--	21,700	--

(a)One nuclear waste mission uses Uprated Shuttle, SDV, OTV, and SOIS.

The expected values of the costs per kilogram if Missions A and B are considered to have totally separate ESRL facilities are presented in Table 8-11. Also given for comparison are the costs for the Reference ESRL Concept (Missions A and B carried out by an integrated ESRL system using same facilities but two launcher tubes). The big shift occurs in Mission A where the advantage is reduced to a factor of 19 from 35. This is due to the low launch rate and indicates that it would be desirable to design the launcher and power station to increase the launch rate above two per day to be able to dispose of nuclear waste from additional sources (defense waste, TRU waste, foreign wastes, etc.).

8.3.3 Effects of Mass Delivery Rates on Costs

The previous calculations assume fixed and high launch rates of two per day for Mission A and eight per day for Mission B. Figure 8-7 is a graph of expected transportation costs in dollars per kilogram versus Mission B payload mass launched per day for the ESRL and the three STS systems. All data assume that the development expenditures are amortized over 30 years, except for the current Shuttle for which development expenditures represent a sunk cost. The ESRL is shown twice--with tungsten and with steel nose cones on the projectiles. The ESRL configurations and data assume that both Mission A and B launchers are built and that two Mission A launches are flown each day with 20 percent of development and annual expenses charged to Mission A and 80 percent charged to Mission B. The ESRL produces lower expected costs over the entire range of mass delivery considered, including the very low range corresponding to one or two conventional vehicle launches per year which is equivalent to one ESRL Mission B launch every other day. The near intersection in this lower range, however, suggests that the ESRL offers no great advantage at low delivery rates.

The worst case for the ESRL has also been examined. This consists of building both Mission A and B launchers and realizing the high estimates for all costs, including those for the projectiles, and then using only the Mission B launcher on the average of once per day. This yields an estimated transportation cost of $2600 per kilogram versus $3100 for the current Shuttle and $1660 and $920 for the Uprated Shuttle and SDV, respectively. Thus, while disposing of nuclear waste in space may be performed very cost-effectively by the ESRL, delivery of only bulk material to LEO does not appear to be nearly as advantageous. Based on the relative closeness of advanced Shuttle designs and the ESRL in terms of dollars per kilogram, it is possible that conventional rocket launchers could outperform the ESRL in bulk delivery. It is very likely, however, that considerable additional development expenditures, significantly above the levels indicated here for STS modifications, would be required to achieve rocket launch costs below those indicated for the ESRL. Also, if comparisons ($/kg) were made for possible traffic to higher altitudes (e.g., GEO), the ESRL system would show a greater economic advantage--the conventional cost per kilogram would grow because of inclusion of an OTV and reduction in payload. The ESRL cost would likely stay the same or be reduced due to decreased on-orbit propulsion, and increased payload (the current ESRL Mission B launcher has been conceptualized with excess capability).

TABLE 8-11. EXPECTED SPACE TRANSPORT COST PER KILOGRAM SUMMARY (1981 $)

Space Transport System	Development Cost, $M	Nuclear Waste Only, $/kg	Mission B Only, $/kg	Mission A and B $/kg
ESRL				
- Mission A and B	5,403	--	--	560/400(A)[a] 590/330(B)[a]
- Mission A only	3,175	1,090/930[a]	--	--
- Mission B only	3,320	--	570/310[a]	--
Current STS	Paid	--	3,100	3,100 (B)
Uprated Shuttle	3,000		1,660	1,660 (B)
Shuttle Derived Vehicle (SDV)	2,000	22,200	1,225	1,225 (B)
Upper Stages (2)	3,000		--	22,200 (A)

[a] Tungsten nose cone/steel nose cone.

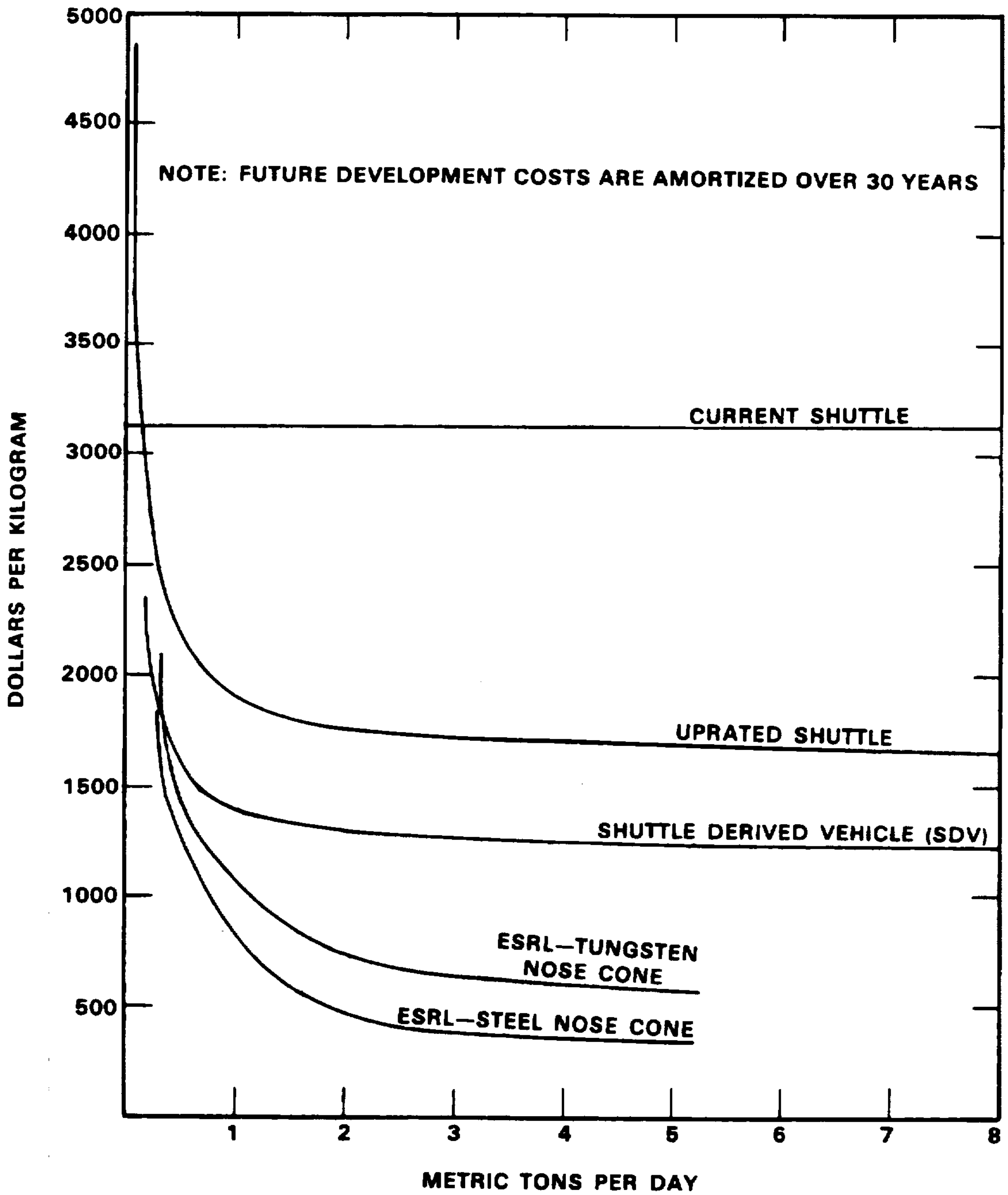


FIGURE 8-7. EXPECTED COMPARATIVE TRANSPORTATION COSTS IN DOLLARS PER KILOGRAM VERSUS MISSION B PAYLOAD IN METRIC TONS LAUNCHED PER DAY

8.3.4 Discount Analysis for Space Transport Costs

The concept of time value of money can shift perspectives of the value of projects and is discussed here in the context of how application of the concept might change or modify the economic assessment of the ESRL.

Briefly, the time value of money concept is based on the fact that a dollar in hand at the present time is worth more than a dollar received or spent in the future and the difference in value is the interest rate which can be obtained from an alternative investment of that dollar. In Circular A-94, the Office of Management and Budget recommends an interest rate of 10 percent for the purpose of evaluating federal programs. Within the overall concept of the time value of money, there are several methods of analysis, most of which require knowing the dollar value of the benefits being pursued. For the ESRL, however, only the space transport costs are estimated, so the analysis must be constrained to the technique of comparing discounted costs. In the discounted cost method, the alternative with the lowest discounted cost is the most advantageous. Computationally, this requires determining the expected annual costs for the alternatives and adjusting or discounting them for the time value of the money; this adjustment is a factor: $1/(1+i)^n = 1/(1.1)^n$ where i is the interest rate (10 percent) and n is the number of years in the future.

Under some circumstances the application of this technique can change the conclusion that a specific alternative has the lowest cost. This typically occurs when one of the higher-cost alternatives requires spending undiscounted dollars later in time than the lower-undiscounted-cost alternative. For the nuclear waste disposal mission (Mission A), the time-value-of money concept does not change the conclusion that the ESRL has a lower cost in comparison to conventional space disposal. Not only would the operating costs for the ESRL be much lower than for conventional space transport, much of the ESRL development expenditures would likely occur later than the development of the conventional space transport capability. This delay in ESRL development expenditures would increase the attractiveness of the ESRL from the standpoint of the time value of money.

Information is not available, however, to make a detailed comparison of the ESRL and mined geologic repositories, so no conclusions about discounted cost advantages can be made, at present, for this comparison.

For the transport of bulk materials to low-Earth Orbit, the discounted cost advantage does not clearly go to the ESRL, even though the annual expenditures for the ESRL are significantly less than for conventional space transport. Here, the alternatives are a lot closer when compared in discounted costs. For purposes of making the analysis, four alternative scenarios are compared. In all but one scenario, an Uprated Shuttle is developed and used from 1992 onward.

The basic alternatives are illustrated by time lines given in Table 8-12.

TABLE 8-12. IOC AND OTHER DATES FOR ALTERNATIVE SCENARIOS FOR BULK TRANSPORT TO LEO (MISSION B)

Space Transport System/ Activity	Current Shuttle Plus ESRL	Conventional[a] Space Transport Only	Uprated Shuttle with ESRL	Conventional[a] Space Transport Plus ESRL
Current Shuttle	1982	1982	1982	1982
Uprated Shuttle	-	1992	1992	1992
Shuttle Derivitive Vehicle (Cargo Only)	-	1995	-	1995
Initiate ESRL Construction	2010	-	2010	2010
ESRL IOC	2019	-	2019	2019
Start Bulk Transport Scenario	2021	2021	2021	2021
End Bulk Transport Scenario	2050	2050	2050	2050

(a) Uprated Shuttle and Shuttle derived cargo vehicle.

In all cases, the Shuttle or Uprated Shuttle are used until about 1995 for all purposes. In the conventional space transport only scenario, in addition to the Uprated Shuttle a Shuttle derived vehicle (SDV) is developed for transporting both large objects and bulk materials, and the recurring costs for bulk transport are considered for the years 2021-2050 to provide an appropriate comparison with other alternatives. In the Uprated Shuttle plus ESRL scenario, the Uprated Shuttle is considered sufficient for transport to large objects to LEO, and the ESRL is used for bulk transport. In the Conventional Space Transport plus ESRL scenario, the SDV is developed to carry large objects and the ESRL is developed to carry bulk materials. In all cases, the recurring costs for bulk transport are considered only for the anticipated economic life of the ESRL facility--from 2021-2050.

TABLE 8-13. ESTIMATED COSTS AND DISCOUNTED COSTS FOR ALTERNATIVE BULK TRANSPORT (MISSION) SCENARIOS

	Current Shuttle plus ESRL (Tungsten Cone)	Current Shuttle plus ESRL (Steel Cone)	Conventional Space Transport (URS+SDV)	Uprated Shuttle plus ESRL (W)(a)	Uprated Shuttle plus ESRL(Fe)(a)	Conventional Space Transport (URS+SDV) plus RSRL(W)(a) for Bulk	Conventional Space Transport (URS+SDV) plus ESRL(Fe)(a) for Bulk
No. of Bulk Transport Launches	75,190	75,190	881	75,190	75,190	75,190	75,190
Estimated Costs($,M,1981)							
Development	5,400	5,400	5,000	8,430	8,430	10,430	10,430
Recurring	25,260	12,478	59,000	25,260	12,478	25,260	25,260
Total	30,660	17,878	64,000	33,690	20,908	35,690	22,908
Discounted Total Cost(10%)	448	348	2,867	1,922	1,822	2,854	2,754

(a) W = Tungsten Nose Cone; Fe = Iron nose cone.

Since all systems are anticipated to be developed and used for other purposes in addition to bulk transport, full development costs as well as the recurring costs for bulk transport are considered in the costing of these scenarios to avoid problems of allocating development costs to other uses. The results of the computations are given in Table 8-13 with estimated actual costs, discounted costs, and numbers of flights costed. This assessment uses the mass transport model of Table 8-6. The detailed calculations are presented in Tables 8-14 and 8-15. The relative cost advantage of the ESRL is not lost by discounting, even when it is assumed that an additional vehicle (the SDV) is required to transport very large objects and that this vehicle could also transport bulk materials in competition with the ESRL. This case has discounted costs which are very close to those of the conventional space transport only scenario. If the SDV were to be developed later than assumed (e.g. 2005 rather than 1995), the effect would be to eliminate the ESRL's advantage if tungsten nose cones are required. If steel nose cones can be used, the ESRL would still retain its discounted cost advantage. The case of the current Shuttle only, with the ESRL for bulk transport, is chiefly presented to show the effects of discounting. While the cost comparison is highly advantageous, it is considered unlikely that advances in conventional space transport will be delayed to await the ESRL.

TABLE 8-14. ESTIMATED STS AND ESRL DEVELOPMENT COST STREAMS

	Discount Factor	Conventional Space Transport		ESRL Development (A+B)	Discounted Values (10%)
		Uprated Shuttle	Shuttle Derived Vehicle		
1982	1.000				
1983	0.9091				
1984	0.8264				
1985	0.7513				
1986	0.6830				
1987	0.6209				
1988	0.5645	500			282.236
1989	0.5132	1,000			513.158
1990	0.4665	1,000			466.507
1991	0.4241	500	500		424.097
1992	0.3855	(3,000)	1,000		385.554
1993	0.3505		500		175.246
1994	0.3186		500		159.315
1995	0.2896		(2,000)		(URS+SDV)(2406.113)
1996	0.2633				
1997	0.2394				
1998	0.2176				
1999	0.1978				
2000	0.1798				
2001	0.1635				
2002	0.1486				
2003	0.1351				
2004	0.1228				
2005	0.1117			10	1.117
2006	0.1015			20	2.031
2007	0.0923			20	1.846
2008	0.0839			50	4.195
2009	0.0763			100	7.628
2010	0.0693			(Initiate 100 Construction)	6.934
2011	0.0630			200	12.607
2012	0.0573			400	22.923
2013	0.0521			700	36.469
2014	0.0474			1,000	47.362
2015	0.0431			1,000	43.056
2016	0.0391			700	27.399
2017	0.0355			500	17.792
2018	0.0323			400	12.940
2019	0.0294			200	5.882
		(5,000)		(5,400)	(ESRL)(250.181)

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TABLE 8-15. ESTIMATED STS AND ESRL OPERATIONS COST STREAMS FOR MISSION B

Year	Discount Factor	No. of SDV Launches	SDV Ops Cost ($,M,81)	Discounted SDV Cost ($,M,81,10%)	ESRL B Launches/Day	ESRL B (Fe Nose Cone) Ops Cost ($,M,81)	Discounted ESRL B (Fe Nose Cone) ($,M,81,10%)	ESRL B (W Nose Cone) Ops Cost ($,M,81)	Discounted ESRL B (W Nose Cone) Ops Cost ($,M,81,10%)
2020	0.02673	-	-	-	-	-	-	-	-
2021	0.02430	24.8	$1661.6	40.384	5.8	358.3	8.708	718.2	17.455
2022	0.02209	24.8	1661.6	36.713	5.8	358.3	7.916	718.2	15.868
2023	0.02009	24.8	1661.6	33.375	5.8	358.3	7.197	718.2	14.426
2024	0.01826	25.7	1718.6	31.382	6.0	369.1	6.740	741.4	13.538
2025	0.01660	25.7	1718.6	28.529	6.0	369.1	6.127	741.4	12.307
2026	0.01509	25.7	1718.6	25.936	6.0	369.1	5.570	741.4	11.188
2027	0.01372	27.0	1805.7	24.772	6.3	385.3	5.286	776.2	10.649
2028	0.01247	27.0	1805.7	22.520	6.3	385.3	4.805	776.2	9.681
2029	0.01139	27.0	1805.7	20.473	6.3	385.3	4.369	776.2	8.800
2030	0.01031	28.2	1890.1	19.482	6.6	401.5	4.138	811.0	8.359
2031	0.00937	28.2	1890.1	17.711	6.6	401.5	3.762	811.0	7.600
2032	0.00852	28.2	1890.1	16.101	6.6	401.5	3.420	811.0	6.909
2033	0.00774	29.5	1976.5	15.306	6.9	417.7	3.235	845.9	6.550
2034	0.00704	29.5	1976.5	13.915	6.9	417.7	2.941	845.9	5.955
2035	0.00640	29.5	1976.5	12.650	6.9	417.7	2.673	845.9	5.414
2036	0.00582	29.5	1976.5	11.500	6.9	417.7	2.430	845.9	4.922
2037	0.00529	30.4	2033.5	10.756	7.1	428.5	2.266	869.1	4.597
2038	0.00481	30.4	2033.5	9.778	7.1	428.5	2.060	869.1	4.179
2039	0.00437	30.4	2033.5	8.889	7.1	428.5	1.873	869.1	3.799
2040	0.00397	30.4	2033.5	8.081	7.1	428.5	1.703	869.1	3.454
2041	0.00361	31.6	2119.2	7.656	7.4	444.7	1.607	903.9	3.265
2042	0.00328	31.6	2119.2	6.960	7.4	444.7	1.461	903.9	2.967
2043	0.00398	31.6	2119.2	6.327	7.4	444.7	1.328	903.9	2.699
2044	0.00371	31.6	2119.2	5.752	7.4	444.7	1.207	903.9	2.453
2045	0.00246	32.5	2176.8	5.371	7.6	455.5	1.124	927.1	2.287
2046	0.00224	32.9	2205.6	4.948	7.7	461.0	1.034	938.7	2.106
2047	0.00203	32.9	2205.6	4.498	7.7	461.0	0.940	938.7	1.914
2048	0.00185	32.9	2205.6	4.089	7.7	461.0	0.855	938.7	1.740
2049	0.00169	32.9	2205.6	3.717	7.7	461.0	0.777	938.7	1.582
2050	0.00153	33.8	2262.6	3.466	7.9	471.8	0.723	962.0	1.474
		881.0	59,027.0	461.037	(75,190 over 30 years)	12,477.5	98.275	25,759.9	198.137

APPENDIX A

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REFERENCES

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APPENDIX B

ACRONYMS AND ABBREVIATIONS

APPENDIX B
ACRONYMS AND ABBREVIATIONS

ACS	attitude control system
A.U.	astronomical unit
ALARA	as low as reasonably achievable
AMZIRC	copper-zirconium alloy
ANU	Australian National University
BCL	Battelle's Columbus Laboratories, Columbus, Ohio
C	degrees centigrade
cc	cubic centimeters (cm^3)
C_D	drag coefficient
CFR	Code of Federal Regulations
Ci	Curies
cm	centimeters
COR	Contracting Officer's Representative
DES	distributed energy store
DOE	U.S. Department of Energy
DOT	U.S. Department of Transportation
EPA	U.S. Environmental Protection Agency
ERDA	U.S. Energy Research and Development Administration
ESRL	Earth-to-Space Rail Launcher
FY	fiscal year
g	grams
GEO	geosynchronous orbit
GW_e	gigawatts electric
μH	micro-Henrys
HLW	high-level waste
HPG	homopolar generator
IAP	International Applied Physics, Dayton, Ohio
I_{sp}	specific impulse
JSC	NASA's Johnson Space Center, Houston
k	thermal conductivity
kg	kilogram
km	kilometer
KSC	Kennedy Space Center, Florida
kW	kilowatt
LASL	Los Alamos Scientifc Laboratory, Los Alamos, New Mexico
LEO	low Earth orbit
LeRC	NASA's Lewis Research Center, Cleveland, Ohio
LH_2	liquid hydrogen
LLNL	Lawrence Livermore National Laboratory, Livermore, California
LN_2	liquid nitrogen
LOX	liquid oxygen
LRB	Liquid Rocket Booster (Uprated Shuttle)
LSS	large space structure
m	meters
MA	mega-amp

MGR	mined geologic repository
MHD	magnetohydrodynamics
MJ	megajoule
mrem	millirem
m/s	meters per second
MT	metric tons
MTHM	metric tons of heavy metal
MSFC	NASA's Marshall Space Flight Center, Huntsville, Alabama
MW_e	Megawatt electric
N	Newtons
N/cm^2	Newtons per square centimeter
n. mi.	nautical mile
NASA	National Aeronautics and Space Administration
NASTRAN	NASA Structural Analysis computer code
NORAD	North American Aerospace Defense Command
NRC	Nuclear Regulatory Commission
NWDS	nuclear waste disposal in space
O/F	oxidizer to fuel ratio
ONI	Office of NTWS Program Integration (DOE's)
ONWI	Office of Nuclear Waste Isolation (DOE's)
ORNL	Oak Ridge National Laboratories
OSU	The Ohio State University, Columbus, Ohio
OTV	Orbit Transfer Vehicle
PAI	Physics Applications, Inc., Dayton, Ohio
PL	payload
PNL	Pacific Northwest Laboratory, Richland, Washington
PSS	payload support structure
QAD	radiation shielding computer code
R	rads
rem	roentgen equivalent, man
RETAC	BCL Reentry Thermal Analysis Code
RSS	Rotating Service Structure (Shuttle)
SDV	Shuttle Derived Vehicle
SES	single energy store
SL	sea level
SOA	state-of-the-art
SOIS	Solar Orbit Insertion Stage
SPS	Solar Power Satellite
SRB	Solid Rocket Booster
SR&T	supporting research and technology
SSME	Space Shuttle Main Engine
STS	Space Transportation System
ΔT	change in temperature
TBD	to be determined
TPS	thermal protection system
TRU	transuranic waste
UT	The University of Texas, Austin, Texas
UT-CEM	The University of Texas Center for Electromechanics, Austin, Texas
Δv	change in velocity
W	watt
WBS	work breakdown structure

APPENDIX C

METRIC TO ENGLISH CONVERSION FACTORS

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 ft (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). . . .	cubic feet (ft^3).	35.32
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

*NOTE: Multiply by 1.8 and then add 32.

To convert	into	multiply by
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

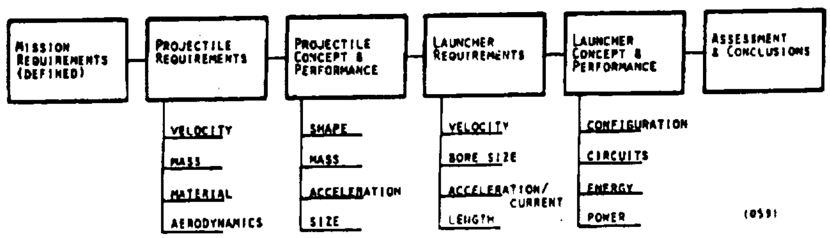
APPENDIX D

CONCEPT DEFINITION MEETING
(Selected Vugraphs by J. Barber, IAP, Research, Inc.)

Held at Battelle Columbus Laboratories
on
12-13 August 1981

RAIL GUN LAUNCHER SYSTEM STUDY

- CAN A RAIL GUN LAUNCHER BE BUILT WHICH COULD LAUNCH OBJECTS FROM THE SURFACE OF THE EARTH INTO EARTH AND/OR SOLAR SYSTEM ESCAPE ORBITS?
- APPROACH

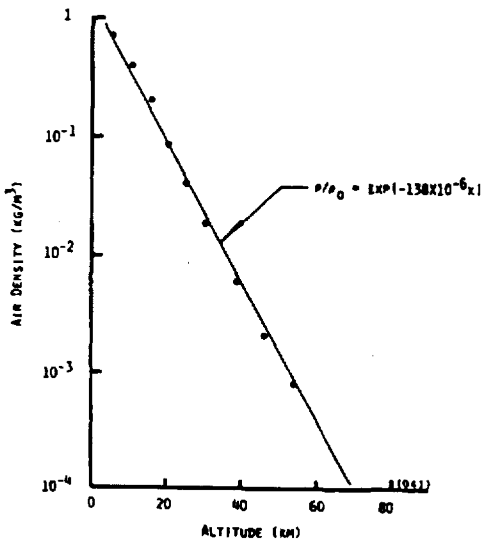


MISSION REQUIREMENTS

- VELOCITY -- 5 KM/S - 25 KM/S
  -- EMPHASIS ON SOLAR SYSTEM ESCAPE AT 16.7 KM/S (EXOATMOSPHERIC)
- PROJECTILE ACCELERATION $\leq$ 300,000 M/S^2
- PROJECTILE MATERIAL -- CERMIT NUCLEAR WASTE/STAINLESS STEEL SHIELDING
- AERODYNAMICS -- DRAG (MUZZLE VELOCITY)
  -- ABLATION (SURVIVAL
- LAUNCH DIRECTION -- VERTICAL
- LAUNCH CAPACITY -- 1MT/DAY OF CERMIT
- SAFETY -- PROJECTILE DECELERATION
  -- PROJECTILE REENTRY

PROJECTILE VELOCITY/AERODYNAMICS

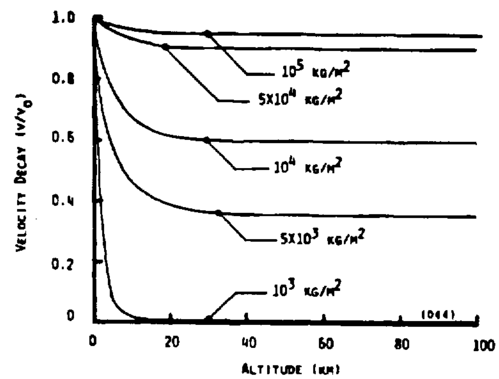
THE ATMOSPHERE



- ρ = 1.392 EXP $(0.138\text{X}10^{-3}\text{X})$

AERODYNAMIC DRAG

- $F_D = -\rho V^2 C_D A/2$
- $A_D = -\rho V^2 C_D A/2M$
- $\beta = M/C_D A$
- $\rho = \rho(X)$
- $\text{LN}(V/V_0) = (-\rho_0/2\beta K)(1-E^{-KX})$

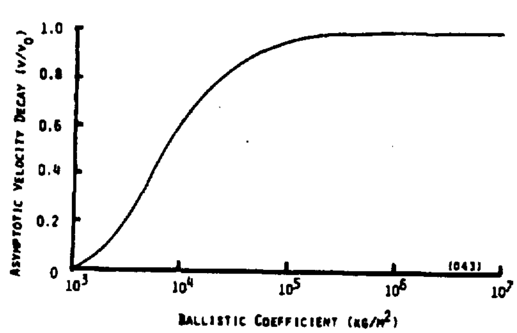


- THE ATMOSPHERE ≃ 7.41 KM @ SEA LEVEL
- 0-20 KM MOST IMPORTANT

PROJECTILE VELOCITY/AERODYNAMICS (2)

ASYMPTOTIC VELOCITY

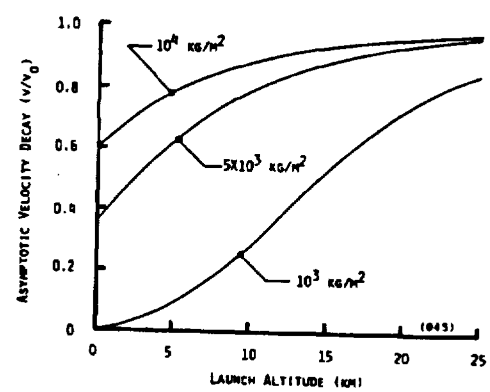
- $X \to \infty$
- $V/V_0 \to \text{EXP}(-\rho_0/2\beta K)$



- $\beta = 10^5$ KG/M^2
- $V/V_0 = 0.955$

LAUNCH ALTITUDE

- $\rho_0 = \rho_0(X_0)$



- MT. MCKINLEY 6020 M
- MAUNA LOA 4170 M

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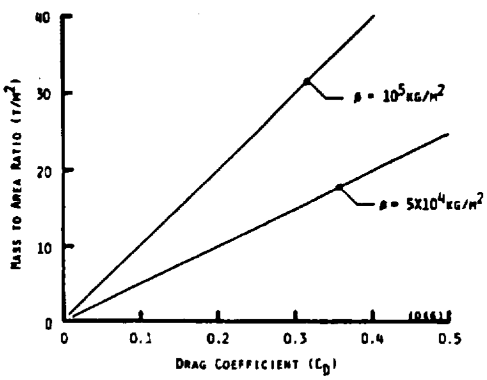
PROJECTILE CONSIDERATIONS (1)

BALLISTIC COEFFICIENT

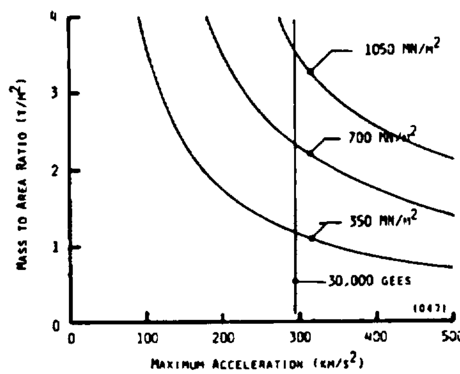
- $\beta = M/C_DA = 10^5$ KG/M^2



- WHAT CONTROLS M/A?
  --ACCELERATION STRESS
- WHAT CONTROLS C_D?
  --SHAPE

MASS TO AREA RATIO

- BASE PRESSURE LAUNCHED
  $M/A = \sigma/A$

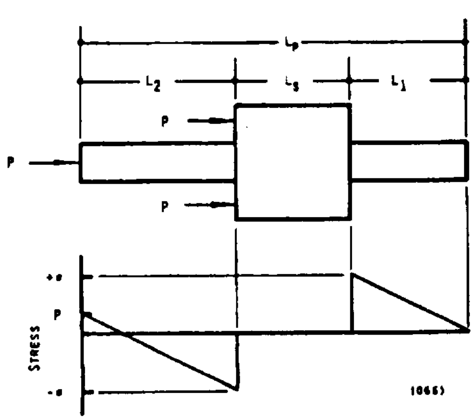


- HIGH STRESS = 700 MN/M^2
- HIGH MASS TO AREA RATIO > 2300 KG/M^2
- DRAG COEFFICIENT < 0.025

PROJECTILE CONSIDERATIONS (2)

CAN SABOTTING HELP?

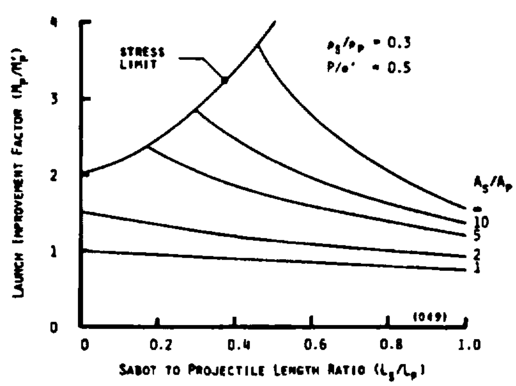
- CUP OR PUSHER SABOT - NO
- RING SABOT - YES - MAYBE



- FIGURE OF MERIT

$$\gamma = \frac{P}{\sigma}\left(\frac{1 + A_S/A_P}{1 + M_S/M_P}\right)$$

- $P < \sigma$
  $\rho_S < \rho_P$



- MAXIMUM IMPROVEMENT ≈ 2-3
- NEGLECTED
  -SHEAR STRESS
  -SEPARATION
  -ARMATURE
  -PROJECTILE SHAPE

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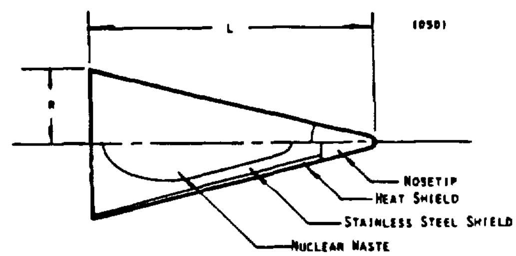
PROJECTILE CONSIDERATIONS (3)

MATERIALS

- PAYLOAD - PW - 4B (PUREX)
  - $-\rho = 6700$ KG/M^3
  - $-\sigma_Y = 345$ MN/M^2 (50KSI)
- SHIELDING - STAINLESS STEEL
  - $-\rho = 8020$ KG/M^3
  - $-\sigma_Y \approx 700$ MN/M^2 (100KSI)
- THERMAL PROTECTION SYSTEM - ?
- REENTRY PROTECTION SYSTEM - ?
- ELECTRONICS
  - $-A \leq 300$ KM/S^2
- ON-BOARD PROPULSION - ?

CONFIGURATION

- RADIATION SHIELDING
  - -STAINLESS STEEL > 90% OF MASS
  - $-\sigma \approx 700$ MN/M^2
  - $-\rho \approx 8000$ KG/M^3
- MASS TO AREA RATIO
  - -2300 KG/M^2 (AT MAXIMUM ACCELERATION)
  - $-C_D < 0.023$
- SLENDER CONE



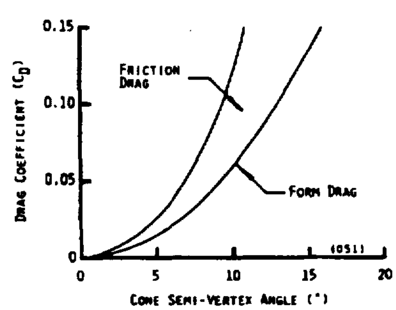
CONICAL PROJECTILE (1)

MASS TO AREA

- MASS $= \rho\pi R^2 L/3$
  - $= \rho\pi L^3/3\tan^2\alpha$
- MASS/AREA - $\rho L/3$
  - --1/3 OF ROD
  - --INDEPENDENT OF α
- ACCELERATION LIMIT
  - $--\rho L/3 \leq 7000\times10^6/A$
- AERODYNAMIC LIMIT
  - $--C_D \leq \rho L/3\times10^5$
- TASK ?
  - --FIND L, C_D AND A WHICH SATISFY LIMITS

DRAG COEFFICIENT

- MACH NO. ≈ 60
- FORM DRAG (CONE)
  - $--C_D \approx 2\sin^2\alpha$
  - --INDEPENDENT OF MACH NO.
  - -- SMALL BLUNTNESS
- FRICTION DRAG ?
- BASE DRAG - NEGLIGIBLE



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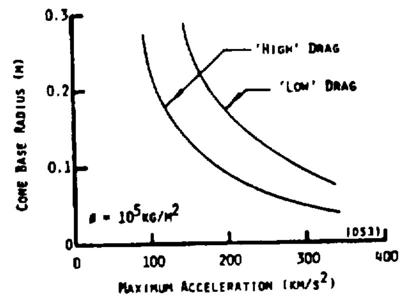
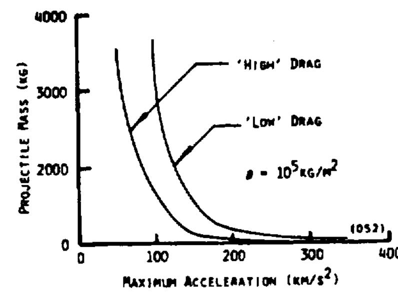
CONICAL PROJECTILE (2)

PROJECTILE SIZE AND MASS

- COMPUTATION
  - --CHOOSE MAXIMUM A
  - --CALCULATE MAXIMUM L (STRESS)
  - --CALCULATE MAXIMUM C_D (BALLISTIC COEFFICIENT)
  - --CALCULATE MAXIMUM α (DRAG COEFFICIENT)
  - --CALCULATE MAXIMUM R (GEOMETRY)
  - --CALCULATE MAXIMUM MASS (GEOMETRY)

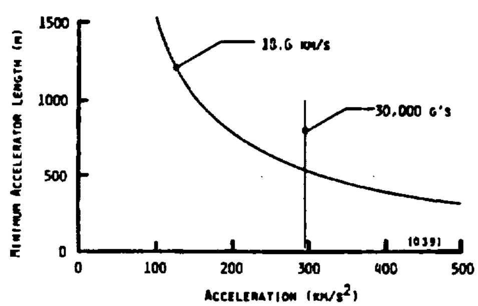


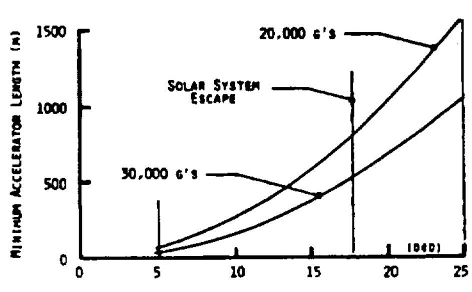


LAUNCHER REQUIREMENTS (1)

LAUNCHER LENGTH CONSTRAINTS

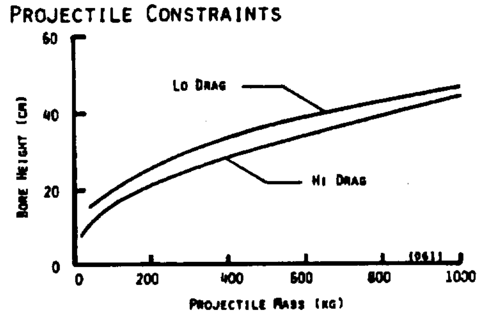
- V_0 = 18.6 KM/S (SOLAR SYSTEM ESCAPE)

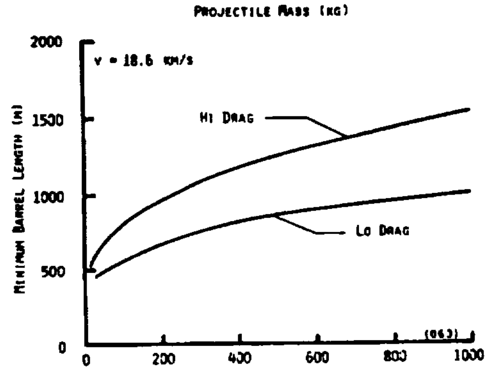




MINIMUM BORE SIZE AND LENGTH

- PROJECTILE CONSTRAINTS





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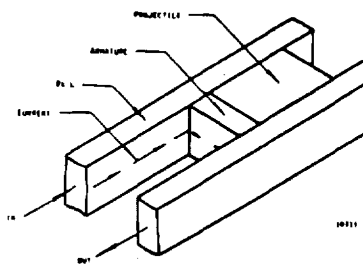
ORIGINAL PAGE IS
OF POOR QUALITY

LAUNCHER REQUIREMENTS (2)

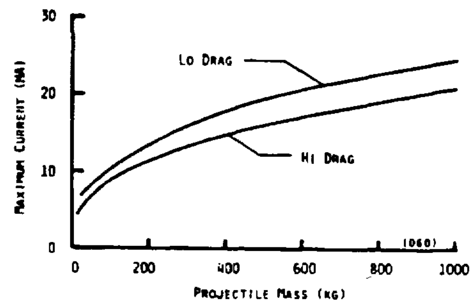
ACCELERATION CURRENT

- RAIL GUN FORCE
  $F = L'I^2/2$
  $I = (2MA/L')^{1/2}$



- SQUARE BORE
  $L' \approx 0.5\ \mu H/M$

- MASS AND ACCELERATION
  PROJECTILE CONSTRAINTS



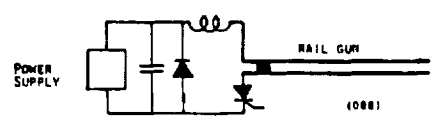
LAUNCHER SYSTEM CONCEPTS

SINGLE OR MULTI STAGE?

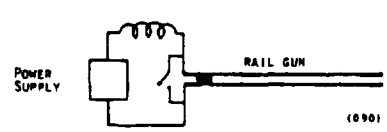
- MULTI-STAGE
  - -- INCREASED EFFICIENCY
  - -- REDUCED ENERGY STORAGE
  - -- REDUCED COST
  - -- INCREASED COMPLEXITY
  - -- MULTIPLE SYSTEMS
  - -- REDUCED RELIABILITY
  - -- INCREASED SIZE
- SINGLE-STAGE
  - -- SIMPLE
  - -- HIGH RELIABILITY
  - -- LOW COST
  - -- LOW EFFICIENCY
  - -- HIGH ENERGY STORAGE
- CHOICE? - TECHNICAL & ECONOMIC

SINGLE STAGE CONCEPTS

- "SINGLE SHOT"



- REPETITIVE



- BOTH USE INDUCTANCE TO CONTROL ACCELERATION.

LAUNCHER SIMULATIONS (1)

OBJECTIVES

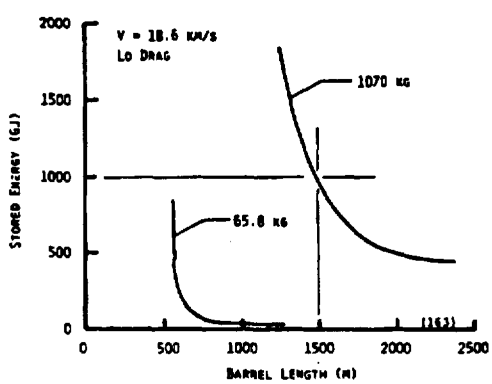
- DETERMINE ENERGY STORAGE REQUIREMENTS
- EVALUATE LOSSES/EFFICIENCY
- EXAMINE FEASIBILITY
  - --ACCELERATOR LENGTH/ENERGY STORAGE
  - --CURRENT/VOLTAGE/POWER
  - --RAIL COOLING

APPROACH

- USE EXISTING SIMULATION PROGRAM
- ASSUMPTIONS
  - --INDUCTIVE DRIVE
  - --$L' = 0.5\ \mu H/M$
  - --COPPER RAILS - ROOM TEMPERATURE RESISTIVITY
  - --SKIN EFFECT IN RAILS
  - --PLASMA ARMATURE

LAUNCHER SIMULATIONS (2)

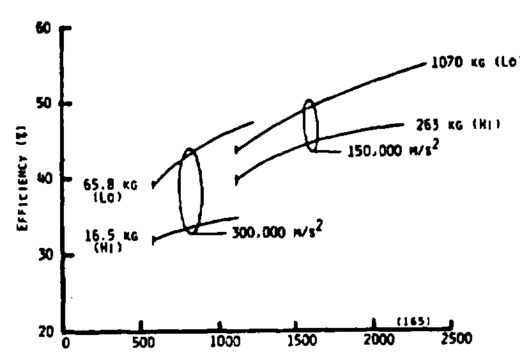
ENERGY STORAGE REQUIREMENTS



- STORED ENERGY REQUIREMENTS STRONGLY DEPENDENT IN GUN LENGTH
- $X \to X_0 \qquad E_0 \to \infty$

CYCLE EFFICIENCY

- η = KINETIC ENERGY/TOTAL ENERGY USED

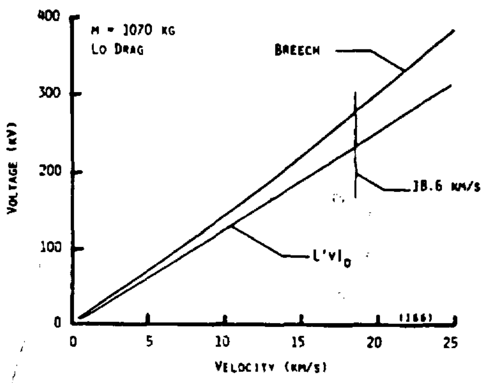


- UNUSED STORED ENERGY MUST BE RECOVERED

LAUNCHER SIMULATIONS (3)

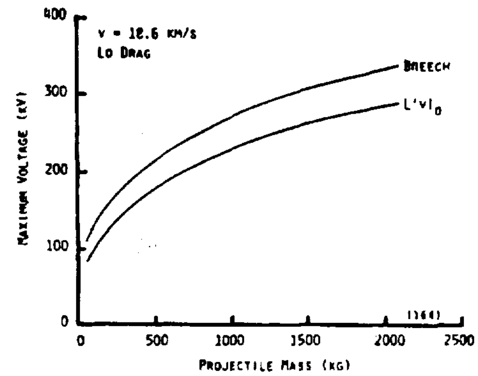
VOLTAGE

- Breech Voltage includes resistive losses
- Back EMF - $L'vI_0$



- Majority of Breech Voltage is back EMF.

- Peak Voltage is dependent on projectile mass.

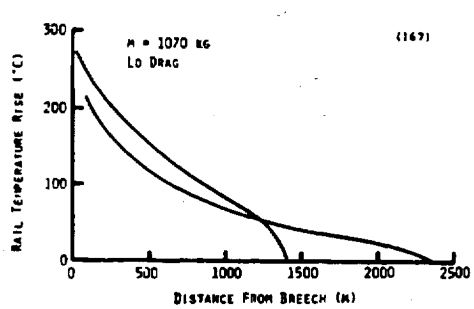


- Voltages are very high (100-300 kV)
- Switching problem for Staged Guns.

LAUNCHER SIMULATIONS (4)

RAIL HEATING

- Resistive Heating of Rails Limits Frequency



- Temperature Rise Highest at Breech
  - --independent of gun length
  - --independent of projectile mass

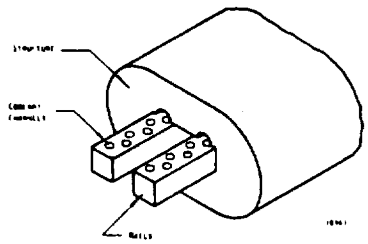
SIMULATION SUMMARY

- Energy Storage Requirements High
  - -- dependent on projectile mass
  - --dependent on gun length
- Acceleration Voltage (power) Very High
  - --not a problem for single stage
- Rail Heating not Severe
  - --cooling should not be difficult
- Conclusions
  - --Nothing Fundamentally Infeasible
  - --System will be Large.

COMPONENTS (1)

BARREL

- COPPER RAILS - WITH COOLING
- FIBER REINFORCED STRUCTURE



- SIZE - PROJECTILE DEPENDENT

BARREL PARAMETER	65.8 KG PROJECTILE	1070 KG PROJECTILE
LENGTH (M)	577 - 1150	1150 - 2300
BORE (M)	0.17	0.48
MASS (MT)	800 - 1700	16,000 - 33,000

(168)

- OPEN LOOP COOLING
  - --12 MT H_2O/SHOT - 65.8 KG
  - --195 MT H_2O/SHOT - 1070 KG
- POINTING
  - --MUST BE FIXED
  - --TOO LARGE, FLEXIBLE TO MOVE
- MOUNTING
  - --ABOVE GROUND - FREE STANDING TOWER
  - --BELOW GROUND - "MINE" SHAFT
  - --EXTERNAL "STIFFENING"

COMPONENTS (2)

INDUCTIVE ENERGY STORE

- TOROID - AL
- REPETITIVE OPERATION (10^{-3}-10^{-1}HZ)
  --REDUCE COIL LOSSES
- RESISTIVE HEATING LIMIT

(169)

COIL PARAMETER	65.8 KG PROJECTILE	1070 KG PROJECTILE
INDUCTANCE (MH)	0.8	1.4
MAJOR RADIUS (M)	10	100
MINOR RADIUS (M)	7	78
MASS (MT)	5000	63,000
ENERGY (GJ)	32	448

- CRYOGENIC AL
  --ENORMOUS REDUCTION IN MASS/SIZE
- CONFIGURATION
  --BROOKS ≈ FACTOR OF 2 REDUCTION IN MASS.

D.C. GENERATOR

- HIGH CURRENT/LOW VOLTAGE - HPG
- TRUNCATED ROTOR DRUM TYPE

HPG PARAMETER	65.8 KG PROJECTILE	1070 KG PROJECTILE
VOLTAGE (V)	87	403
CURRENT (MA)	8.89	25.3
ROTOR RADIUS (M)	0.29	1.34
ROTOR LENGTH (M)	0.78	1.64
MASS (MT)	5.3	317
POWER (MW)	600	7800
ENERGY (GJ)	60/SHOT	780/SHOT

(170)

- ALTERNATIVES
  --MHD ?

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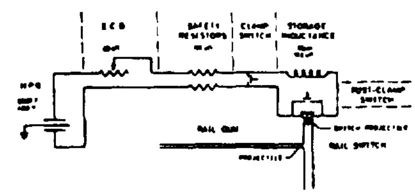
COMPONENTS (3)

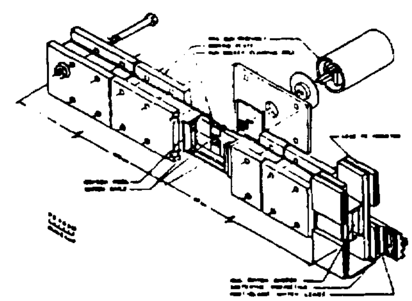
PRIMARY POWER

- ENERGY STORAGE - FLYWHEELS ≈ 20 KJ/KG
  --1500 MT - 22,500 MT/SHOT
- CHEMICAL CONVERSION
  --MHD
  --EXPLOSIVE FLUX COMPRESSION
  --TURBO MACHINERY
- TURBO MACHINERY
  --AIR BREATHING ≈ 0.1 KG/KW
  60 MT - 780 MT
  --ROCKET TURBINES ≈ 0.02 KG/KW
  12 MT - 156 MT
  2.4 M - 7.2 M DIA

SWITCHING

- SWITCHING INTO LOW IMPEDENCE
- A.N.U. RAIL SWITCH CONCEPT





SUMMARY AND CONCLUSIONS

SUMMARY

- PROJECTILE - SLENDER CONE
  --SIZE - ACCELERATION/AERODYNAMICS
- BARREL - SQUARE BORE, COPPER RAILS
  --BORE SIZE - PROJECTILE FITTING (NO SABOT)
  --LENGTH - ACCELERATION/STORED ENERGY
  --CURRENT - ACCELERATION
- ENERGY STORAGE - INDUCTOR
  --BROOKS CONFIGURATION
  --CRYOGENIC AL
- POWER SUPPLY - HPG/TURBINE OR HPG/FLYWHEEL
- SWITCHING - A.N.U. RAIL SWITCH

CONCLUSIONS

- NOTHING DEFINITELY INFEASIBLE
- VERY LARGE SYSTEM
- TECHNOLOGY STRETCHED IN EVERY COMPONENT
- CRITICAL QUESTIONS
  --LARGE PLASMA ARMATURES
  --HIGH PRESSURE ACCELERATION
  --GENERATORS
  --POWER SUPPLIES
  --SIZE
    --BARREL
    --SWITCH
    --COIL
    --GENERATOR
    --TURBINE

APPENDIX E

CONCEPT DEFINITION MEETING
(Selected Vugraphs by R. Hawke, LLNL)

Held at Battelle Columbus Laboratories
on
12-13 August 1981

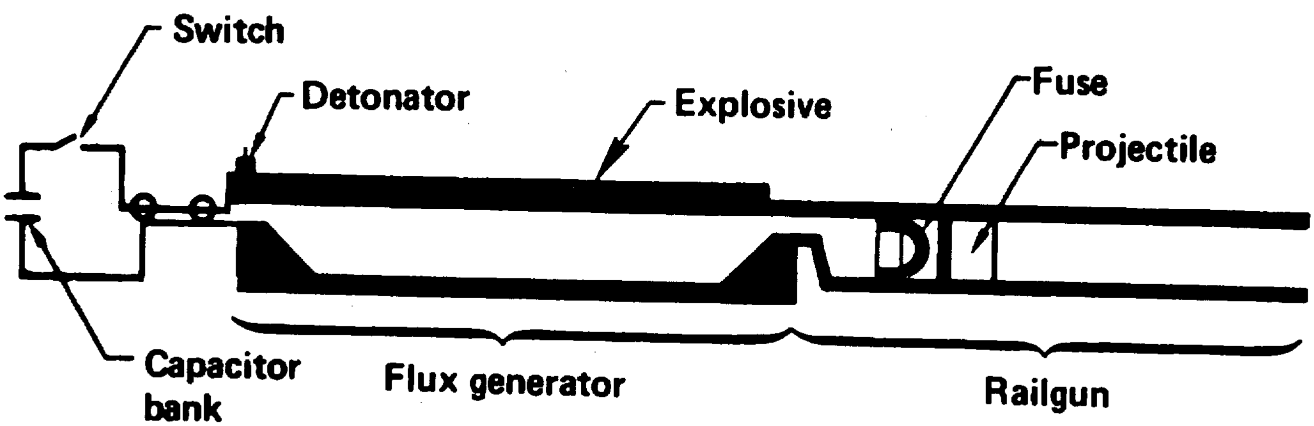
EARTH TO SPACE RAIL LAUNCHER GOALS

V_{MIN}	17.7 KM/S
V_{RANGE}	5-25 KM/S
PAYLOAD	1000 KG/DAY

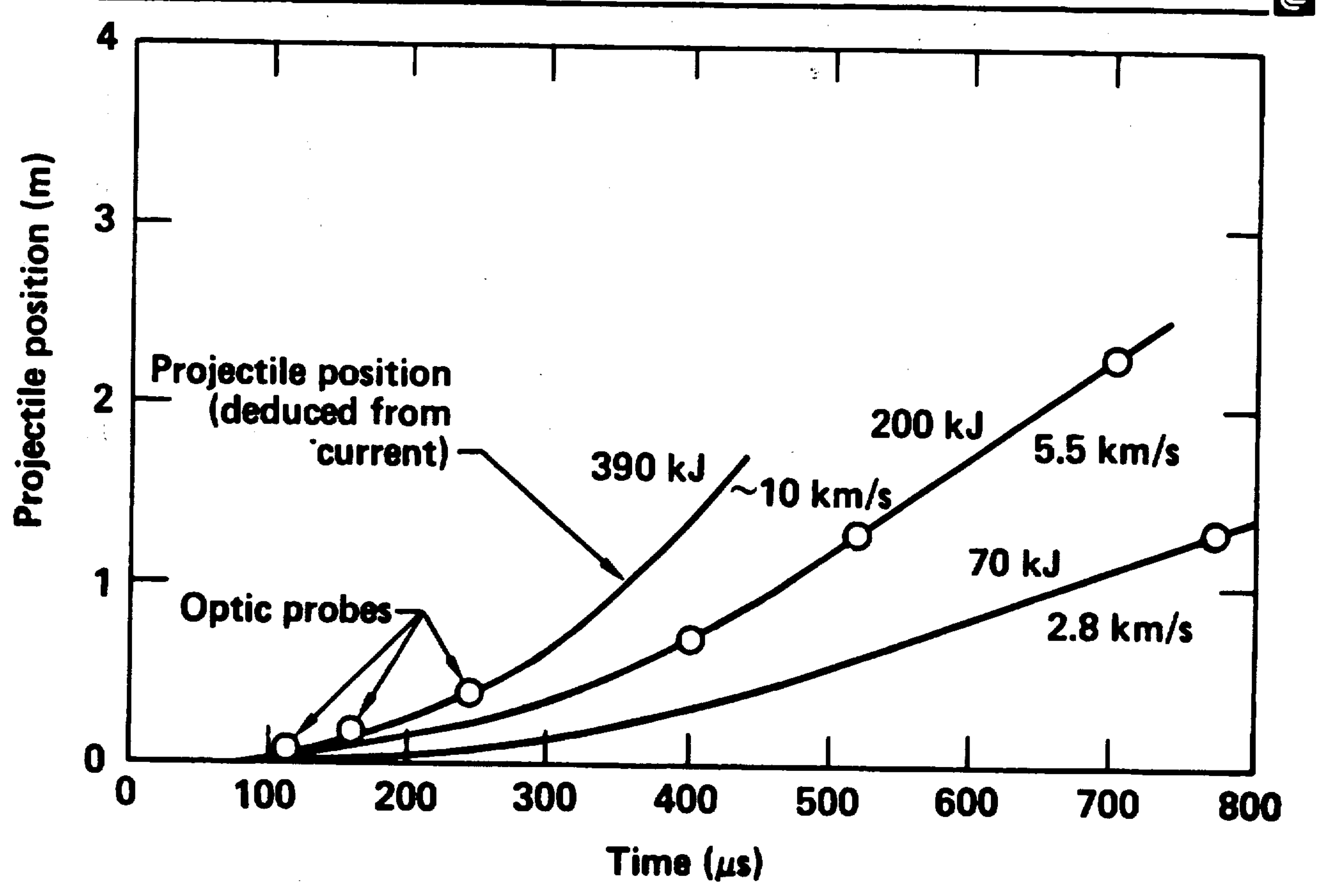
PREVIOUS RAILGUN RESULTS WERE ENCOURAGING

	Prime energy	Mass (g)	Velocity (km/s)	KE (kJ)
GAF (1944)	Battery	10	1.2	7.2
MBA (1963)	MFCG	0.21	9.5	9.5
MBA (1964)	Cap. Bank	0.031	5-6	0.39-0.56
ANU (1978)	HPG-ind.	3	5.9	52

MAGNETIC FLUX COMPRESSION GENERATORS PROVIDED AN IMMEDIATE POWER SOURCE



POSITION VERSUS TIME FOR FCG POWERED RAILGUN EXPERIMENTS



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SUMMARY OF RESULTS

Initial capacitor bank energy (kJ)	70	200	390	390
Bore size (mm)	12.7	12.7	12.7	50
Accelerator length (m)	0.9	1.8	1.8	0.3
Projectile mass (g)	2.9	3.1	3.1	165
Peak current (MA)	0.6	0.8	1.2	[illegible]
Launch velocity (km/s)	2.8	5.5	~10	0.35
Projectile integrity verified	YES	YES	NO	YES

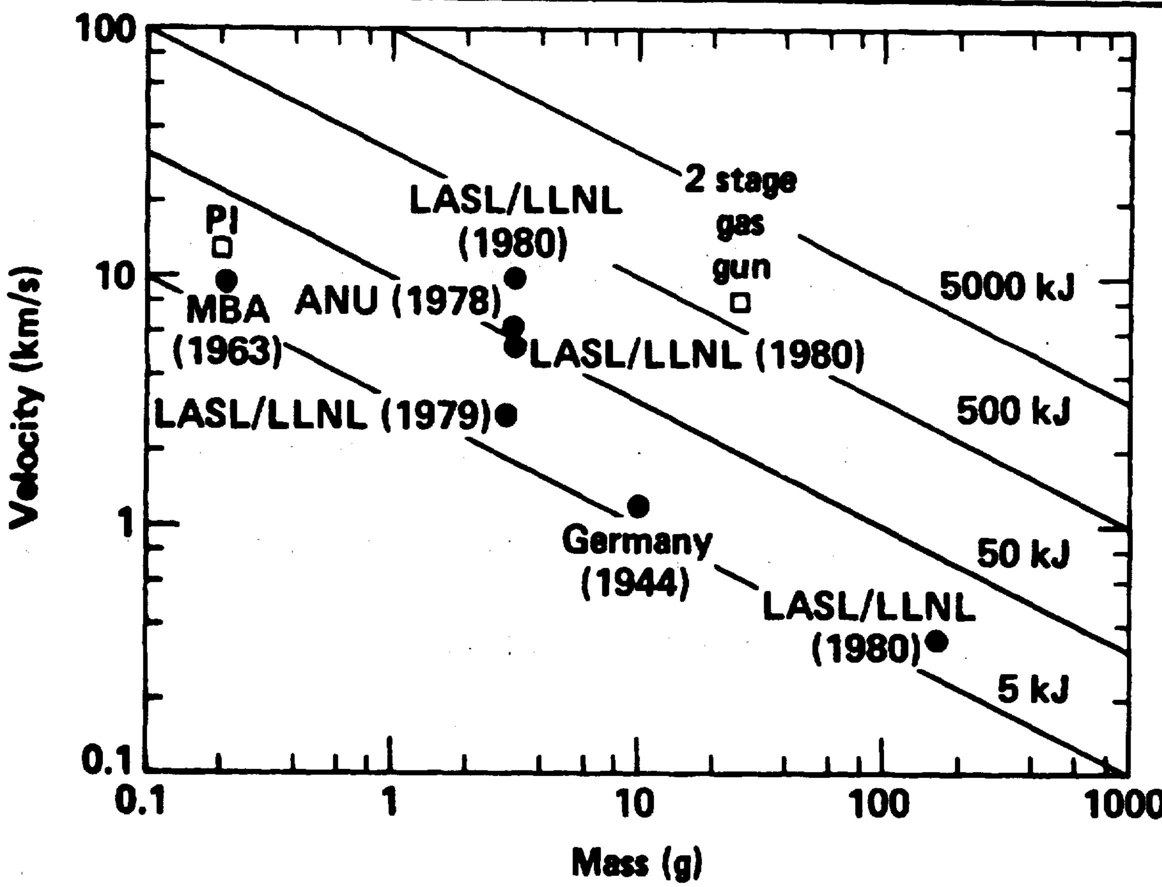
THE 12.7 mm BORE EXPERIMENTS DEMONSTRATED

- Railguns can be operated with megamp currents
- Elastic limit of projectile is not an operational limit
- Rail melting and deformation are limiting factors

THE 50 mm BORE EXPERIMENT DEMONSTRATED

- Plasma armature railguns can be operated without evacuation
- Large plasma arc can be sufficiently stable and uniform to launch large massive projectiles
- Better rail materials are needed

RAILGUNS HAVE BEEN USED TO LAUNCH A VARIETY OF PROJECTILES



EARTH TO SPACE RAIL LAUNCHER GOALS

V_{MIN}	17.7 KM/S
V_{RANGE}	5-25 KM/S
PAYLOAD	1000 KG/DAY

EARTH TO SPACE RAIL LAUNCHER CONSTRAINTS

A MAX $2.9(10^5)$ M/S^2

PAYLOAD SHIELDING 1R AT 1M

HIGH RELIABILITY

3-4 HR. LAUNCH WINDOW AT SUNRISE

VARIABLE LAUNCH DIRECTION

LAUNCH ABORTION CAPABILITY

DESIGN REQUIREMENTS DETERMINE LAUNCHER PARAMETERS

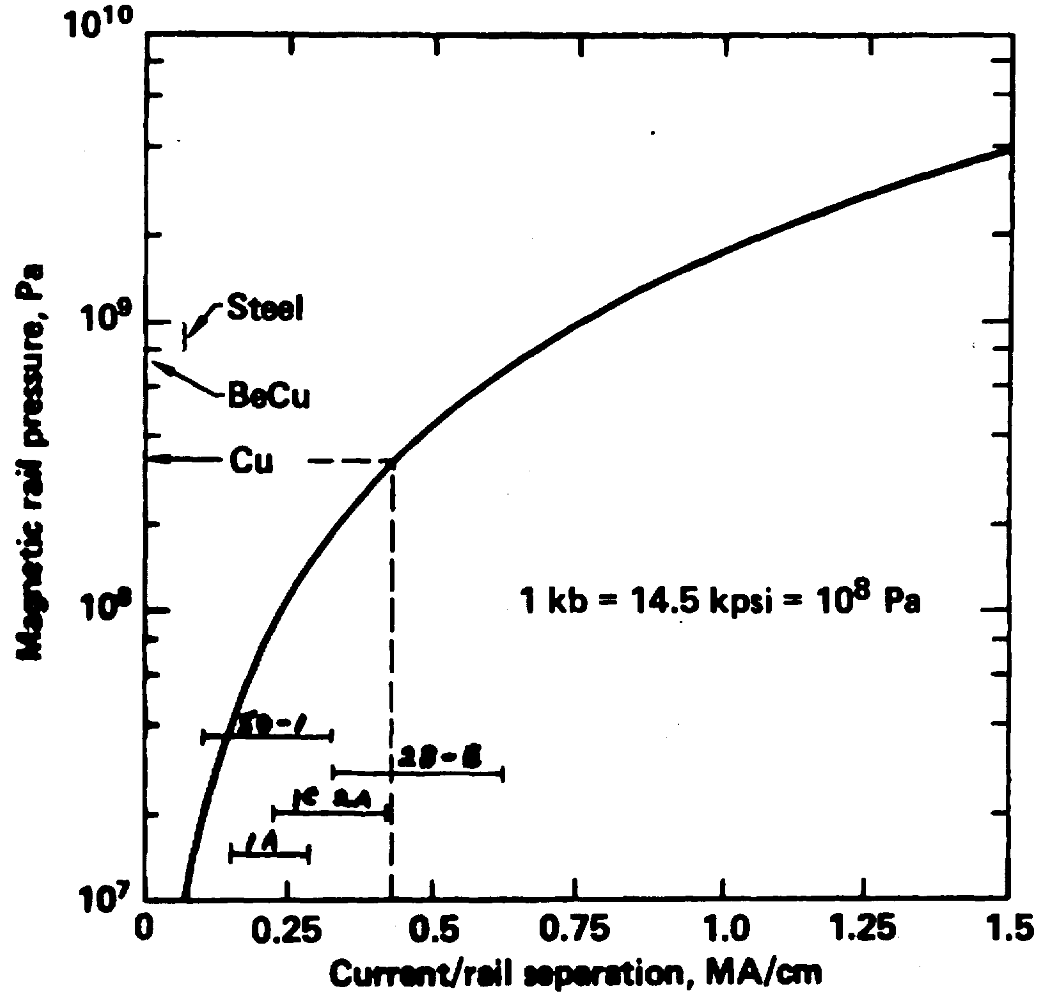
GIVEN:		THEN:
$v = 25$ km/s $A_M = 2.9\ (10^5)$ m/s^2	$\longrightarrow$	$Z = 2.16$ km $T = 86$ ms
		$I_M = \left(\frac{2MA_M}{L_1}\right)^{1/2}$
$L_1 = 0.5$ H/m	$\longrightarrow$	$= 1.08\ (10^6)\ M^{1/2}$ A
$I/H = 50$ kA/mm	$\longrightarrow$	$H = .021\ M^{1/2}$ m

CURRENT CONCENTRATION HAS EXCEEDED THE STRENGTH LIMIT

BORE (mm)	I_{PK} (MA)	H (mm)	I/H (kA/mm)	PRESSURE (MP_A)	(KSI)
12.7	1.3	19	68	820	(119)
90	2.0	60	33	200	(28)
30	1.0	40	25	110	(16)
18	0.92	18	51	460	(67)
9.5	0.37	9.5	39	270	(39)

$T_Y/_{CU} = 350\ MP_A$

THE EXPERIMENTS TESTED THE RAIL DEFORMATION CRITERIA



ACCELERATION AND PROJECTILE/SABOT STRENGTH LIMIT CURRENT

$$P_{AVE} = \frac{F}{A} = \frac{L_1 I^2}{2A}$$

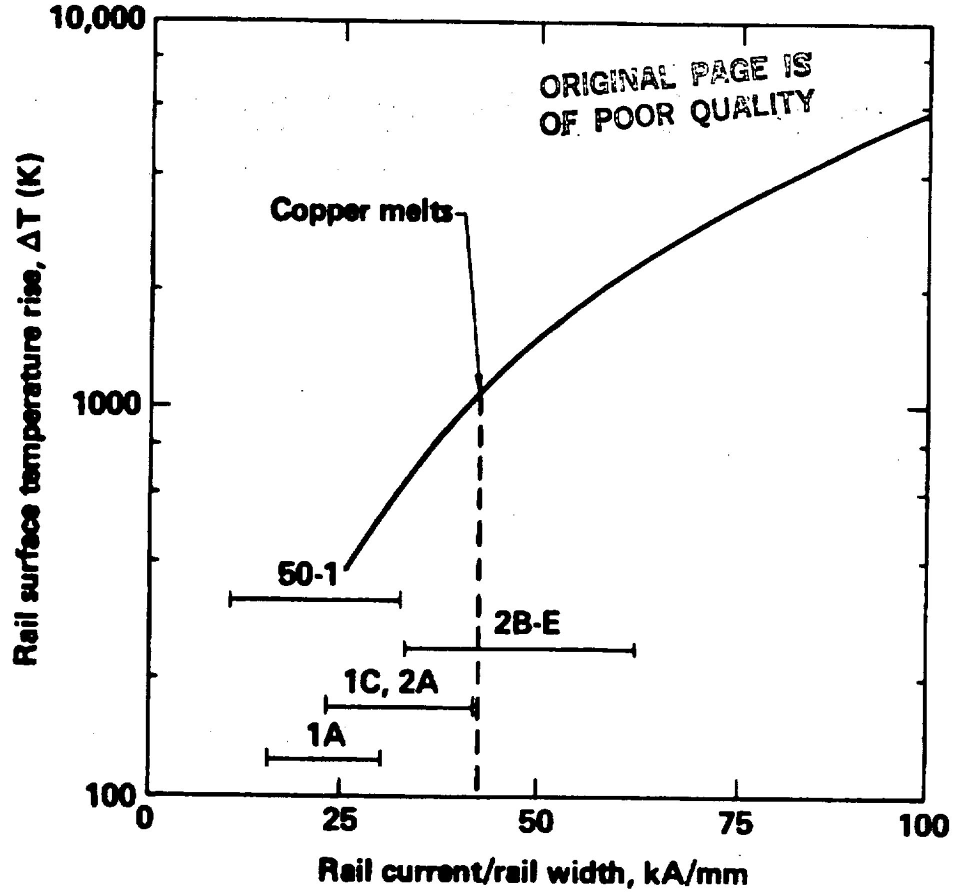
$$\frac{I}{\sqrt{A}} = \left[\frac{2}{} \frac{P_{AVE}}{L_1}\right]^{1/2}$$

LEXAN T_Y = 70 MPA

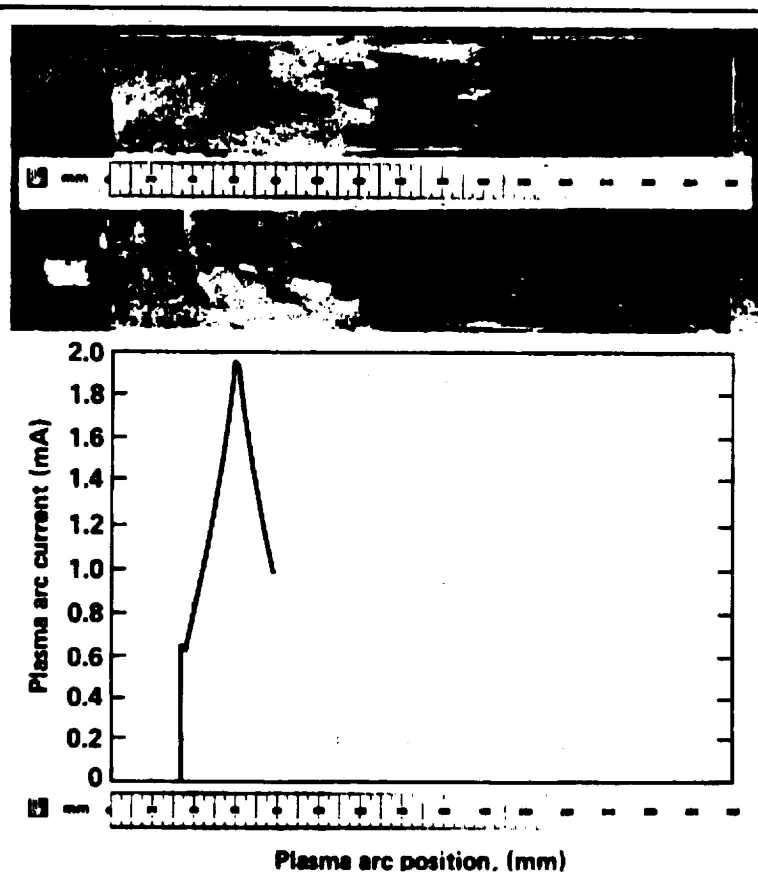
IF: $P_{AVE} = 10\ T_Y$

THEN: $\frac{I}{\sqrt{A}}$ = 53 KA/MM

THE EXPERIMENTS TESTED THE RAIL-MELTING CRITERIA



CURRENT MAGNITUDE IS NOT THE SOLE RAIL DAMAGE MECHANISM



CURRENT CONCENTRATION IS LIMITED BY SEVERAL FACTORS

LIMITING FACTOR		THEORETICAL LIMIT	EXPERIMENTAL LIMIT
1) RAIL STRENGTH	(STEEL)	75 KA/MM	
	(CU)	45	70?
2) SABOT STRENGTH	(LEXAN)		
	($P = T_Y$)	17	
	($P = 10\ T_Y$)	53	70
3) RAIL MELTING	(CU)		
	(RESISTIVE)	43	70?
	(EROSIVE)	?	10 (V=0)

RAIL LAUNCHER PERFORMANCE IS PARTIALLY UNDERSTOOD

FOR I = CONSTANT:

ENERGY LOSS IN RAILS:

$$E_R = \frac{32}{15HI} (\pi \mu_0 \rho)^{1/2} (M/L_1)^{3/2} V^{5/2}$$

ENERGY LOSS IN PLASMA:

$$E_A = \int V_A I \, DT$$

$$V_A = F(I, H, V \ldots)$$

ENERGY REMAINING IN MAGNETIC FIELD:

$$E_I = 1/2\, L_{\ell}\, Z I^2$$

KINETIC ENERGY OF PROJECTILE

$$E_P = 1/2\, MV^2$$

COMPARISON OF FOUR RAIL LAUNCHER CONCEPTS

	CONVENTIONAL		AUGMENTED FIELD	
	SINGLE-STAGE	MULTI-STAGE	SINGLE-STAGE	MULTI-STAGE
ADVANTAGES				
SIMPLE SWITCHING	X		X	
GRADUATED INPUT		X		X
LAUNCH INTERUPT		X		X
VELOCITY CONTROL		X		X
LOW RAIL CURRENT			X	X
"CONSTANT" CURRENT		X		X
DISTRIBUTED ENERGY STORE		X		X
INTEGRAL ENERGY STORE				X
DISADVANTAGES				
LARGE POINT ENERGY STORE	X		X	
LAUNCHING COMPLEXITY		X		X
SWITCHING COMPLEXITY		X		X
EFFICIENCY (%)	10-20	20-50	20-50	20-50+

APPENDIX F

CONCEPT DEFINITION MEETING
(Selected Vugraphs by H. Swift, PAI, Inc.)

Held at Battelle Columbus Laboratories
on
12-13 August 1981

DART LAUNCH CRITERIA

- VELOCITY LOSS THROUGH ATMOSPHERE - BUDGET 1 KM/S
- DART SURVIVAL DURING LAUNCH - BUDGET 30 KG'S
- DART SURVIVAL THROUGH IDEAL ATMOSPHERIC FLIGHT
  - AERODYNAMIC HEATING
  - MECHANICAL COMPRESSIVE STRESSES
  - MECHANICAL SHEAR STRESSES
- DART SURVIVAL THROUGH NONIDEAL ATMOSPHERIC FLIGHT
  - WIND
  - NONPRECIPITATING CLOUDS
  - PRECIPITATING CLOUDS

VELOCITY LOSS THROUGH ATMOSPHERE

- $U_\infty = U_m \exp -\left(\frac{\rho_\infty X_{ct} A_d C_d}{2 M_d}\right)$
- $\beta = \frac{M_d}{A_d C_d}$
- $U_\infty = U_m \exp -\left(\frac{\rho_\infty X_{ct}}{2 \beta}\right)$
- $X_{ct} = \frac{P_o}{g \rho_\infty}$
- $\beta_{ct} = \frac{P_o}{2 g \ln U_\infty / U_m}$
- $U_\infty = U_m \exp -\left(\frac{P_o}{2 g \beta}\right)$

U_m = MUZZLE VELOCITY = 20 KM/S
U_∞ = DART VELOCITY BEYOND ATM. = 19 KM/S
ρ_∞ = ATM DENSITY, SEA LEVEL = 1.3 KG/M^3
X_{ct} = FLIGHT DISTANCE AT SEA LEVEL TO INTERCEPT ONE ATM = 7.98 KM
A_d = MAX CROSS SECT. AREA OF DART
C_d = DRAG COEF
M_d = DART MASS
β = BALLISTIC COEF
P_o = ATMOSPHERIC PRES, SEA LEVEL
g = ACCEL. OF GRAVITY
β_{ct} = CRIT B TO MEET 1 KM/S VELOCITY LOSS CRITERION = 1.007 x 10^5 KG/M^2

DRAG COEFFICIENT

- $C_d = C_{d_n} + C_{d_b} + C_{d_s}$

- $C_d \approx C_{d_n}$

C_{d_n} = NUTIONAN PRESSURE DRAG

C_{d_b} = BASE DRAG
$\propto P_\infty A_b$

C_{d_s} = SKIN FRICTION DRAG

FOR HYPERSONIC FLIGHT
THROUGH DENSE ATMOSPHERE

- BASE DRAG FORCE $\ll$ PRESSURE DRAG FORCE

- SKIN DRAG FORCE NEGLIGIBLE BECAUSE OF ABLATION "BLOWING"

DRAG COEFFICIENT
FOR
SPHERICALLY BLUNTED CONES AND CONE DARTS

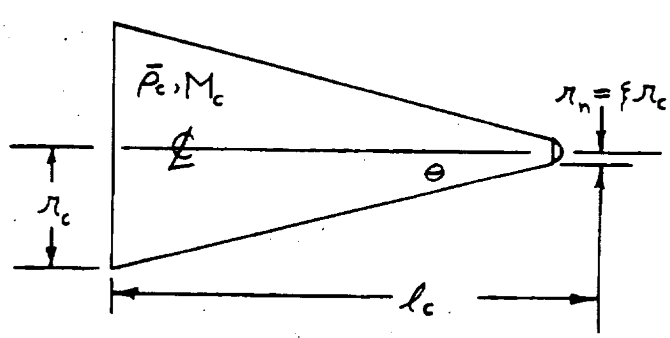
- $C_d \approx C_{d_n} = 0.915\left[(1 - \xi^2\cos^2\theta)(2\sin^2\theta) + \xi^2\cos^2\theta\right]$

  WHERE: $\xi = r_n / r_c$

- $C_d \approx C_{d_n} \approx 1.83 \sin^2\theta$

  WHEN: $\xi \leq 0.1$

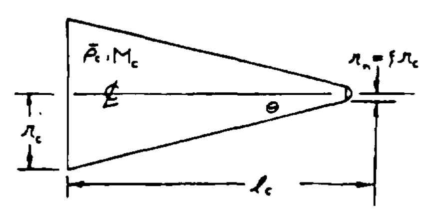
- $C_{d_{ce}} = \dfrac{\beta_{ce} M_d}{A_d}$

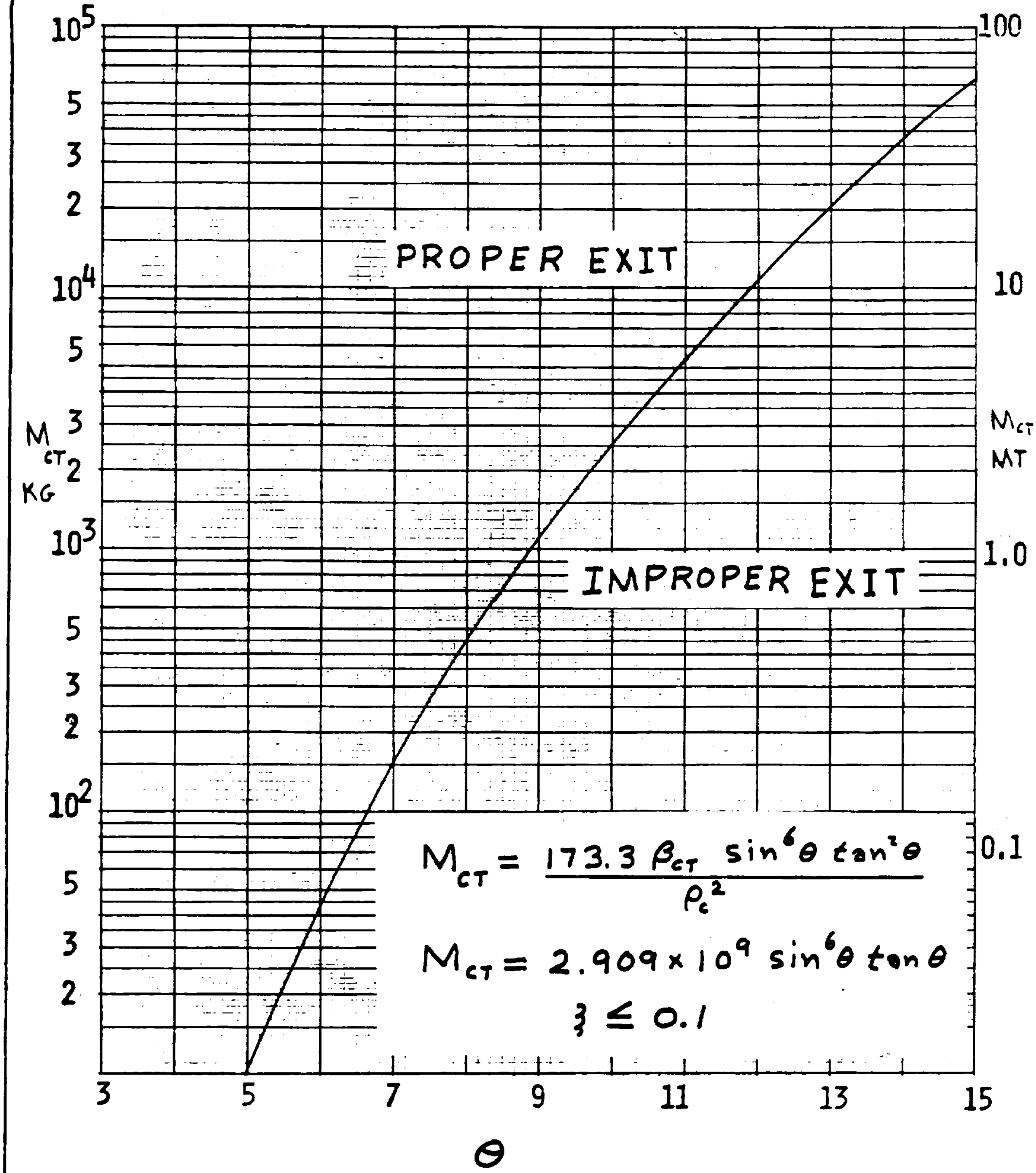


SPHERICALLY BLUNTED CONES

- ASSUME AVERAGE DENSITY $= \bar{\rho}_c = 7.8\ MT/M^3$
- $M_c = \dfrac{\pi \bar{\rho}_c l_c^3 \tan^2\theta}{3} = \dfrac{\pi \bar{\rho}_c r_c^3}{3\tan\theta}$
- $A_c = \pi l_c^2 \tan^2\theta = \pi r_c^2$
- $\beta_c = \dfrac{0.1822\,\bar{\rho}_c l_c}{\sin^2\theta} = \dfrac{0.1822\,\bar{\rho}_c r_c}{\sin^2\theta\tan\theta}$; $\xi \leq 0.1$

  FOR CRITICAL DRAG CONDITIONS
- $M_{cr} = 173.3\,\beta_{cr}^3 \sin\theta^6 \tan^2\theta$
- $M_{cr} = 2.909\times10^9 \sin^6\theta\tan^2\theta$

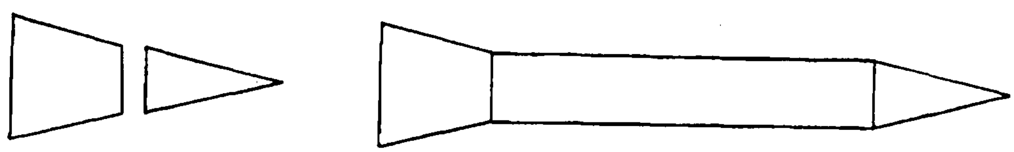




SPHERICALLY BLUNTED CONE MASS VS CONE HALF ANGLE FOR CRITICAL EXIT CONDITIONS.

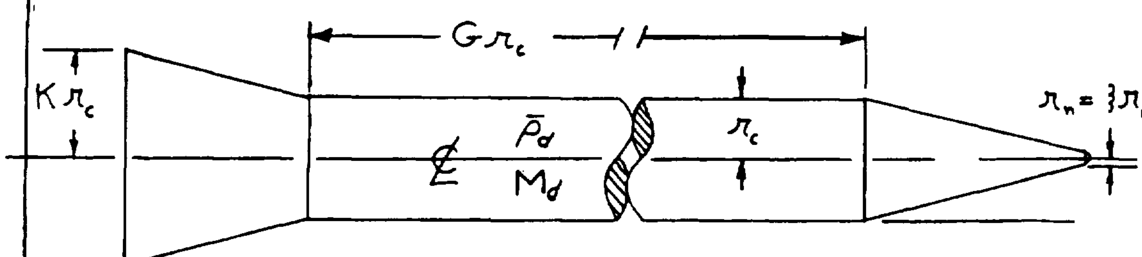
SPHERICALLY BLUNTED CONE DART

- AN EQUIANGLE CONE DART IS A DIVIDED CONE WITH A CYLINDRICAL CENTRAL SECTION.
- DRAG COEFFICIENT, C_d, IS IDENTICAL FOR HYPERSONIC FLOW THROUGH DENSE ATMOSPHERE BECAUSE SKIN AND BASE DRAG ARE NEGLIGIBLE.



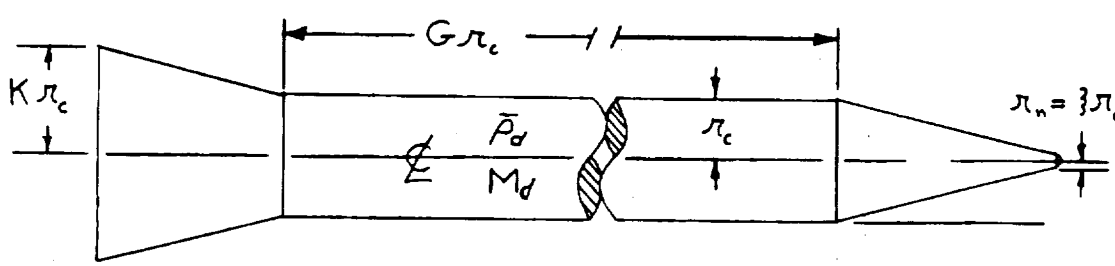
SPHERICALLY BLUNTED CONE DART

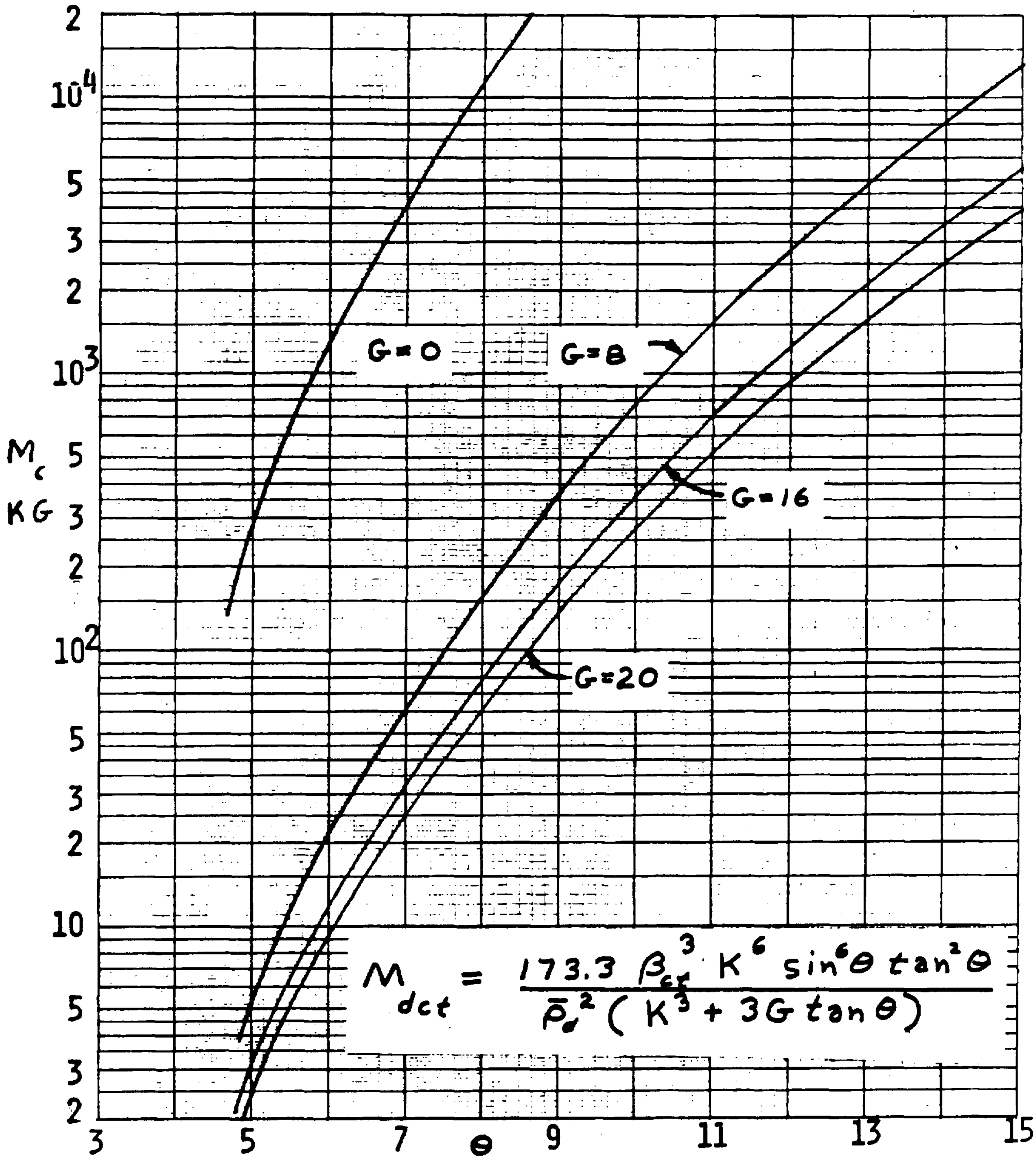
- ASSUME AVERAGE DENSITY, $\bar{\rho}_d = 7.8\ MT/M^3$
- $M_d = \pi \bar{\rho}_d r_c^3 \left(\frac{K^3}{3 \tan\theta} + G \right)$
- $A_d = \pi K^2 r_c^2$
- $\beta_d = \frac{0.5465\ \bar{\rho}_d r_c}{K^2 \sin^2\theta} \left(\frac{K^3}{3 \tan\theta} + G \right)$



CONE DART DESIGN CONSIDERATION

- K SHOULD BE AS SMALL AS FEASIBLE TO PROVIDE AERODYNAMIC STABILITY.
- G SHOULD BE LARGE - LIMITED BY LAUNCH AND STRUCTURAL RIGIDITY CONSIDERATIONS.
- LET : $K = 1.75$; $8 \leq G \leq 20$
- $$M_{dct} = \frac{8.356 \times 10^{10} \sin^6\theta \tan^2\theta}{5.539 + G \tan\theta}$$





SPHERICALLY BLUNTED CONE DART MASS VS
HALF CONE ANGLE FOR CRITICAL EXIT CONDITIONS

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PROJECTILE SURVIVABILITY DURING LAUNCH

- PROJECTILE/SABOT PACKAGE MUST SURVIVE 30 KG'S ACCELERATION
- LOAD MUST BE TRANSMITTED TO PROJECTILE WITHOUT DEFORMATION OR FAILURE
- SABOTS: REDUCE REQUIRED LAUNCH PRESSURE
  ; CAN REDUCE LAUNCH STRESS APPLIED TO PROJECTILE

BASE-LOADING SABOTS

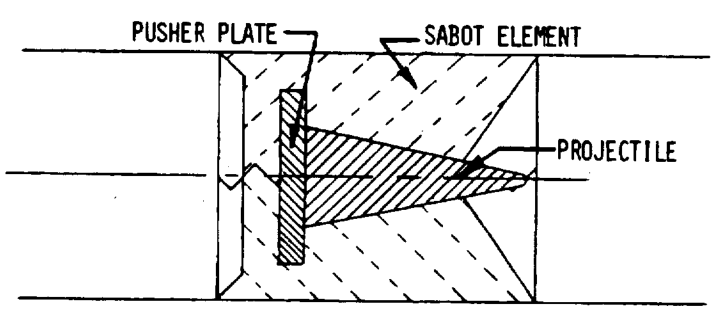
- OVERSIZE SABOT APPLIES ACCELERATION FORCE VIA BASE PRESSURE
- FOR A CONE

$$\sigma_{bc} = \frac{\bar{\rho}_c a_c r_c}{3 \tan\theta}$$

- FOR A CONE DART

$$\sigma_{bc} = \bar{\rho}_d a_c r_c \left(\frac{K}{3 \tan\theta} + \frac{G}{K^2} \right)$$

σ_b = COMP. STRESS AT BASE OF PROJECTILE
$\sigma_{b_{max}}$ = 1.0GPA (150 KSI)
a_c = ACCELERATION LEVEL
= 3×10^5 M/S^2
r_c = CONE BASE RADIUS
θ = CONE HALF ANGLE
G, K = DART SHAPE PARAM.



CRITICAL PROJECTILE PARAMETERS
FOR
BASE LOADING SABOTS

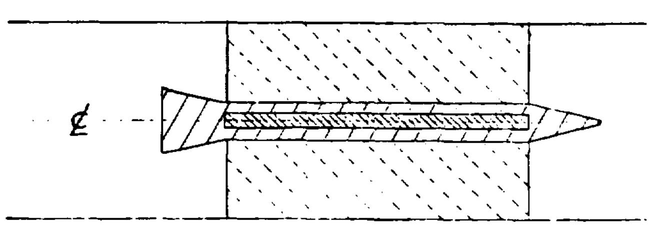
- LIMIT SET BY MINIMUM ACCELERATION AND MAXIMUM MATERIAL STRENGTH
  - $a_c = 3 \times 10^5$ M/S^2 (30 Kg's)
  - $\sigma_b = 1.0$ GPA (147 KPSI)
- CONE

  $\theta \leq 8°$; $r_c \leq 0.175$ M ; $M_c \leq 430$ KG
- CONE DART (K = 1.75 ; G = 16)

  $\theta \leq 8°$; $r_c \leq 0.055$; $M_c \leq 80$ KG

SIDE LOADING SABOTS

- SABOT GRIPS CYLINDRICAL SIDE-WALL OF PROJECTILE (DART)
  - LOAD TRANSFERRED VIA SHEAR STRESS
  - AREA FOR TRANSFER GROWS NEARLY LINEARLY WITH DART MASS WHEN MASS IS INCREASED BY LENGTHENING DART
  - LAUNCH STRESS INCREASES WITH DART RADIUS,
  - THEREFORE, MATERIAL SHEAR STRENGTH LIMIT PLACES LIMIT ON DART DIAMETER



SHEAR LOAD ON A CONE DART

- $\sigma_s = \frac{\bar{\rho}_d r_c a_c}{2}\left(\frac{K^3}{3G\tan\theta} + 1\right)$

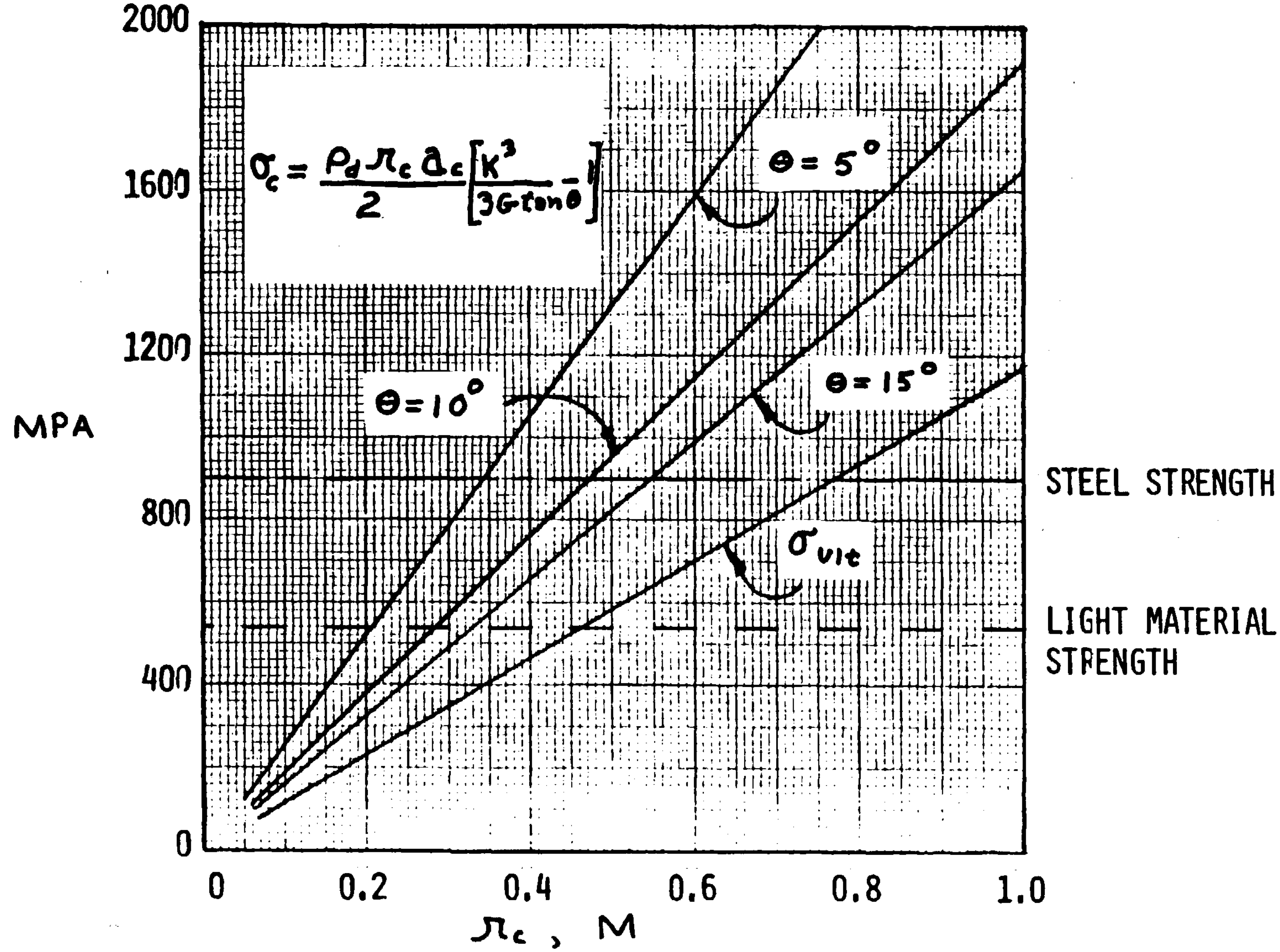
  $r_{max} = 0.211$ M $(\theta = 17°)$

- ULTIMATE SHEAR LOAD,
  - NO CONIC SECTIONS
  - $\sigma_{SUT} = \frac{\bar{\rho}_d a_c r_c}{2}$
    $= 1.17 \times 10^9 r_c$
  - $r_{c\,MAX} = \frac{2\sigma_{SUT}}{\bar{\rho}_d a_c}$

ρ_d = DART MATL DENSITY
= 7.8×10^3 KG/M^3
r_c = DART CYL. RADIUS
a_c = DART ACCELERATION
= 3×10^5 M/S^2
K = BASE RADIUS/CYL. RADIUS
= 1.75
G = CYL. LENGTH/CYL. RADIUS
= 16
θ = CONE HALF ANGLE

SHEAR STRESS ON A CONE DART (STD. CONFIG)

APPLIED BY A SIDE-LOADING SABOT



SIZE OF SIDE LOADING SABOT

- ASSUME SABOT IS A HOLLOW CYLINDER SURROUNDING CYLINDRICAL SECTION OF DART
  - DART MATL. DENSITY = E TIMES AVG. SABOT MATL DENSITY
  - SABOT DIA IS B TIMES DART CYL. DIA.

- $$B = \frac{2E}{G + P_s/\rho_s r_c a_c}\left(\frac{K^3}{3\tan\theta} + G\right)$$

- $$B = \frac{2E\left(K^3/3\tan\theta + G\right)}{G + 0.5465\, P_s E/(\beta_{dc} a_c K^2 \sin^2\theta\,[\ ])}$$

$$[\ ] = \frac{K^3}{3\tan\theta} + G$$

G = 16; K = 1.75; ; DART SHAPE PARAMETERS
ρ_s = AVG. SABOT DENSITY = 1.6 x 10^3 KG/M^2
P_s = AVG. SABOT BASE PRES. = 680 MPA
r_c = DART CYL RADIUS
a_c = PACKAGE ACC. = 3 x 10^5 M/S^2
E = 4.875
β_{dc} = CRITICAL DART BALLISTIC COEF = 1.007 x 10^5 KG/M^2

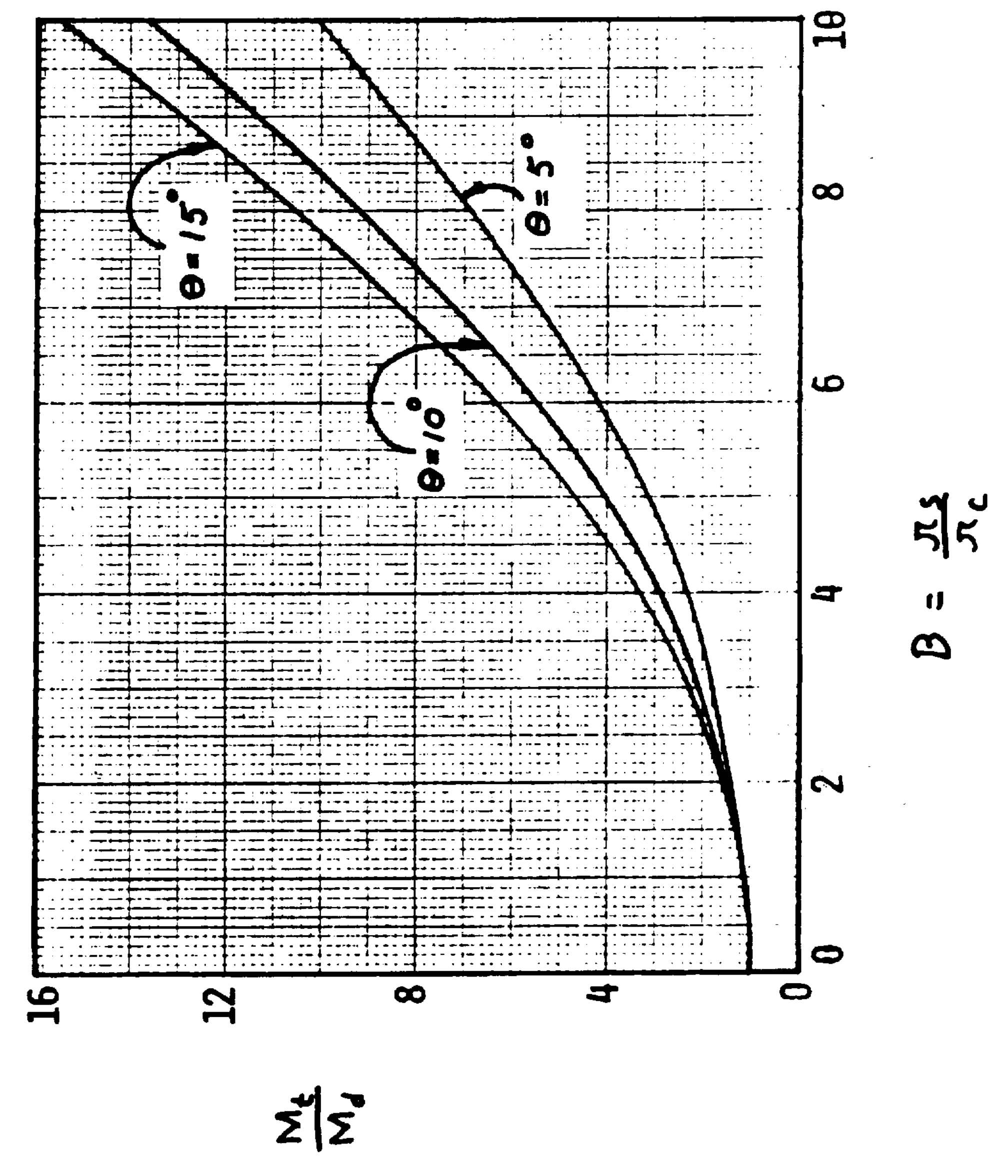
MASS OF A SIDE-LOADING SABOT

- $$\frac{M_t}{M_d} = \frac{B^2}{E\left(K^3/3\tan\theta - G\right)} + 1$$

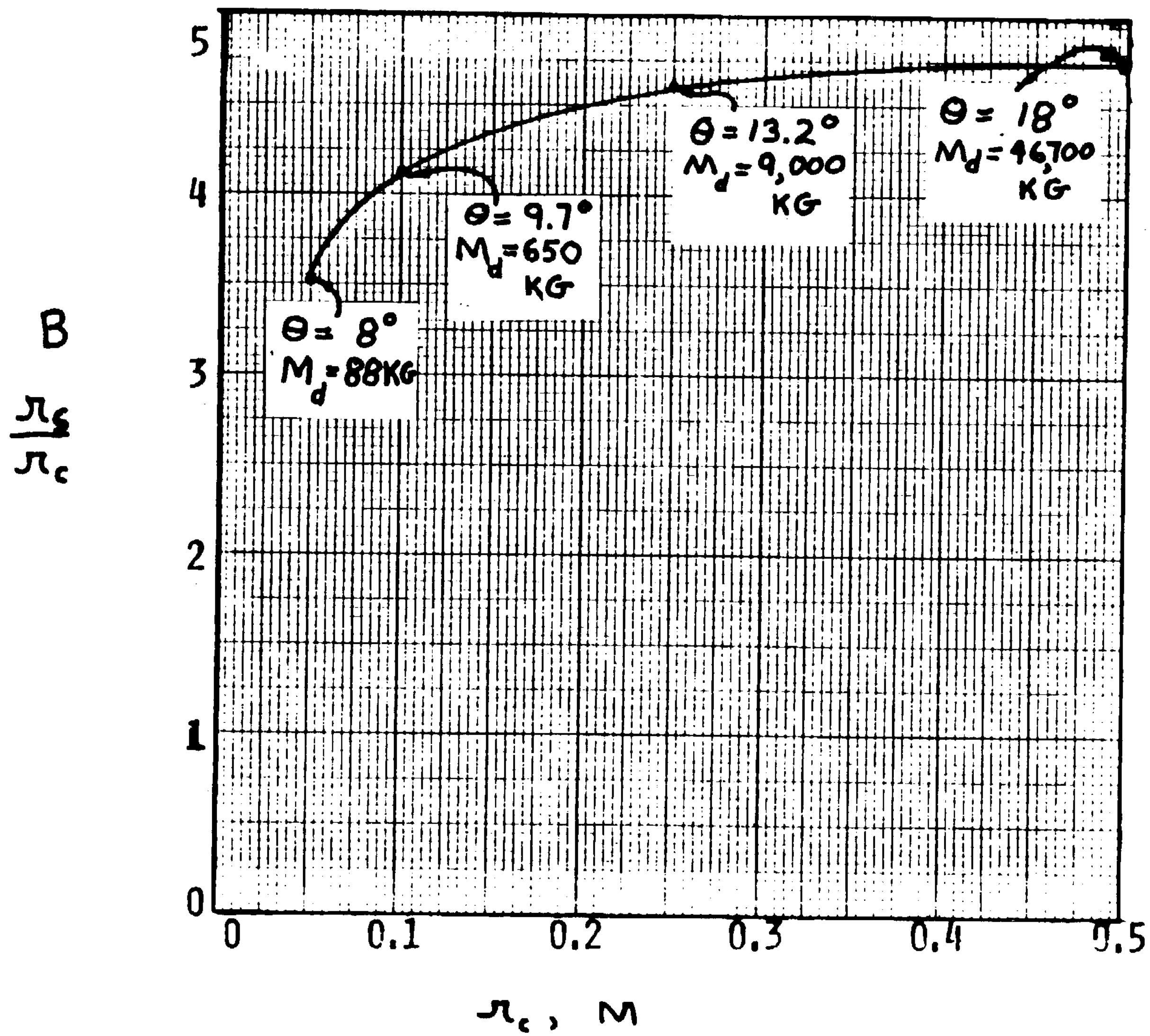
RATIO OF PACKAGE MASS TO DART MASS

VS

RATIO OF SABOT DIA TO DART DIA



RATIO OF SABOT/DART DIAMETER
VS
DART CYLINDER RADIUS
FOR
ACHIEVING REQ. ACC. WITH Ps = 680 MPA



AERODYNAMIC HEATING

- HEATING INTENSE BECAUSE OF HIGH VELOCITY AT LOW ALTITUDE
- HEATING & ABLATION SAME FOR 8.0 KM FLIGHT HORIZONTALLY
- TOTAL HEAT INPUT LESS THAN NORMAL ICBM REENTRY BECAUSE OF SHORTENED TRAJECTORY
- HEATING PREDICTIONS AT STAGNATION POINT & ELSEWHERE REQUIRE SEPARATE ANALYSES

STAGNATION POINT HEATING AND ABLATION

EXPERIMENTAL DATA AVAILABLE TO:

U = 18 KM/s
ρ_∞ = AMBIENT AT SEA LEVEL

STAGNATION CONDITIONS

- $T_s = 30{,}000\,°K$
- $P_s = 467$ MPA (68.6 KPSI)
- $H_s = 4.12 \times 10^7$ J/KG

HEATING RATE AT SEA LEVEL

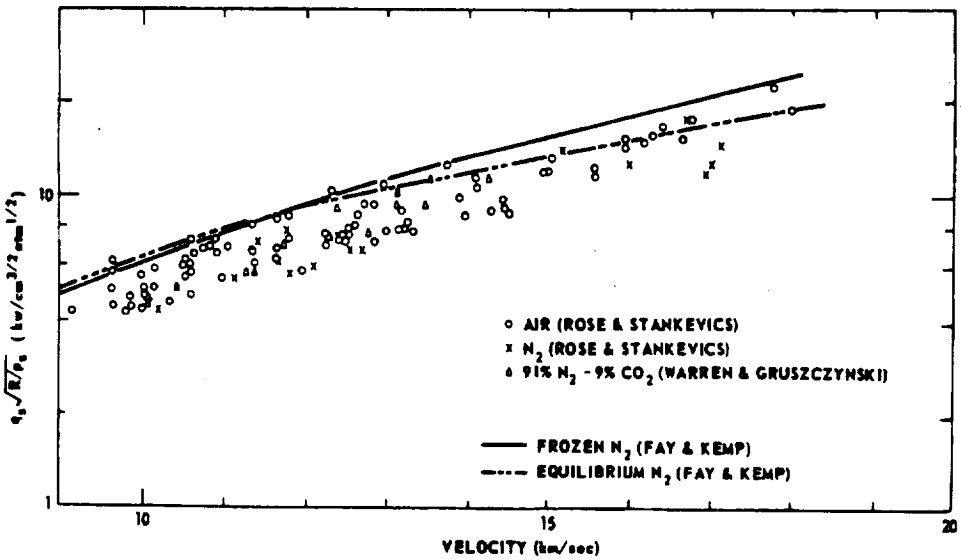
$$\dot{Q}_s = 1.108 \times 10^{11}\sqrt{\rho_\infty}\ W/M^2$$

SURFACE RECESSION OF AN EXCELLENT ABLATOR, S

$$S = \frac{\dot{Q}_s\,\tau\,(1 - H_w/H_s) - \dot{Q}_R\,\tau}{\rho_A\,Q^*}$$

τ = FLIGHT TIME = 0.45 S
H_w = WALL ENTHALPY
= 9.57×10^5 J/KG
$\dot{Q}_R$ = RADIANT FLUX
= 6.46×10^6 W/M^2
ρ_A = ABLATOR DENSITY
= 1.42×10^3 KG/M^3 FOR NOVALAK
Q^* = HEAT OF ABLATION
= 9.57×10^7 J/KG (WITH BLOWING CORRECTION)

$S = 7.42$ mm



STAGNATION POINT HEATING DATA AT VELOCITIES UP TO 18 KM/SEC AND ρ_∞ = AMBIENT

ABLATION AWAY FROM STAGNATION POINT

- ABLATION RATE MAINLY CONTROLLED BY CONE ANGLE SINCE IT CONTROLS AIR HEATING & DENSITY
- ABLATION RATE VARIES SLOWLY WITH DISTANCE FROM STAGNATION POINT
- ANALYSIS IS COMPLEX BUT ALL STEPS ARE PROVEN
- ABLATION OF NOVALAK 30 CM BEHIND STAGNATION PT:

●	S (SURFACE RECESSION)
5°	1.21×10^{-3} M
10°	2.29×10^{-3} M
15°	3.73×10^{-3} M
STAG. PT.	7.42×10^{-3} M

MECHANICAL COMPRESSIVE STRESS

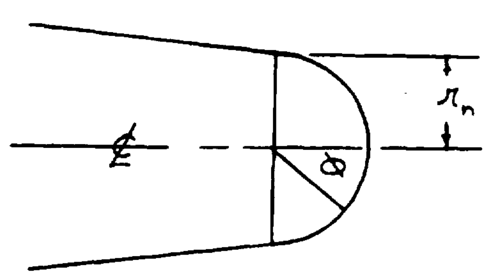
- COMPRESSIVE STRESS GREATEST AT STAGNATION POINT, σ_c
- COMPRESSIVE STRESS DIMINISHES WITH $\cos^2$ OF PRESENTED ANGLE
- $\sigma_c = \rho_\infty U^2 \cos^2 \phi$
- STAGNATION STRESS AT LAUNCHER MUZZLE:

  $\sigma_c = 474\ \text{MPA} = 69.7\ \text{KPSI}$

ρ_∞ = AMBIENT AIR DENSITY
= 1.186 KG/M^3 AT S.T.P.
U = VEHICLE VELOCITY
ϕ = ANGLE BETWEEN VELOCITY VECTOR AND LOCAL SURFACE NORMAL

MECHANICAL SHEAR STRESS

- LOCAL SURFACE SHEAR STRESS PRODUCED BY AIR FLOW EQUALS GRADIENT OF NORMAL STRESS
- SHEAR STRESS BEYOND MATERIAL STRENGTH WILL PRODUCE RAPID MATERIAL CRUMBLING
- $\sigma_s = \dfrac{\sigma_c (1 - \cos^4 \phi)}{4 \sin \phi \cos \phi}$ for $\phi \leq 60°$
- $\sigma_s = \dfrac{1.904 \times 10^4 (1 - \cos^4 \phi)}{\sin \phi \cos \phi}$



NON IDEAL ATMOSPHERIC CONSIDERATIONS

- WIND - UNIMPORTANT
- NON PRECIPITATING CLOUDS - SERIOUS
- PRECIPATATING CLOUDS - IMPOSSIBLE

NON PARCIPITATING CLOUDS

- FINE WATER/ICE PARTICLE WILL ERODE PROJECTILE
- AT M $\approx$ 60 SHOCKWAVE STANDOFF INSUFFICIENT TO PROTECT VEHICLE VIA PRRTICLE BREAKUP
- APPROXIMATE CONDITIONS
  - GROSS PARTICULATE DENSITY $\rho_p = 2.0 \times 10^{-4}$ KG/M^3
  - CLOUD THICKNESS $L_c \approx 2 \times 10^3$ M
  - MATERIAL REMOVAL EFFICIENCY $\eta_{ep} = 1.00 \times 10^{-8}$ M^3/J
- SURFACE RECESSION

$$X_R = \frac{\eta_{ep} \rho_p L_c U_p^2 \sin^2\theta}{2}$$

$$X_{R_{STGN}} = 0.80 \text{ M} \; !$$

U_p = PROJ VEL
= 2 x 10^4 M/S
θ = ANGLE OF LOCAL SURFACE NORMAL TO VELOCITY

NONPRECIPITATING CLOUDS II

- PARTICULATE EROSION IS UNACCEPTABLE
- ALTERNATIVES
  - FIRE ONLY INTO A CLEAR SKY
  - CHOOSE A TOUGHER ABLATOR, METALS WHICH DON'T OXIDIZE

POINT DESIGN
DART/SABOT PACKAGE FOR LAUNCHING A USEFUL PAYLOAD

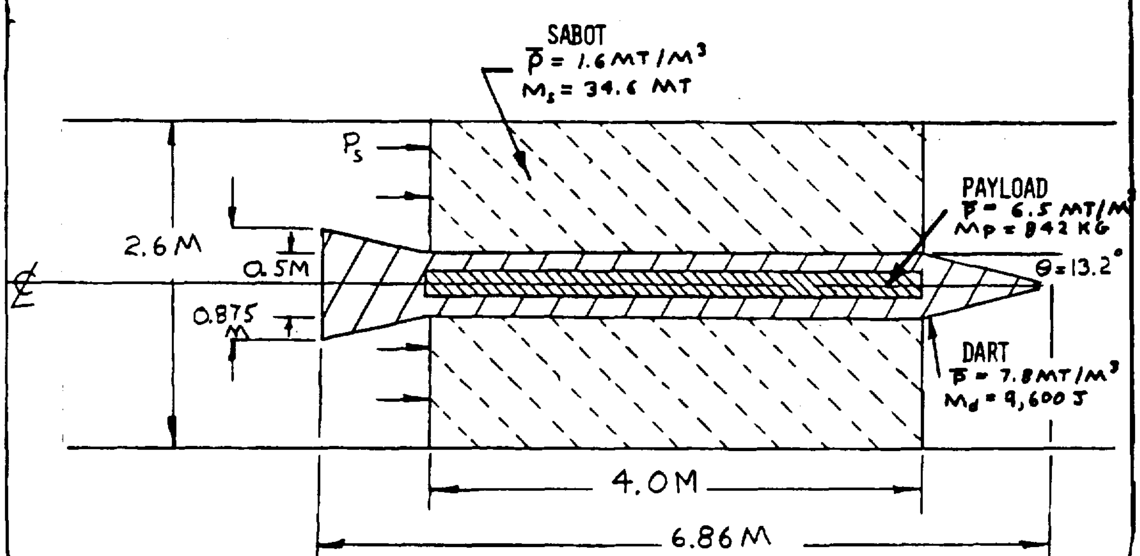
- PAYLOAD
  0.2 M DIA X 4 M LONG CERMET ROD M = 842 KG
  0.49 M DIA X 4 + M LONG STEEL SHIELD M = 5300 KG
- REAR CONE FLAIR MUST BE K = 1.75 TIMES DIA. OF CYLINDER SECTION (.50 M) FOR STABILITY
- CYLINDRICAL SECTION LENGTH IS 6=16 X RADIUS
- MAX CONE ANGLE TO PRODUCE NEEDED FOR PROPER ESCAPE $\theta = 13.2°$
- DART MASS M_D = 9.04 MT.
- SABOT DIAMETER TO PRODUCE ACCELERATION OF 3 X 10^5 M/S^2 WITH BASE PRES. Ps = 680 MPA (100 KS I)
  B = 5.2 (SABOT DIA = 2.6 M)

POINT DESIGN
DART/SABOT PACKAGE FOR LAUNCHING
A USEFUL PAYLOAD (CONT)

- SABOT MASS: $M_s = 34.6$ MT
  $\rho_s = 1.6 \times 10^3$ KG/M^3
- PACKAGE MASS: $M_T = 44.16$ MT
- KINETIC ENERGY $E_P = 8.33 \times 10^{12}$ J
  - TOTAL ENERGY/SHOT (EFFICIENCY≈30%) $E_{PT} = 2.78 \times 10^{13}$ J
  - FUEL TO SUPPLY TOTAL ENERGY (40% CONVERSION EFF) 33,800 MT
  - 3520 TONS OF FUEL/TON OUT OF SOLAR SYSTEM

LARGE SABOT DART CONFIGURATION

- TOTAL MASS 44.26 MT
- VELOCITY 20.0 KM/S
- KINETIC ENERGY 8.33 TJ
- SABOT/DART SHEAR STRESS 292 MPA



CONCLUSIONS

- PROJECTILE MUST HAVE MINIMUM BALLISTIC COEFFICIENT TO FUNCTION
  - 1.007×10^5 KG/M^2
- PROJ. SHAPE ALTERNATIVES
  - SPHERICALLY BLUNTED CONE
  - CONE DART
- AERODYNAMIC DRAG IS ALMOST EXCLUSIVELY NUTONIAN PRESSURE DRAG
- PROJECTILES MUST BE SABOTED DURING LAUNCH
  - BASE LOADING SABOTS ARE LIMITED IN USABLE SIZE OF PROJECTILES BY COMPRESSIVE STRESS.
  - SIDE LOADING SABOTS HAVE MUCH MILDER STRESS LIMITATIONS - USABLE WITH CONE DARTS ONLY

CONCLUSIONS (CONT2)

- SABOT LAUNCH LIMITS

PROJECTILE		
CONE	0.175M	430 KG
CONE DART	.055 M	80 KG
CONE DART WITH SIDE LOAD SABOT	0.35 M	23,800 KG

- SABOT MASS MUST BE SEVERAL TIMES DART MASS
- AERODYNAMIC HEATING IS EXTREMELY INTENSE ... BUT TOTAL HEAT INPUT IS LESS THAN TYPICAL ORBITAL REENTRY
  - STAGNATION RECESSIONS FOR FINE-QUALITY ABLATORS $\approx$ 1 CM.
  - ABLATION ENTHALPY INCREASE DUE TO "BLOWING" NEEDS MORE INVESTIGATION
- COMPRESSIVE & SHEAR STRESSES FROM AERODYNAMICS WILL LIMIT ABLATOR CHOICES
- EROSION FROM PARTICULATES IN LIGHT CLOUDS IS DEVASTATING TO HIGH-PERFORMANCE ABLATORS

CONCLUSIONS (CONT3)

- POINT DESIGN BASED ON INITIAL BATTELLE PAYLOAD RECOMMENDATION IS POSSIBLE BUT IS ENORMOUS.

- NO FUNDAMENTAL OBJECTIONS TO THE CONCEPT WERE LOCATED
- AREAS NEEDING MORE CONSIDERATION
  - AEROTHERMAL HEAT INPUT
  - ABLATOR PERFORMANCE AUGMENTATION FROM INTENSE BLOWING
  - INTENSE ABLATION CHARACTERISTICS OF UNCONVENTIONAL ABLATORS
    REFRACTORY METALS
    OXIDATION RESISTANT ALLOYS
  - STRESS DISTRIBUTION WITHIN PRACTICAL SABOT DESIGNS
  - EROSION RESPONSE OF MATERIALS AT $U_p > 10$ KM/S

APPENDIX G

CONCEPT DEFINITION MEETING
(Selected Vugraphs by A. Buckingham, LLNL)

Held at Battelle Columbus Laboratories
on
12-13 August 1981

OBJECTIVES

Restate questions, analysis, modeled behavior, solutions in framework of continuum Fluid–Solid Dynamics.

- Launch Friction, Drag, Energy & Mass Loss ?
- Gasdynamic Loads, Launch Stresses ?
- Interaction between gas cap viscous htg. & radiation ?
- Drag vs. Ablation mass loss – Initiate cascade ?
- Implication on Materials, Launcher and Projectile Config.

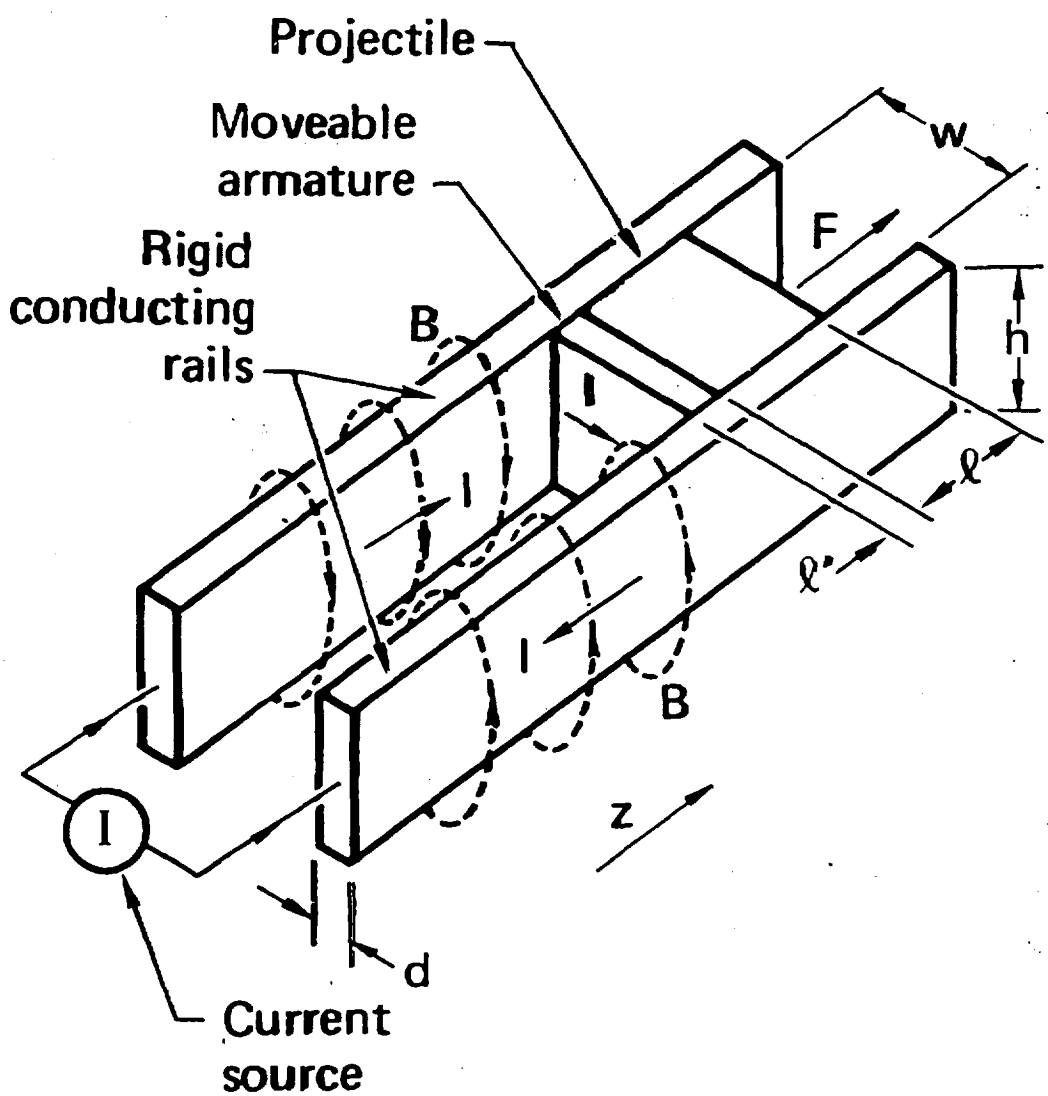
CONTENTS

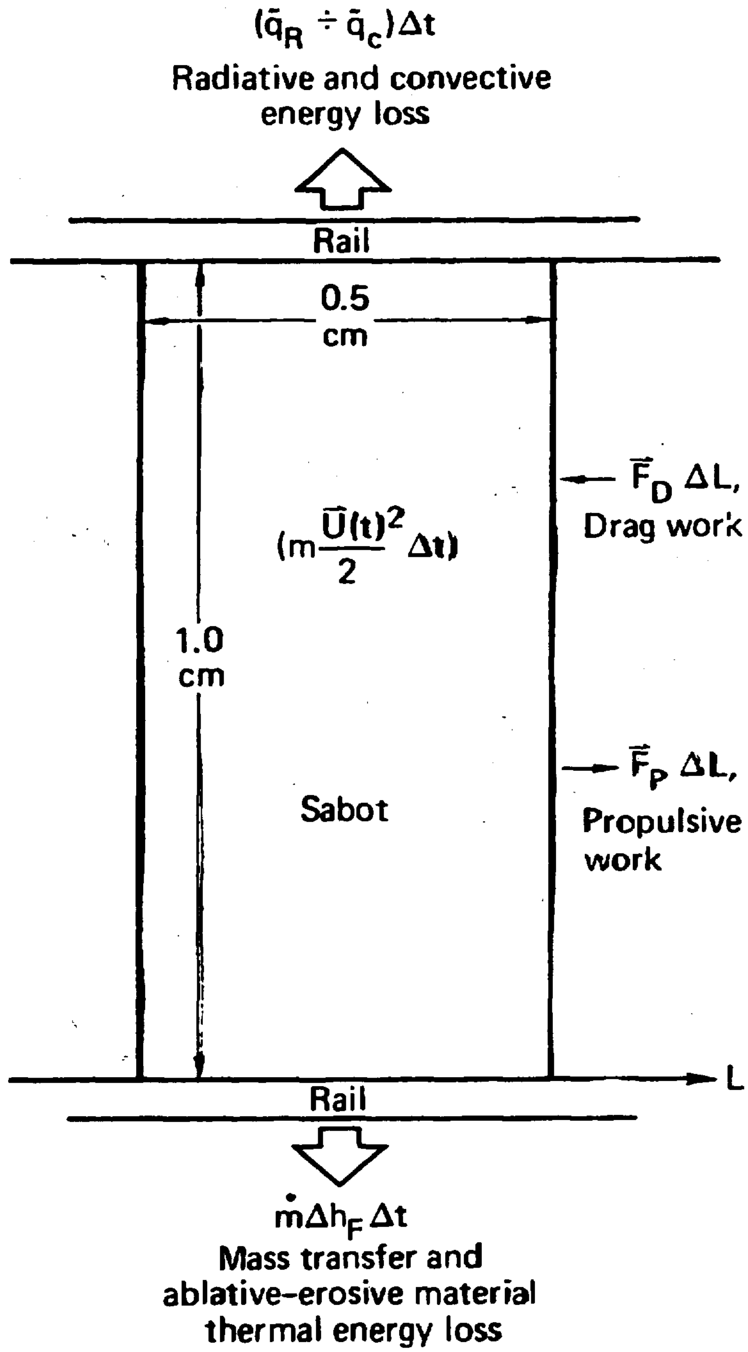
- BACKGROUND
- LAUNCH PERIOD
- MUZZLE EXIT
- HYPERVELOCITY HEATING
- COUPLED DRAG & ABLATION
- CONFIGURATIONS & TRAJECTORIES
- SUMMARY

BACKGROUND

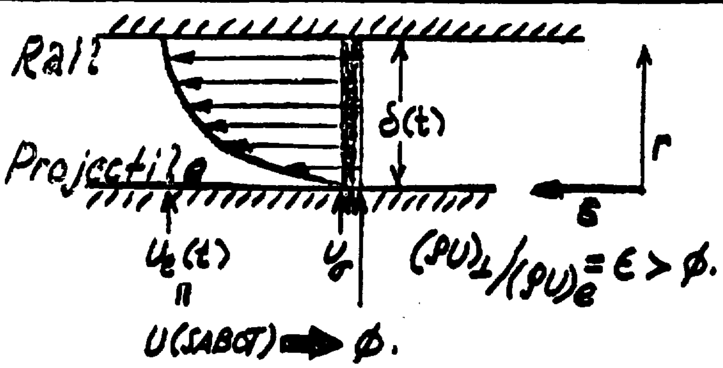
- Berlin, 1944, Storage Batteries, 10 g at 1 km/s
- San Ramon, Calif., 0.3 g at 9.5 km/s
- A.N. University, Canberra, homopolar, 3 g at 5.9 km/s
- a. Matl's. Res. Lab., Melbourne, Capacitor Banks
- LLNL & LANL
- Univ. of Texas, Capacitor Bank, 50 MJ homopolar
- Westinghouse Corp., 15 MJ homopolar 300 g at 3 km/s
- Japan, Tokyo Institute of Tech., High Pressure EOS.

  - Current revision, gas injection.




$(\tilde{q}_R \div \tilde{q}_c)\Delta t$
Radiative and convective energy loss
Rail
0.5 cm
$\vec{F}_D \, \Delta L$,
Drag work
$(m\frac{\overline{U}(t)^2}{2}\Delta t)$
1.0 cm
$\vec{F}_P \, \Delta L$,
Propulsive work
Sabot
L
Rail
$\dot{m}\Delta h_F \, \Delta t$
Mass transfer and ablative-erosive material thermal energy loss

DECOMPOSING PROJECTILE - COUETTE FLOW



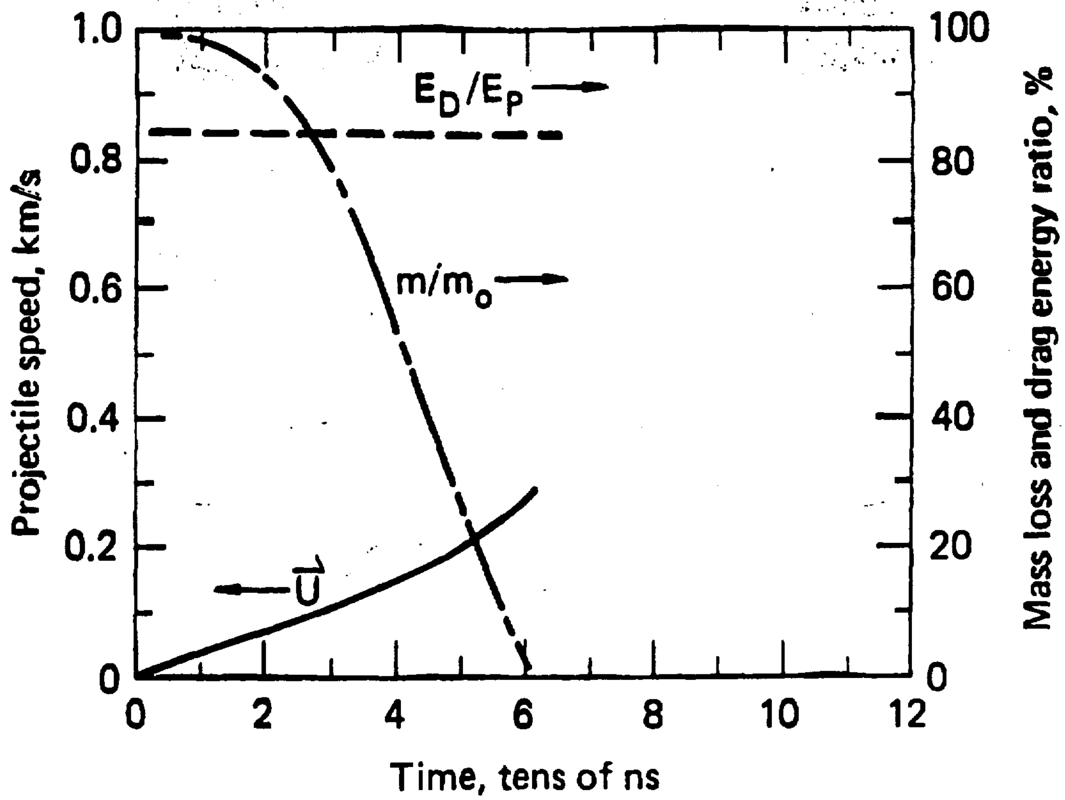
$$\left\{\begin{array}{l} \frac{\partial \rho}{\partial t} + \frac{\partial \rho u r^k}{\partial s} + \frac{\partial \rho u_\perp r^k}{\partial r} = \phi \\ \rho \frac{\partial K_i}{\partial t} + \rho u \frac{\partial K_i}{\partial s} + \rho u_\perp \frac{\partial K_i}{\partial r} = \\ \qquad \frac{1}{r^k}\frac{\partial}{\partial r}\left[r^k \frac{\partial}{\partial r}\left(\rho \epsilon_{D_i} \frac{\partial K_i}{\partial r} - j_i\right)\right] + \underline{\dot w_i} \end{array}\right.$$

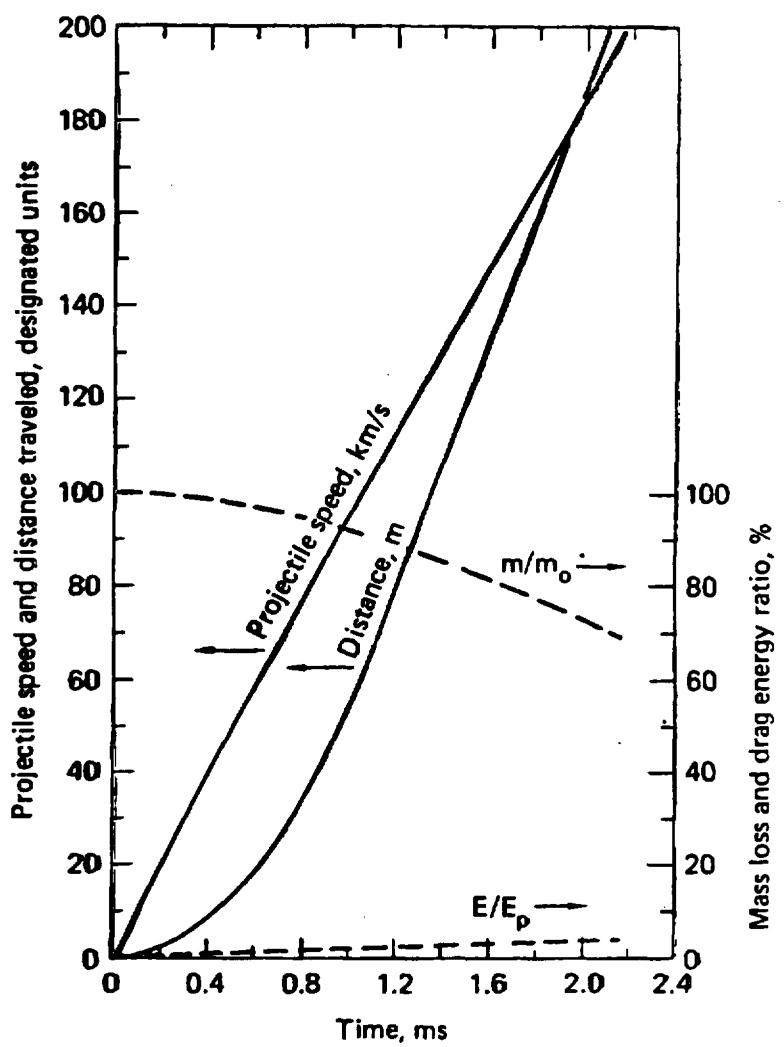
$$\frac{\partial \rho u}{\partial t} + \rho u \frac{\partial u}{\partial s} + \rho u_\perp \frac{\partial u}{\partial r} = \frac{1}{r^k}\frac{\partial}{\partial r}\left[\rho r^k (\nu + \epsilon_M) \frac{\partial u}{\partial r}\right] - \frac{\partial p}{\partial s} \quad ; \quad k = \left\{ \begin{array}{l} 0 \\ 1 \end{array}\right.$$

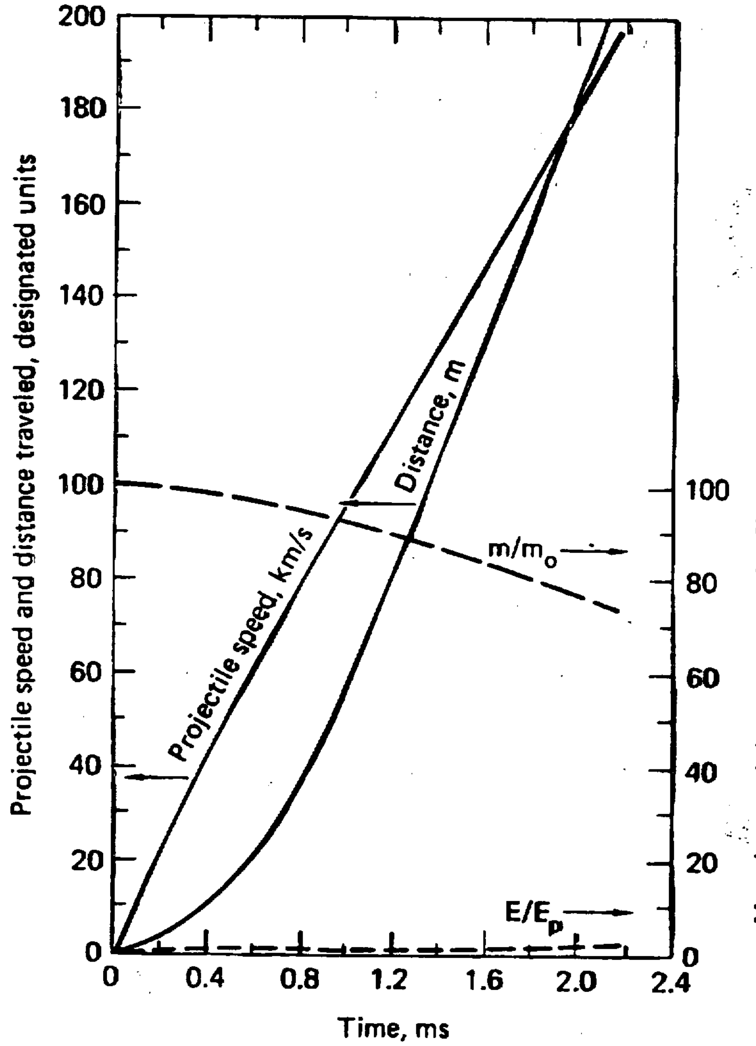
$$\rho \frac{\partial H_T}{\partial t} + \rho u \frac{\partial H_T}{\partial s} + \rho u_\perp \frac{\partial H_T}{\partial r} = \frac{1}{r^k}\frac{\partial}{\partial r}\Big[\rho r^k (\epsilon_M + \nu) \frac{\partial (u^2/2)}{\partial r} + r^k (\lambda + \rho \epsilon_H c_p) \frac{\partial T}{\partial r}$$

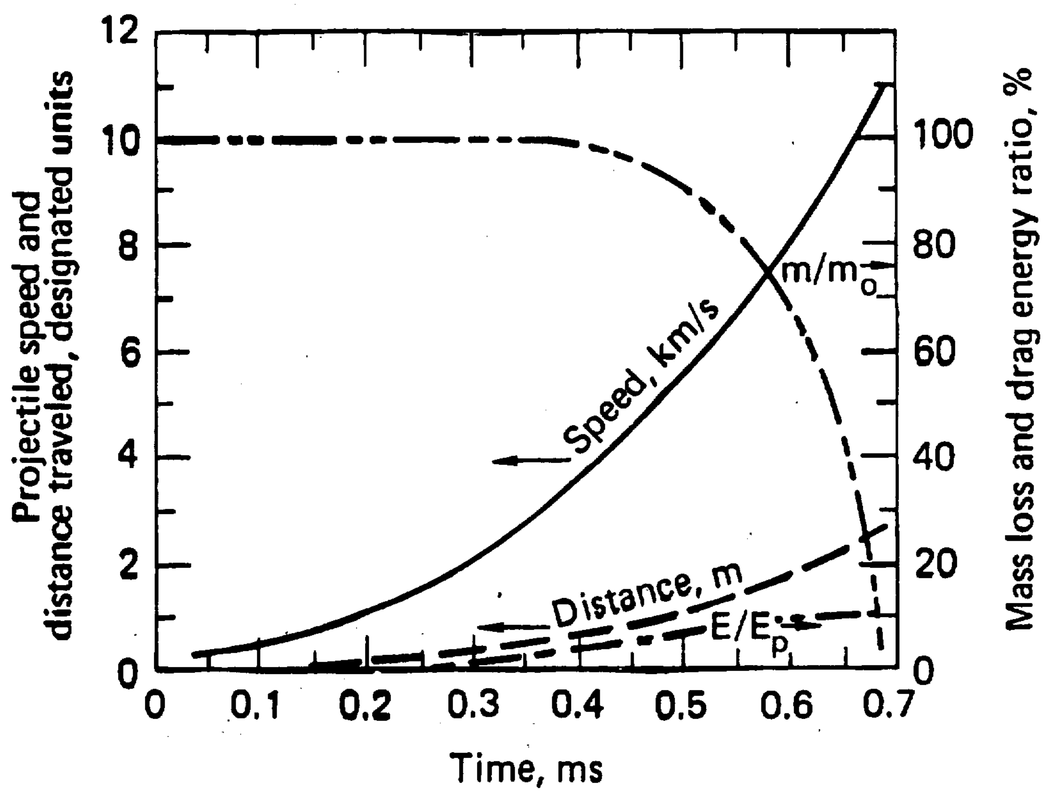
$$+ r^k \sum_i \left(\rho \epsilon_D \frac{\partial K_i}{\partial r} - j_i\right) h_i - \frac{r^k R T}{\rho} \sum_i \sum_j \frac{x_j D_i^T}{M_i \mathcal{D}_{ij}} \left(\frac{j_i}{K_i} - \frac{j_j}{K_j}\right) r^k q^r \Big]$$

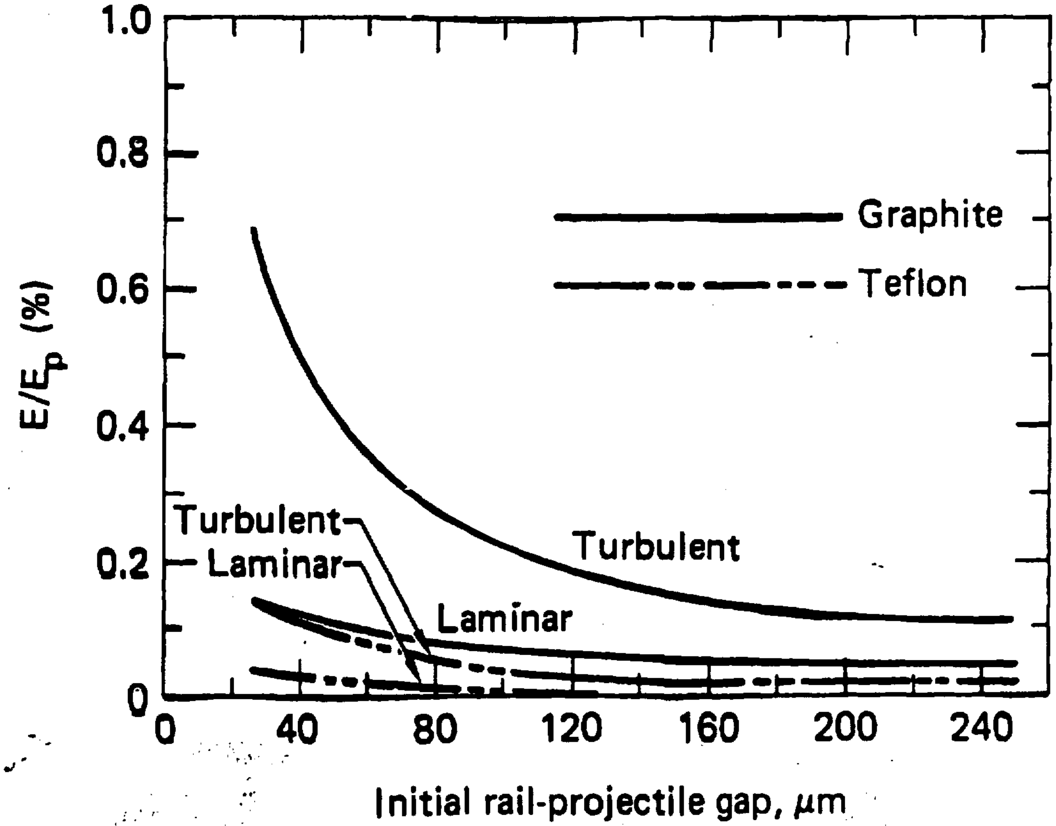
$$H_T \equiv h + \frac{u^2}{2} \qquad h \equiv \sum_i K_i h_i \qquad h_i \equiv \int_{T^0}^{T} c_{p_i}\, dT + \underline{h_i^0}$$

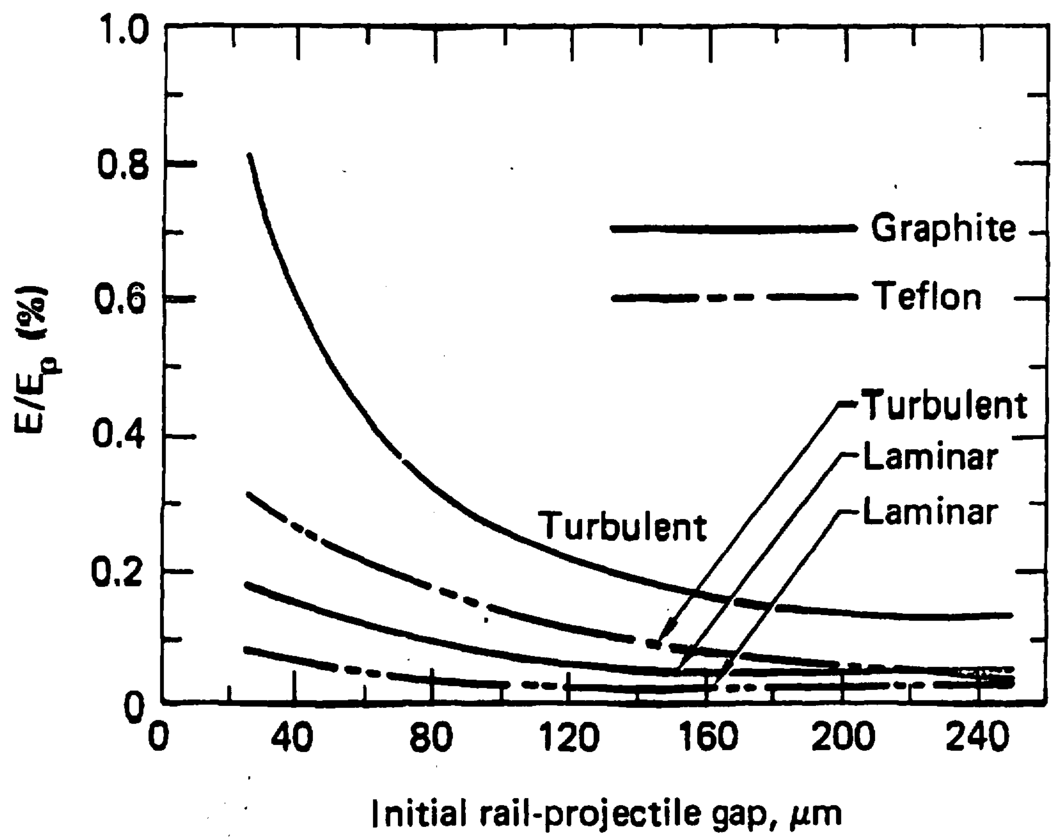

Projectile speed, km/s
Mass loss and drag energy ratio, %
E_D/E_P
m/m_o
$\vec{U}$
Time, tens of ns
1.0
0.8
0.6
0.4
0.2
0
100
80
60
40
20
0
0
2
4
6
8
10
12

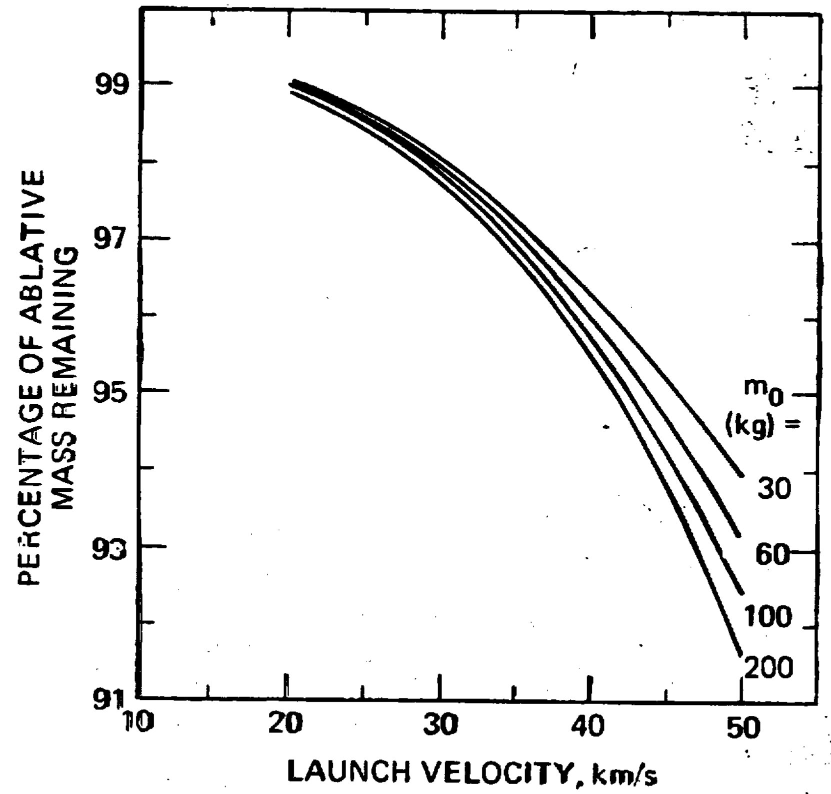

Projectile speed and distance traveled, designated units
Mass loss and drag energy ratio, %
Projectile speed, km/s
Distance, m
m/m_o
E/E_p
Time, ms
200
180
160
140
120
100
80
60
40
20
0
100
80
60
40
20
0
0
0.4
0.8
1.2
1.6
2.0
2.4

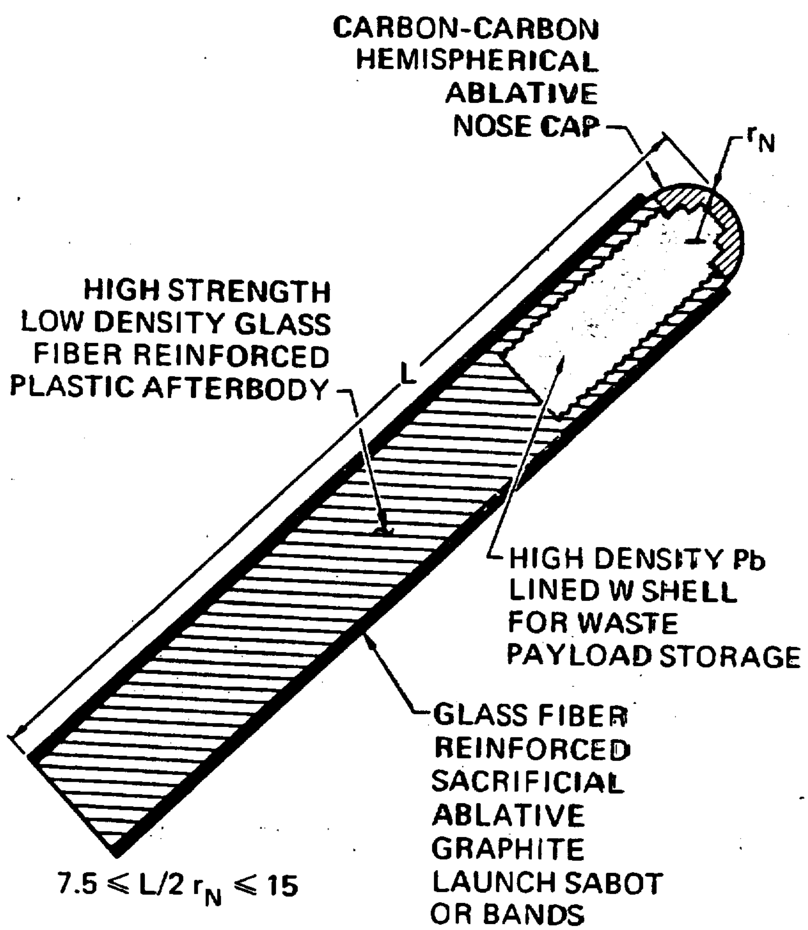

Projectile speed and distance traveled, designated units
Mass loss and drag energy ratio, %
Time, ms
Projectile speed, km/s
Distance, m
m/m$_o$
E/E$_p$

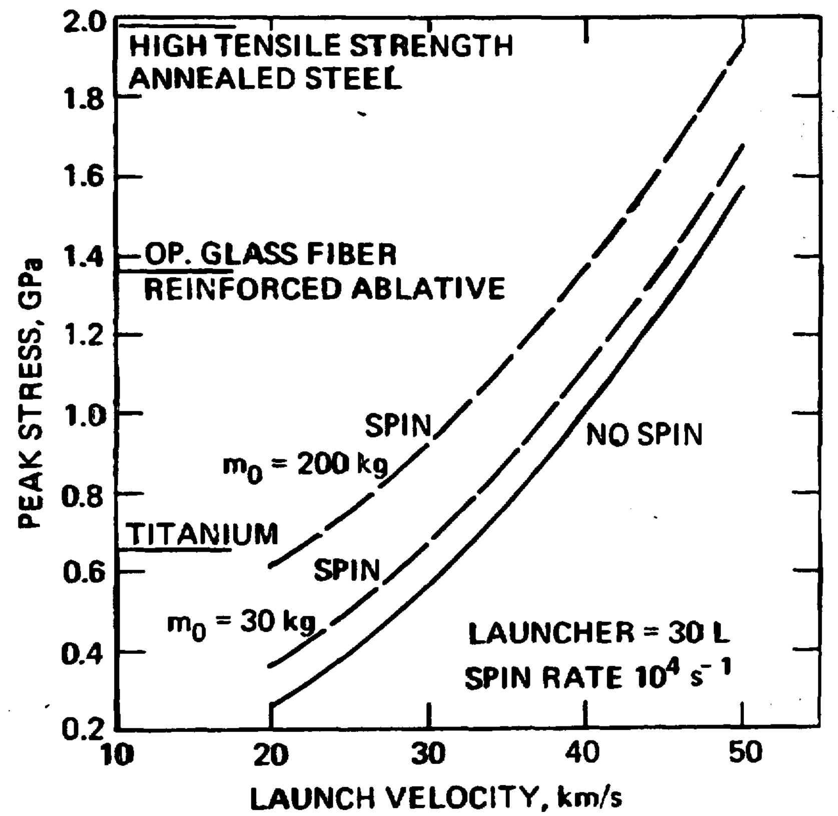

Projectile speed and distance traveled, designated units
Mass loss and drag energy ratio, %
Time, ms
Speed, km/s
Distance, m
m/m$_o$
E/E$_p$


1.0
0.8
0.6
0.4
0.2
0
E/E_p (%)
Graphite
Teflon
Turbulent
Laminar
Turbulent
Laminar
0
40
80
120
160
200
240
Initial rail-projectile gap, μm

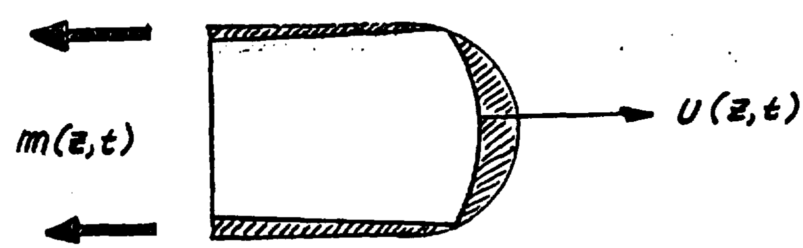

1.0
0.8
0.6
0.4
0.2
0
E/E_p (%)
Graphite
Teflon
Turbulent
Laminar
Laminar
Turbulent
0
40
80
120
160
200
240
Initial rail-projectile gap, μm


PERCENTAGE OF ABLATIVE MASS REMAINING
99
97
95
93
91
10
20
30
40
50
LAUNCH VELOCITY, km/s
m_0 (kg) =
30
60
100
200


CARBON-CARBON HEMISPHERICAL ABLATIVE NOSE CAP
r_N
L
HIGH STRENGTH LOW DENSITY GLASS FIBER REINFORCED PLASTIC AFTERBODY
HIGH DENSITY Pb LINED W SHELL FOR WASTE PAYLOAD STORAGE
GLASS FIBER REINFORCED SACRIFICIAL ABLATIVE GRAPHITE LAUNCH SABOT OR BANDS
$7.5 \leqslant L/2\,r_N \leqslant 15$



ABLATION/EROSION WITH FRICTION/FLUID DRAG

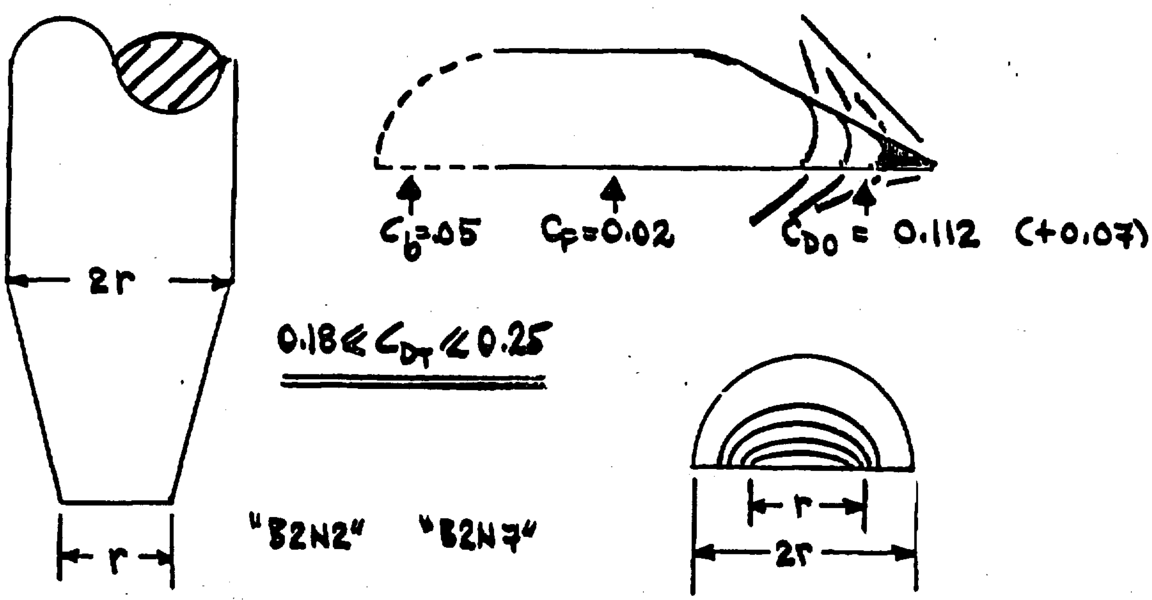


$$\frac{dU}{dt} = -\frac{C_D(U,z,t)\,\rho(z,t)\,U^2}{2\,m(z,t)}\left\{\pi A(z,t)\right\} - \frac{F(z)}{m(z,t)}$$

$$z = \int_{t_0}^{t} U(t')\,dt' + z(t_0)$$

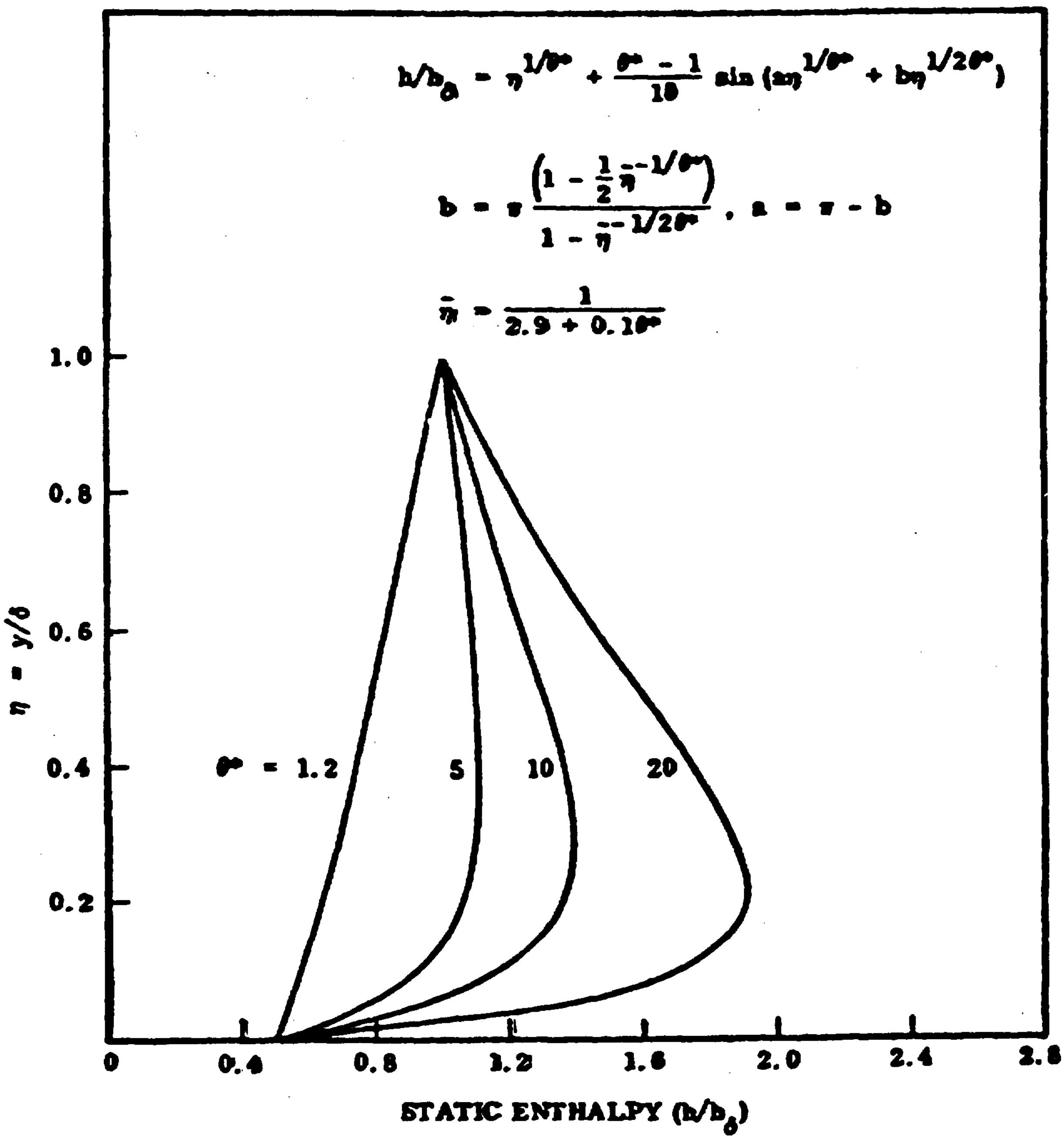
$$U = \int_{t_0}^{t} \dot{U}(t')\,dt' + \underset{\substack{\uparrow \\ \text{INJECTION}}}{U(t_0)}$$

LOWEST? DRAG



R.J. KRIEGER

MCDONNELL-DOUGLAS ~ 1978 (USAF)


$h/h_\delta = \eta^{1/\theta^*} + \frac{\theta^* - 1}{18} \sin(a\eta^{1/\theta^*} + b\eta^{1/2\theta^*})$
$b = \pi \frac{\left(1 - \frac{1}{2}\bar{\eta}^{-1/\theta^*}\right)}{1 - \bar{\eta}^{-1/2\theta^*}}, \quad a = \pi - b$
$\bar{\eta} = \frac{1}{2.9 + 0.1\theta^*}$
1.0
0.8
0.6
0.4
0.2
$\eta = y/\delta$
$\theta^* = 1.2$
5
10
20
0
0.4
0.8
1.2
1.6
2.0
2.4
2.8
STATIC ENTHALPY (h/h_δ)

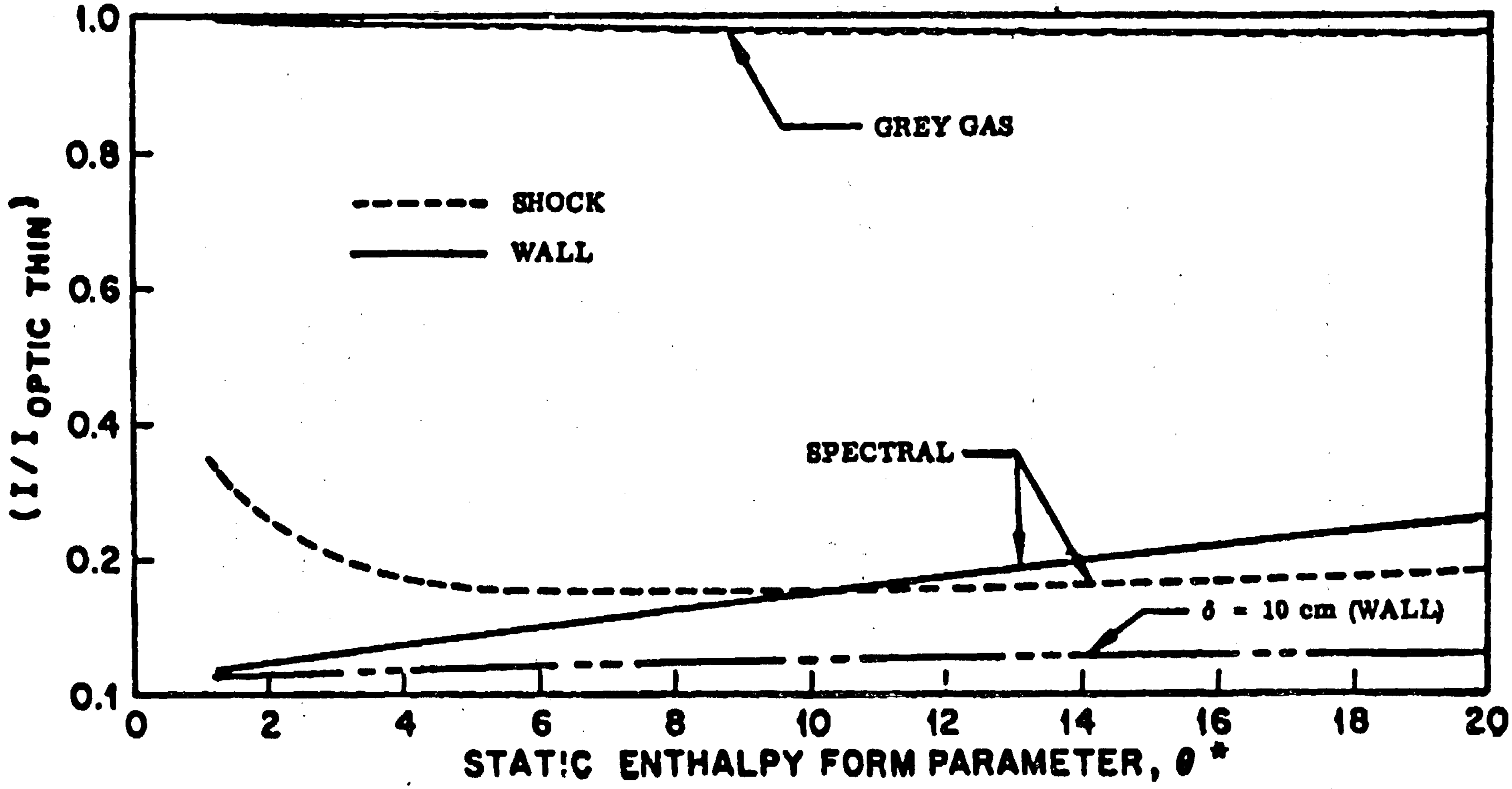
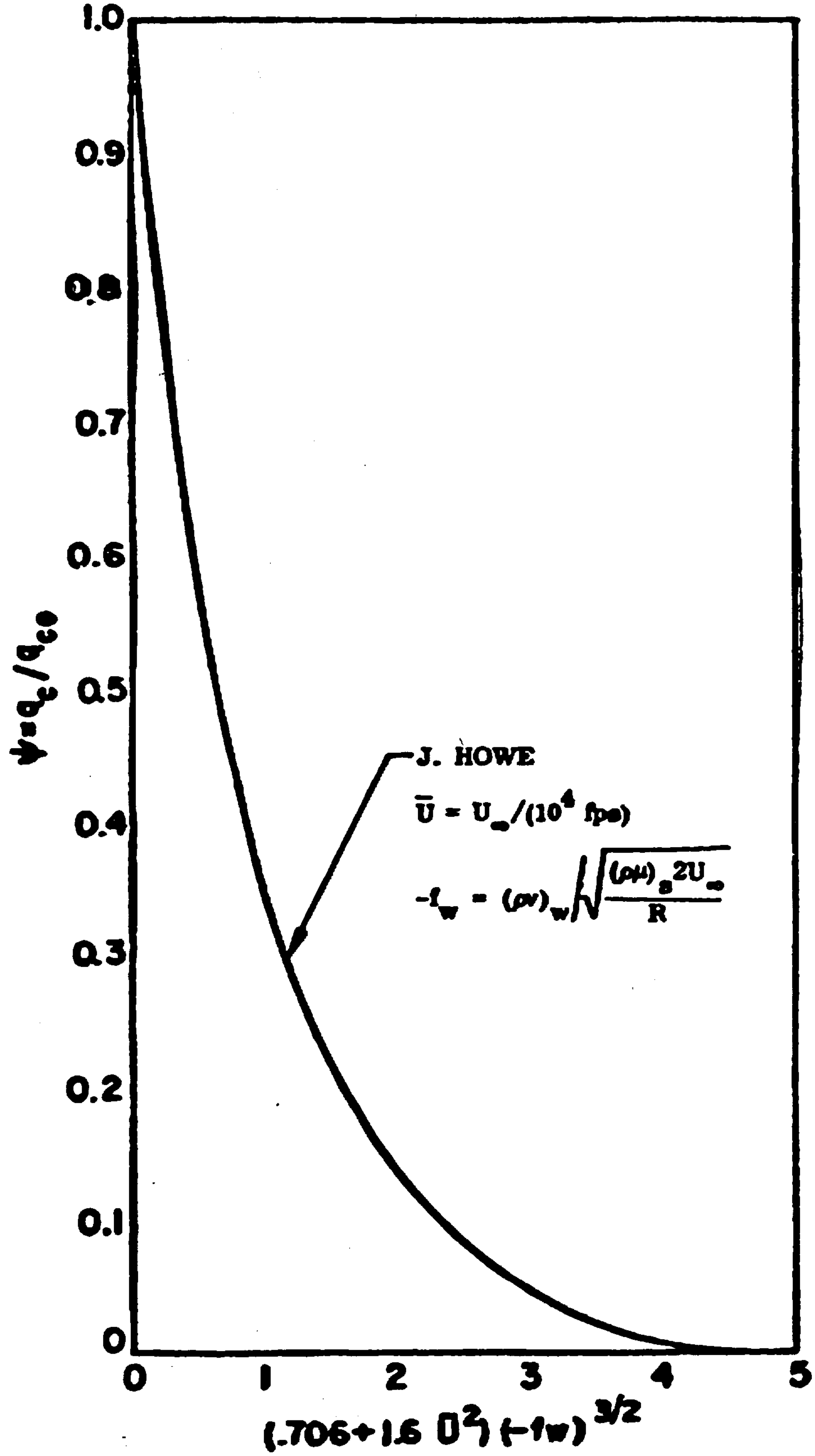


Fig. 41 Ratio of Radiative Flux With and Without Self-Absorption For a Slab of Gas 1 cm Thick, p = 1 atm Temperature Behind Shock = 11,400 °K, $\rho_\delta/\rho_e = 1.1 \times 10^{-2}$



Reduction of Convective Heating due to Mass Injection

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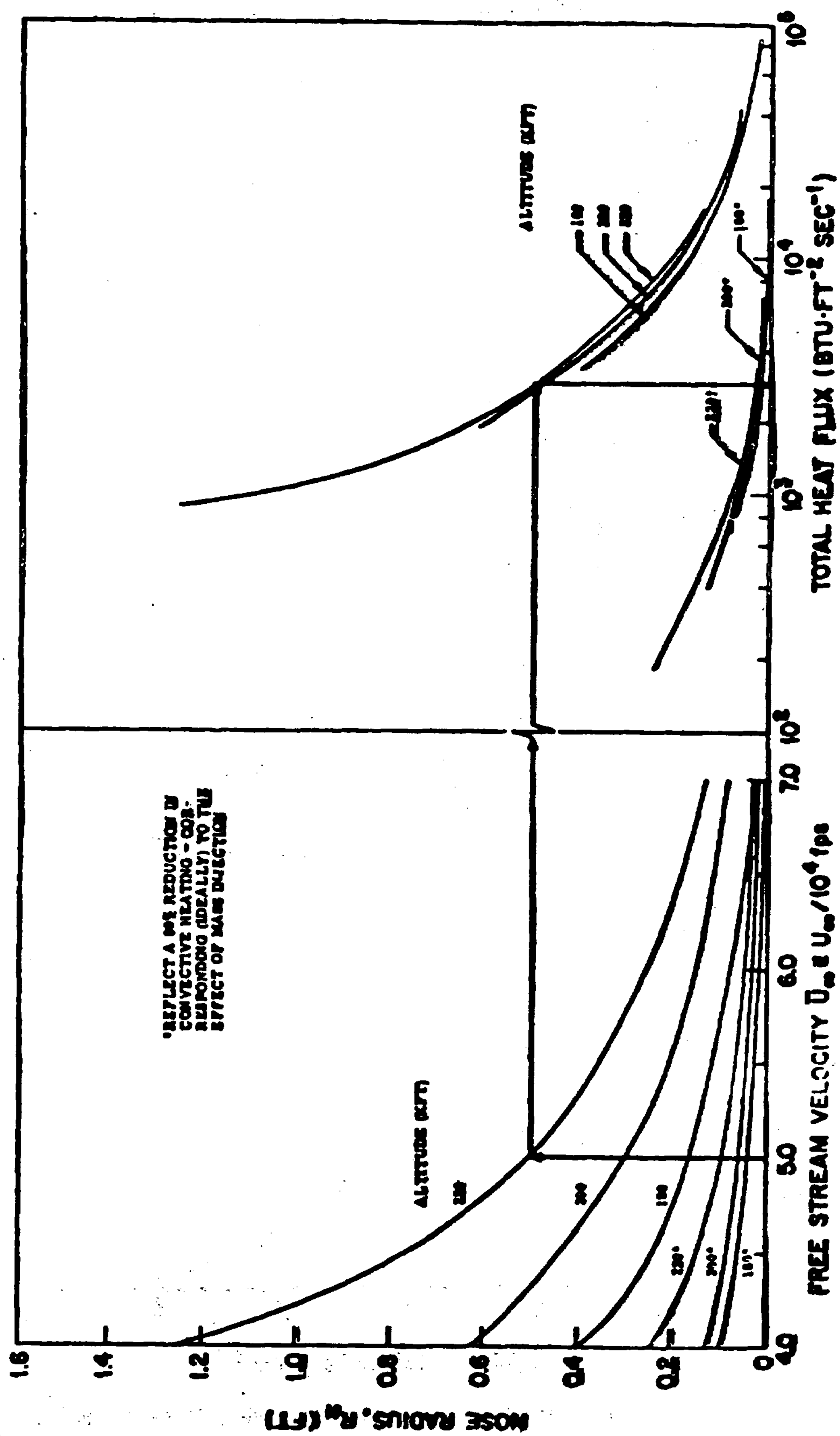
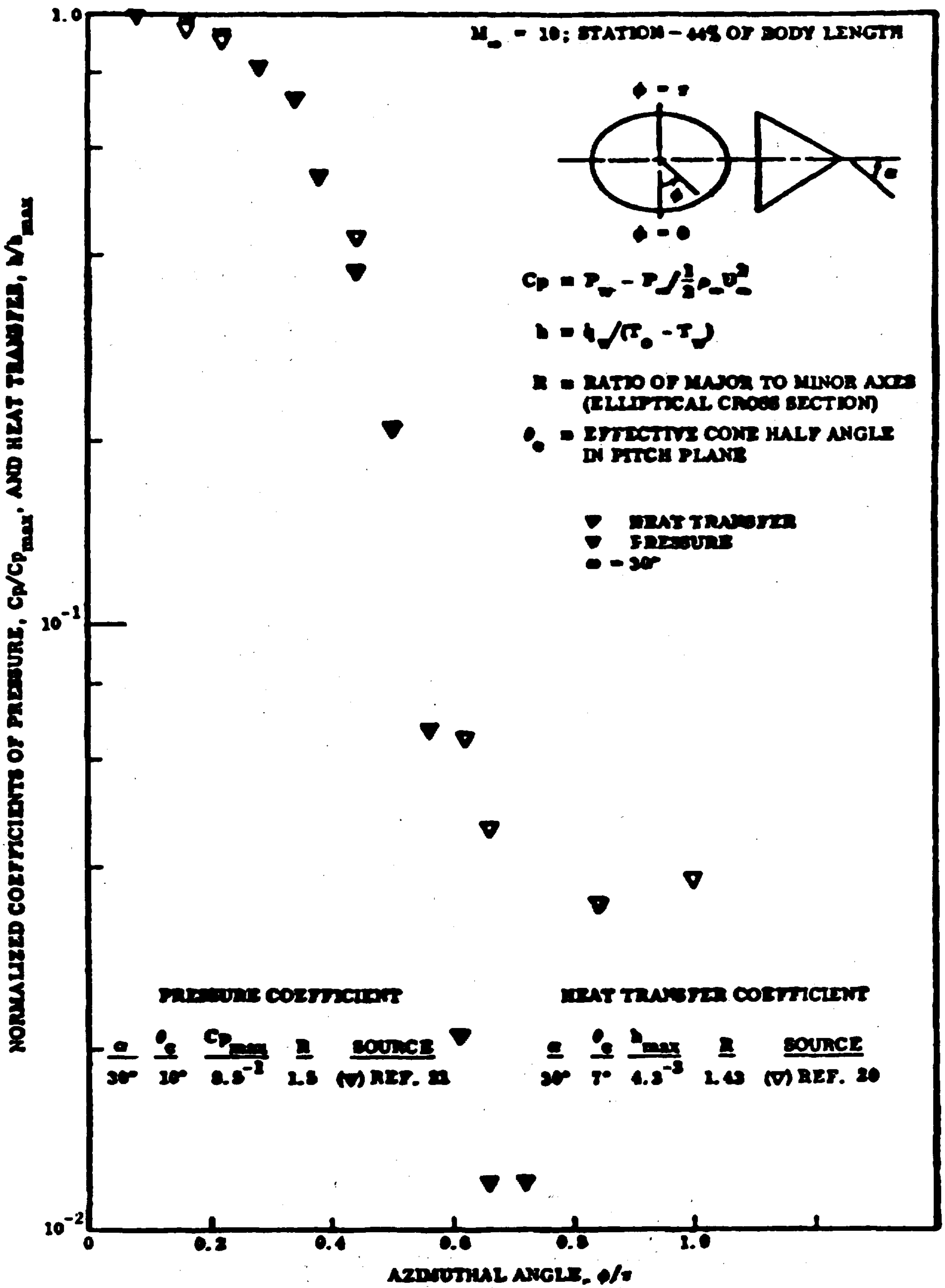


Spherical Nose Radius for Minimum (Stagnation Point) Total Heating



·zimuthal Distribution of Surface and Heat Transfer Over an Elliptical Cone
·a Angle of Attack

C-5

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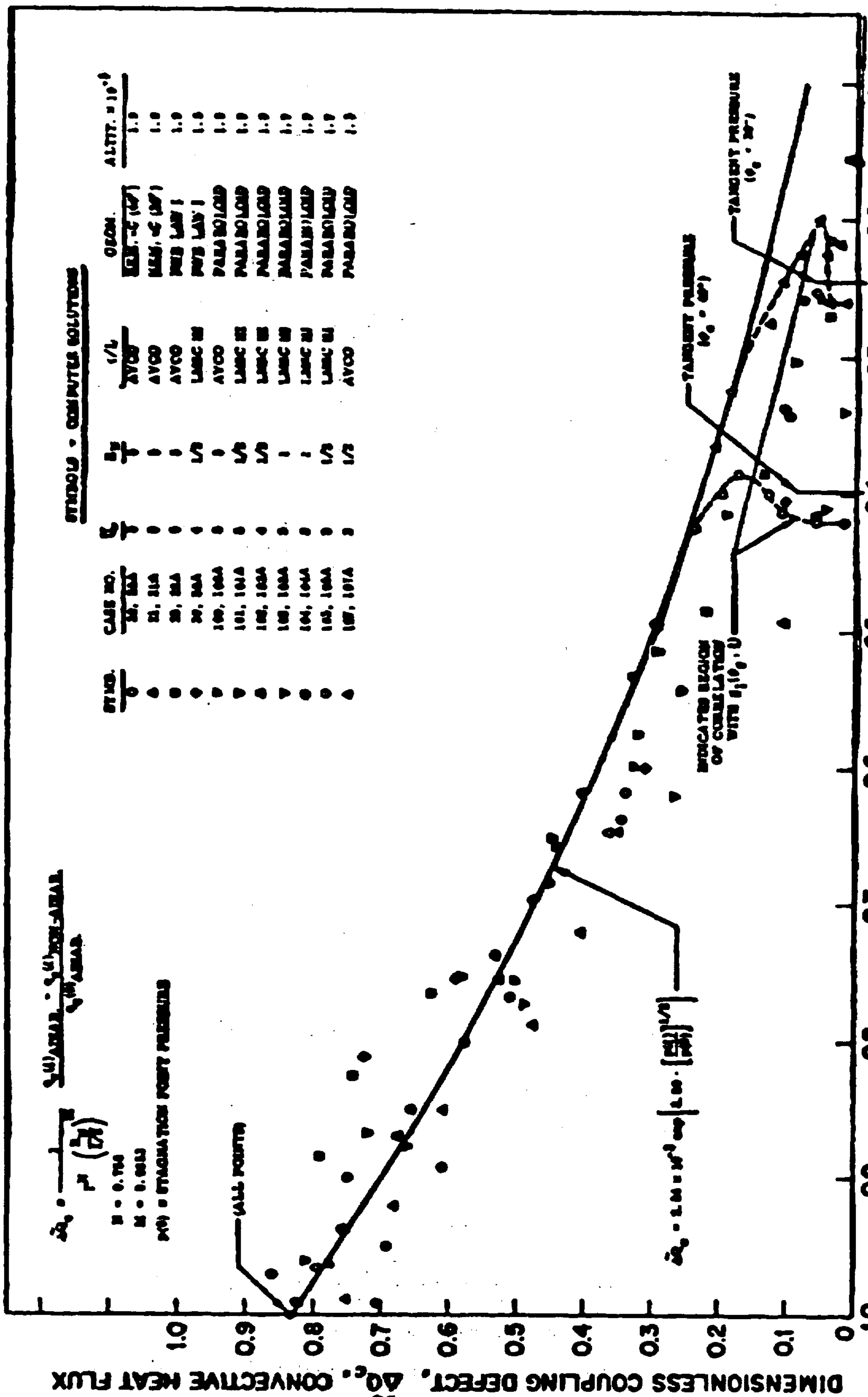

Correlation of Convective Heat Flux Coupling Defect With Pressure Distribution

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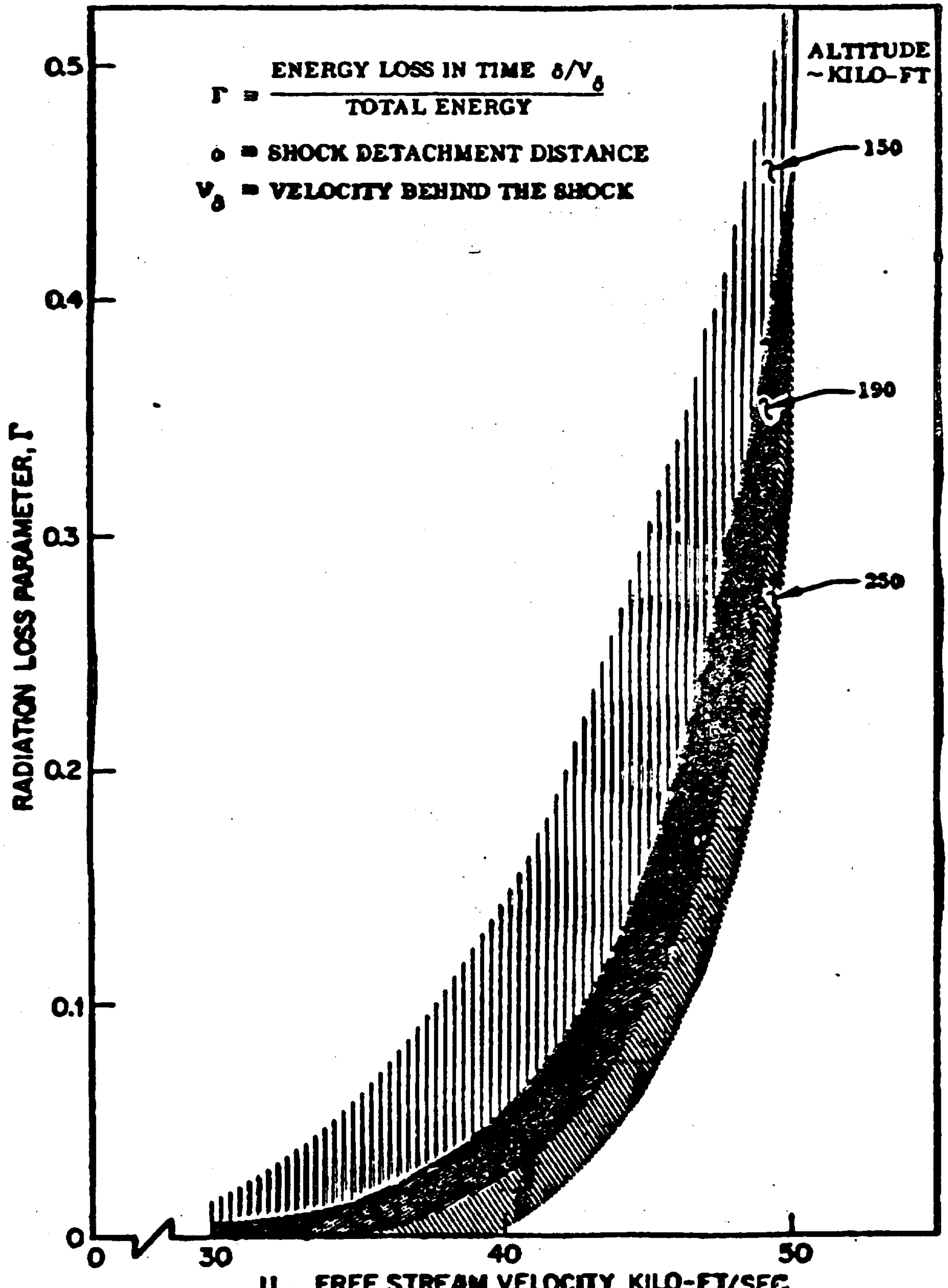

Variation in Radiation Loss Parameter Over Trajectory Range Analyzed

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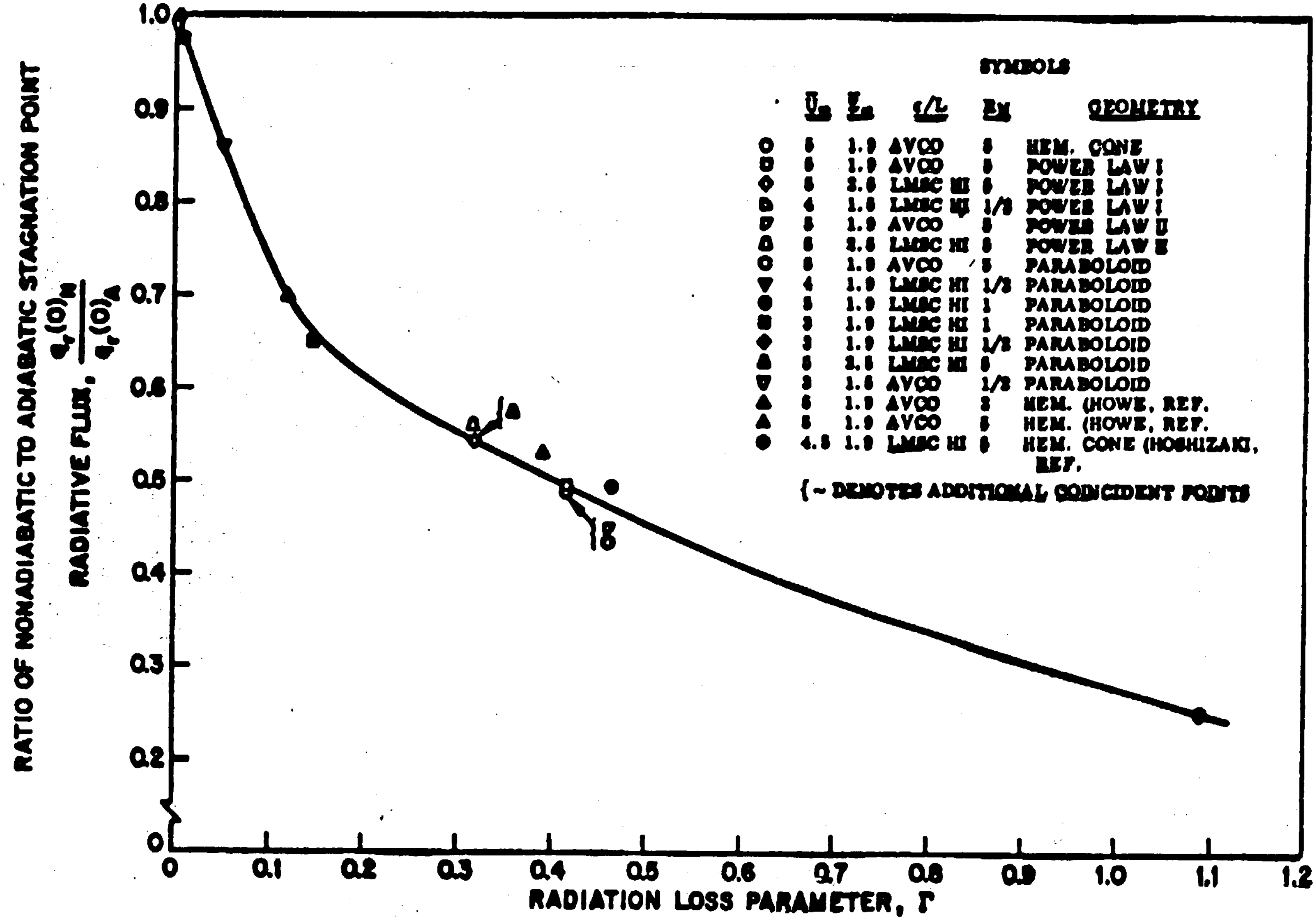
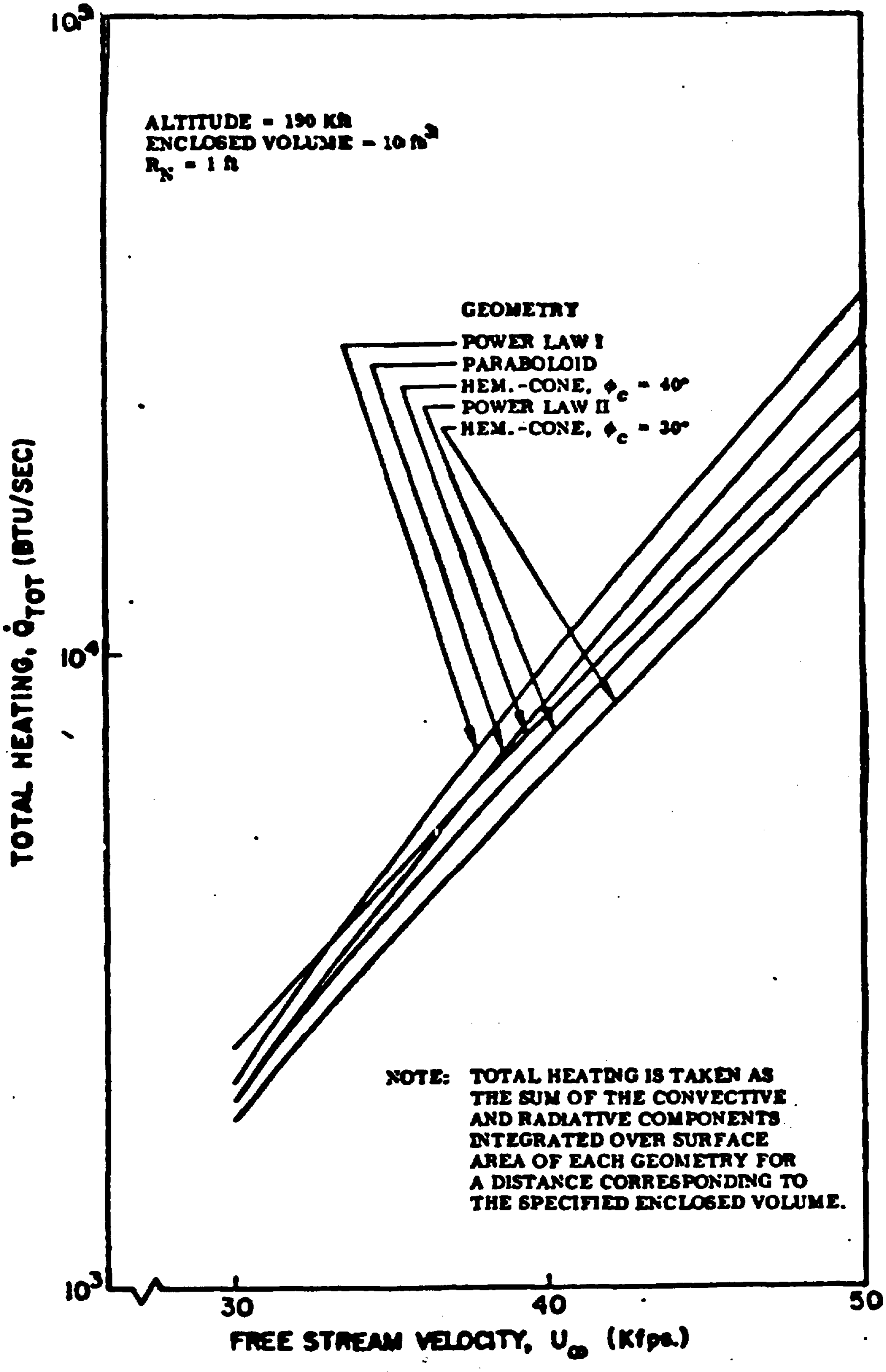


Reduction in Stagnation Point Radiative Heat Flux as a Function of Radiation Loss Parameter



Effect of Geometry on Total Heating for Fixed Volume

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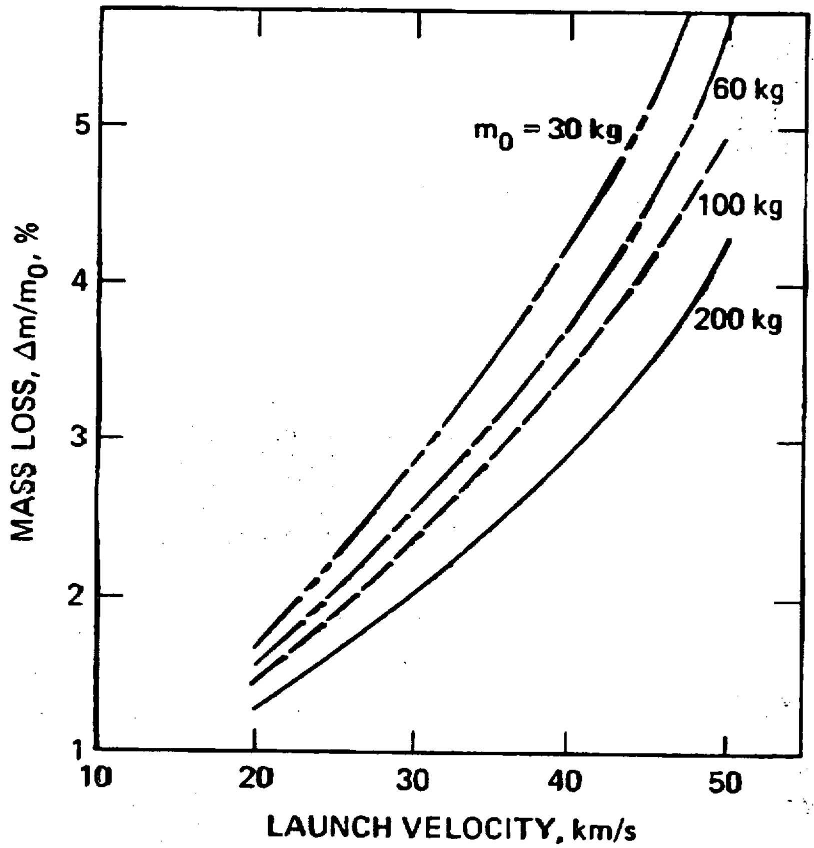
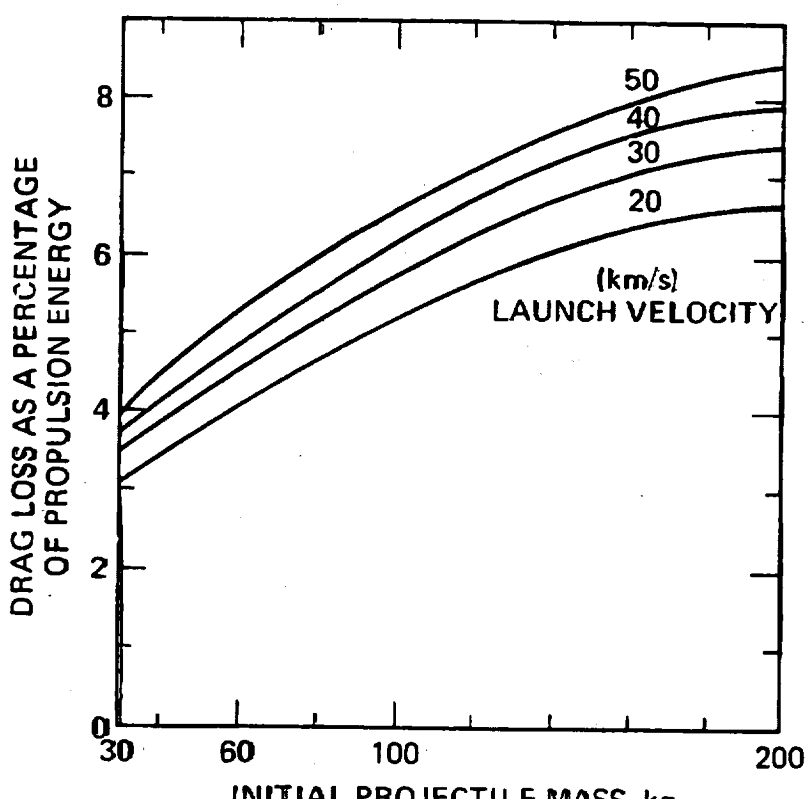
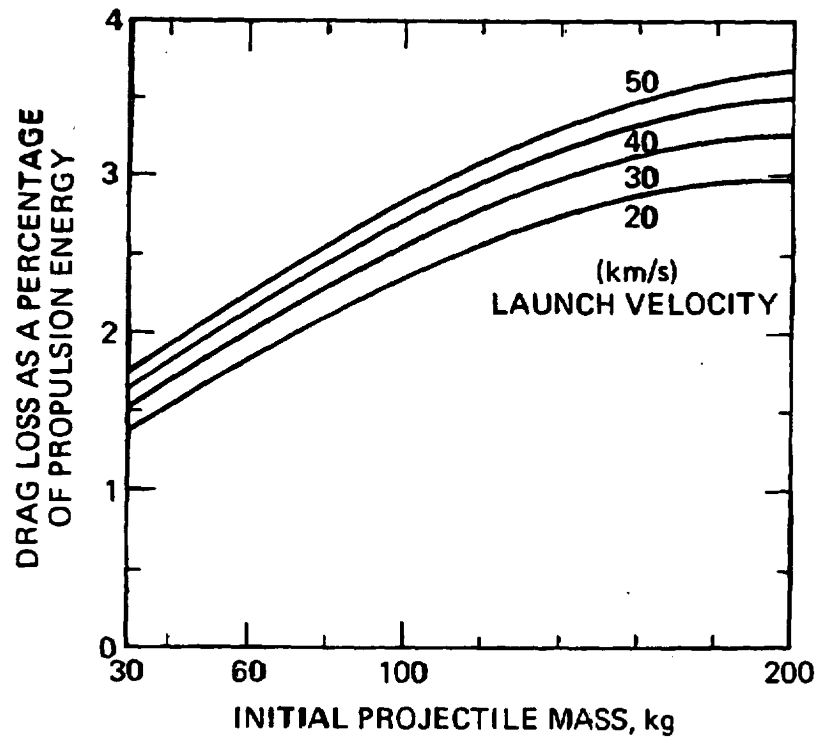
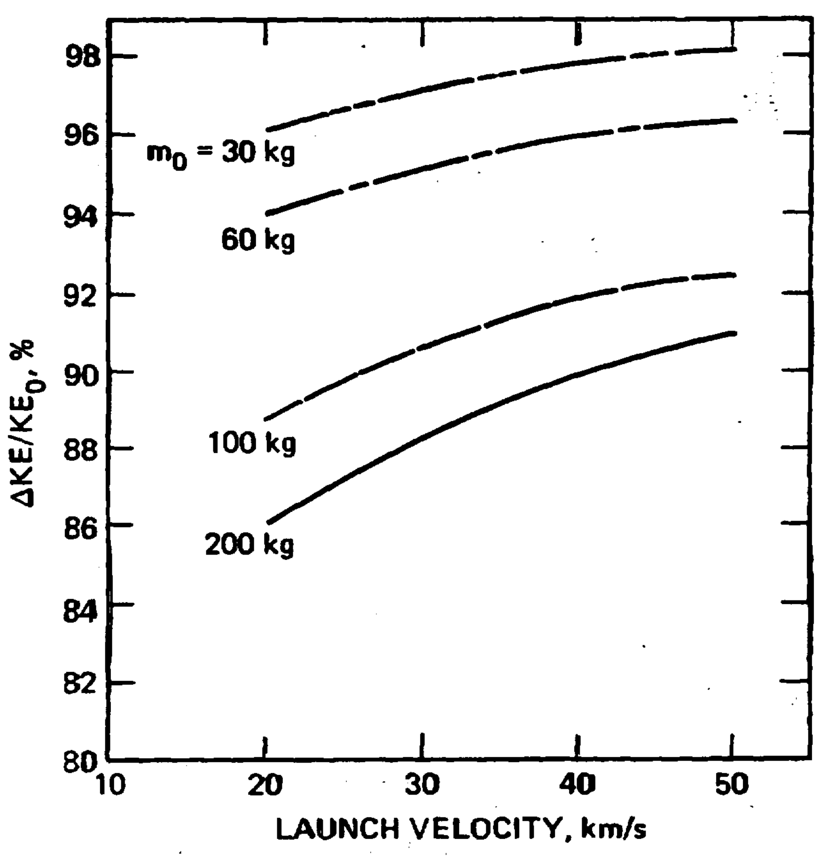
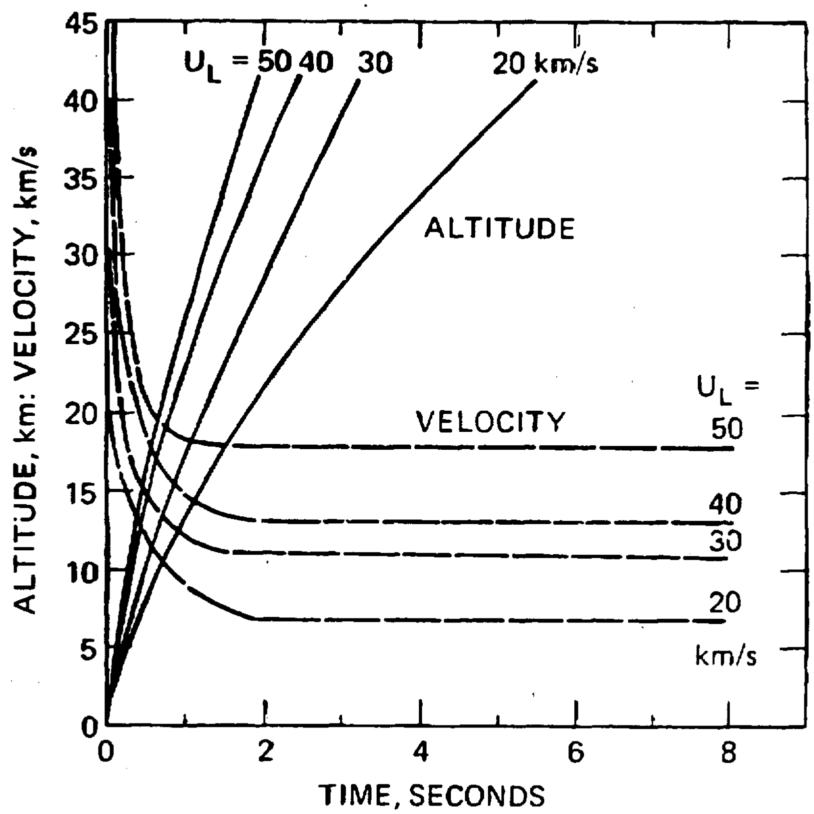
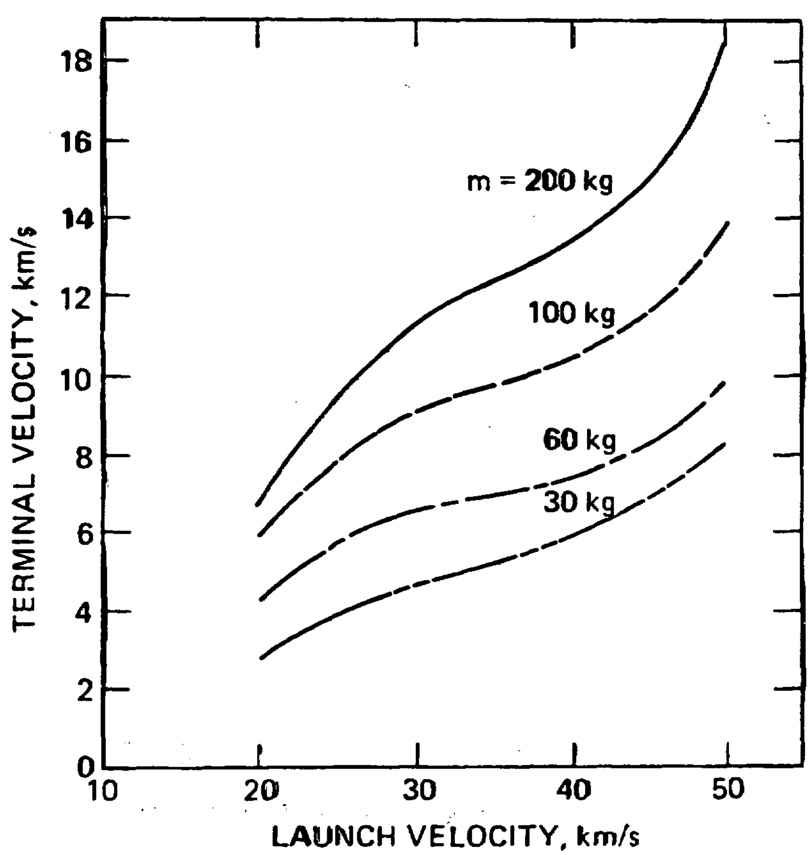


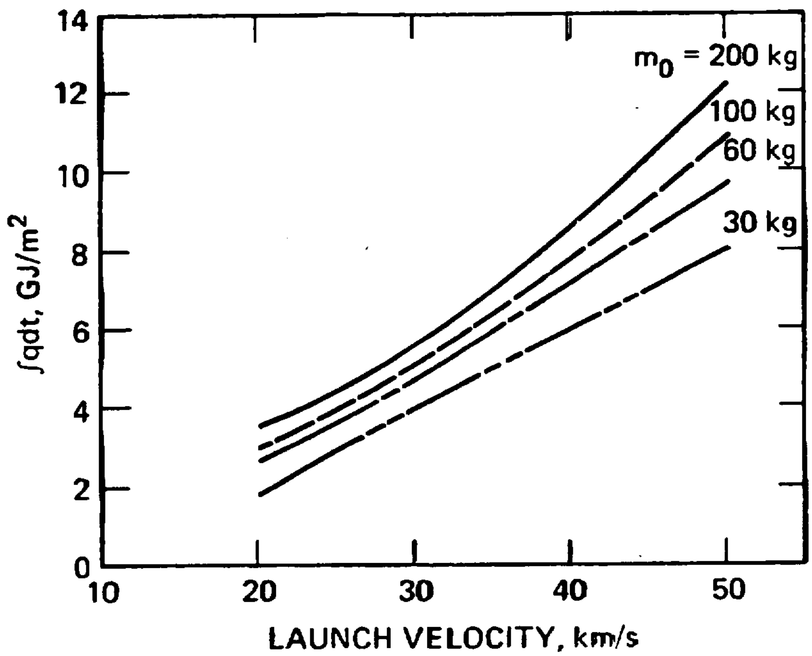



DRAG LOSS AS A PERCENTAGE OF PROPULSION ENERGY
0
1
2
3
4
30
60
100
200
INITIAL PROJECTILE MASS, kg
50
40
30
20
(km/s)
LAUNCH VELOCITY


ΔKE/KE0, %
80
82
84
86
88
90
92
94
96
98
10
20
30
40
50
LAUNCH VELOCITY, km/s
m0 = 30 kg
60 kg
100 kg
200 kg


U_L = 50 40 30 20 km/s
ALTITUDE
VELOCITY
U_L =
50
40
30
20
km/s
ALTITUDE, km: VELOCITY, km/s
TIME, SECONDS


m = 200 kg
100 kg
60 kg
30 kg
TERMINAL VELOCITY, km/s
LAUNCH VELOCITY, km/s



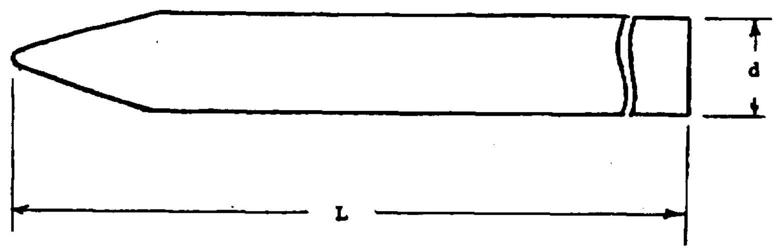
SUMMARY

- Launch: Injection, Ablative sabot, Bands, Couette Film
- Launch Stress, Spin penalty, Aerodynamic Loads
- Ablation & Erosion
  - High Drag, Large Mass, Lowered Ablation
  - Low Drag, Small Mass, High Ablation
- Current LLNL Launcher & Projectile Design
  - Materials Science
  - Launch Drag, Erosion, Stress
  - Trajectories, Scientific Potential, EOS

APPENDIX H

CONCEPT DEFINITION MEETING
(Selected Vugraphs by J. Lee, OSU)

Held at Battelle Columbus Laboratories
on
12-13 August 1981



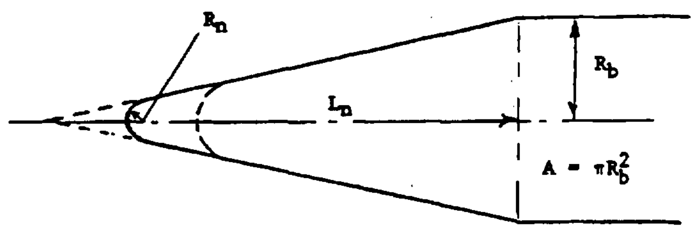


FIGURE 1. VEHICLE GEOMETRY

$$\frac{dV}{dt} - g - \frac{D}{m} = 0 \qquad (1)$$

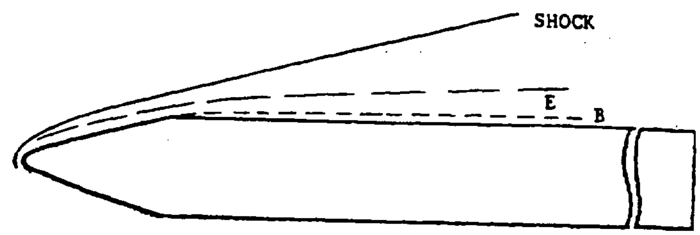
V = VELOCITY AT t
g = ACCN DUE TO GRAVITY (CONSTANT)
D = DRAG (TOTAL)
m = VEHICLE MASS

$$D = C_D \frac{\rho}{2} V^2 A \qquad (2)$$

$$\rho = \rho_o e^{-h/a} \quad \text{(ISOTHERMAL ATMOS.)} \qquad (3)$$

$$m = \rho_m L A \qquad (4)$$

C_D = DRAG COEF. (CONST.)
ρ = LOCAL DENSITY
ρ_o = SEA LEVEL DENSITY (CONST.)
A = VEHICLE CROSS SECTION AREA
L = VEHICLE LENGTH
ρ_m = MEAN VEHICLE DENSITY
h = ALTITUDE
a = CONSTANT (6.705 KM, 22000 FT.)
V_i = LAUNCH VELOCITY



E = ENTROPY LAYER
B = BOUNDARY LAYER

$$Re = Re_\infty\left(\frac{5}{M_\infty^2}(1.075M_\infty^{0.57} - 1)\right)^{1.75}$$

$$Re_\infty = 113 \times 10^6\ V_i\ e^{-h/a}$$

$$C_{D_f} = 3 \times 10^{-4}\ \frac{L/d\ \ e^{h/2a}}{\sqrt{V_i \cdot L}}$$

V_i in KM/SEC
L in METERS

$$2\frac{V}{V_i}\frac{d(V/V_i)}{d(h/a)} = -\frac{2ga}{V_i^2} - \left(\frac{V}{V_i}\right)^2 \frac{1}{B}\ e^{-(h/a)}$$

$$B = \frac{m}{C_D A \rho_o a} = \frac{\rho_m\ L}{\rho_o\ a}\ \frac{1}{C_D}$$

(PRIMARY VEHICLE PARAMETER)

NEGLECTING $\frac{2ga}{V_i^2}$:

$$\frac{V}{V_i} = e^{-\left(\frac{1 - e^{-h/a}}{2B}\right)}$$

$$\frac{V_e}{V_i} = e^{-\frac{1}{2B}}$$

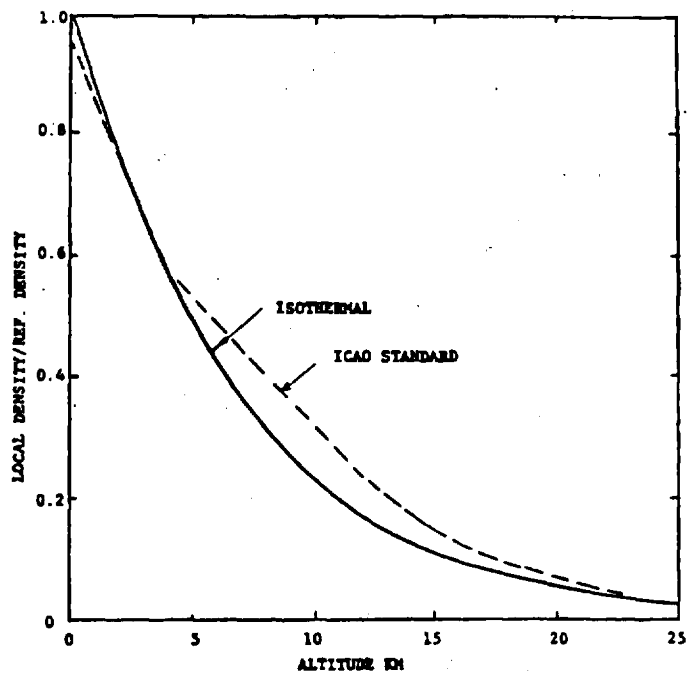


FIGURE 2. MODEL ISOTHERMAL ATMOSPHERE USED IN ANALYSES

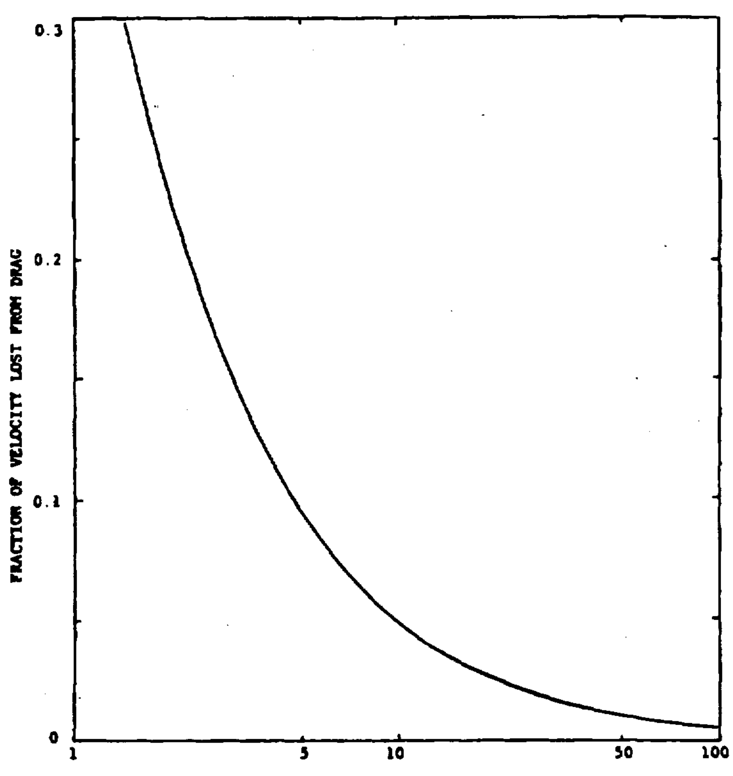


FIGURE 3. DRAG-LOSS ONLY IN TRANSATMOSPHERIC FLIGHT AS FRACTION OF LAUNCH VELOCITY

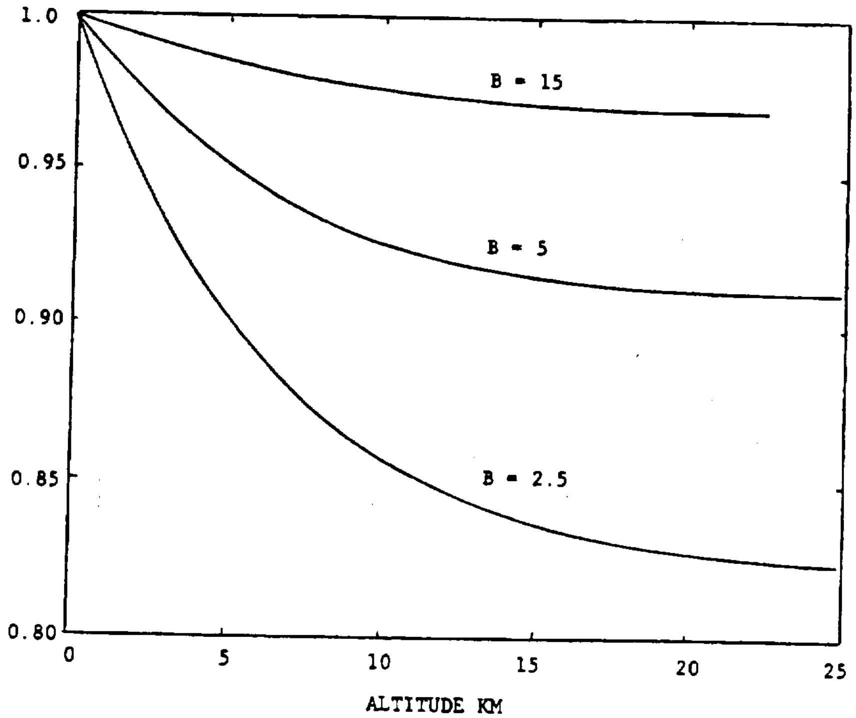


FIGURE 4. VELOCITY ALONG FLIGHT PATH FOR DIFFERENT VEHICLE PARAMETER VALUES

HEAT TRANSFER RATES
(AT NOSE STAGNATION POINT)

CONVECTIVE -

$$\dot{q}c = 2.72 \times 10^{-3} \frac{V^{4.5}}{\sqrt{r_n}} e^{-h/2a}$$

RADIATIVE -

$$\dot{q}r = 4.9 \times 10^{-5} r_n V^{8.5} e^{-h/a}$$

IN KCAL/SEC/M^2

r_n = NOSE-TIP RADIUS, METER

V = VEHICLE VELOCITY, KMPS

h = ALTITUDE, KM.

a = 6.705 KM.

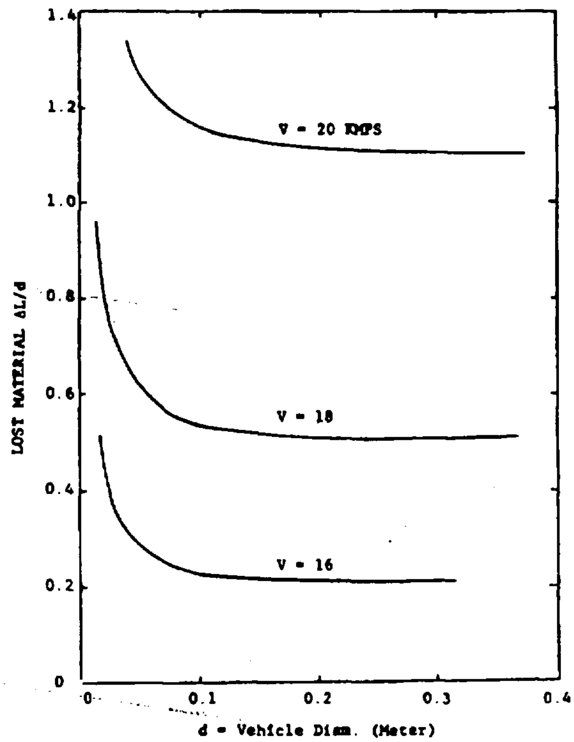


FIGURE 5. EXAMPLE CASE SHOWING MATERIAL LOST DURING FLIGHT THROUGH ATMOSPHERE

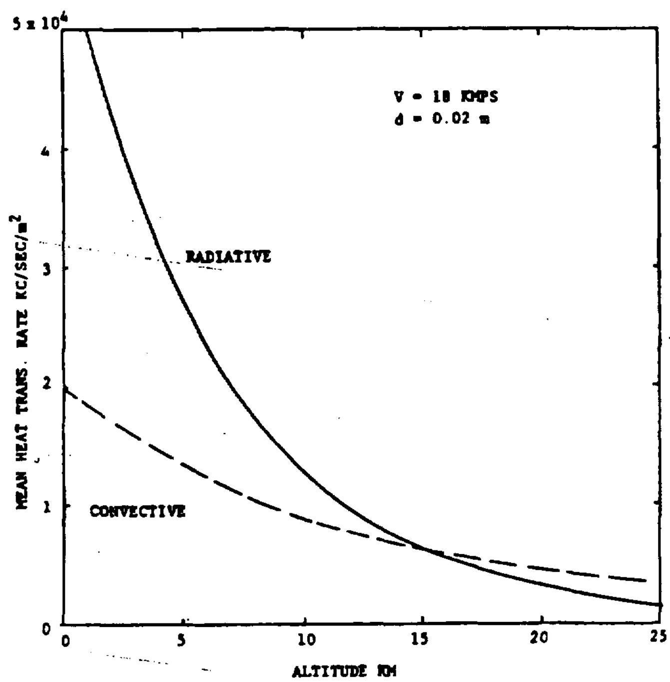


FIGURE 6. MEAN HEATING RATES TO NOSE CAP. V = 18 KMPS, d = 0.02 m.

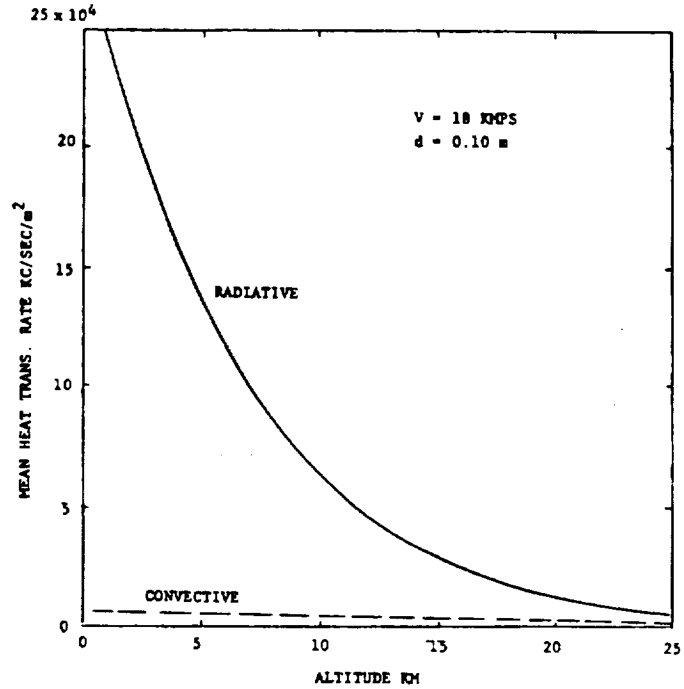


FIGURE 7. MEAN HEATING RATES TO NOSE CAP. V = 18 KMPS, d = 0.1 m.

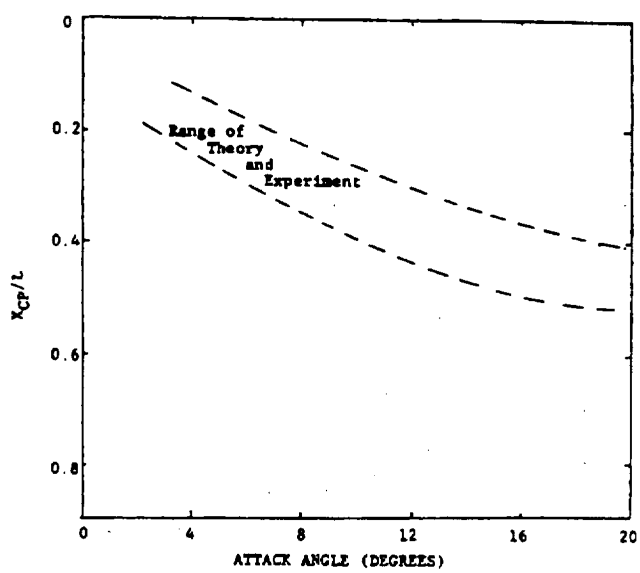


FIGURE 8. CENTER OF PRESSURE VARIATION FOR A SLENDER BODY

EXHAUST SHOCK STRENTGH
(VIA BLAST-WAVE THEORY)

$$\Delta p_S = \frac{0.082\ \rho_o\ V_1^2}{X^2} \cdot C_{D_T} \cdot D_T^2$$

X = DISTANCE FROM SITE

C_{D_T} = TOTAL VEHICLE DRAG COEF.

D_T = MEAN DIAM. OF TOTAL VEHICLE

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