12-1995

Systems Analysis of GPS Electrical Power System Redesign

Kevin J. Walker

SYSTEMS ANALYSIS OF GPS ELECTRICAL POWER SYSTEM REDESIGN

THESIS

Kevin J. Walker, Captain, USAF

AFIT/GSO/ENY/95D-03

DEPARTMENT OF THE AIR FORCE
AIR UNIVERSITY
AIR FORCE INSTITUTE OF TECHNOLOGY

Wright-Patterson Air Force Base, Ohio

SYSTEMS ANALYSIS OF GPS ELECTRICAL POWER
SYSTEM REDESIGN

THESIS

Kevin J. Walker, Captain, USAF

AFIT/GSO/ENY/95D-03

The views expressed in this thesis are those of the author and do not reflect the official policy or position of the Department of Defense or the U. S. Government.

AFIT/GSO/ENY/95D-03

SYSTEMS ANALYSIS OF GPS ELECTRICAL POWER SYSTEM REDESIGN

THESIS

Presented to the Faculty of the Graduate School of Engineering

Air Education and Training Command

In Partial Fulfillment of the Requirements for the Degree of

Masters of Science in Space Operations

Kevin J. Walker, B.S.

Captain, USAF

December 1995

Acknowledgments

I would like to extend a special thank you to all the faculty, friends, and family who supported me throughout the trials and tribulations of graduate school. A special thank you goes out to Dr. Chris Hall, my thesis advisor, for all of the insight and assistance he gave me in technical areas and writing skills.

Additional recognition should go to Lieutenant Colonel Bowman for helping me find detailed information on alternative energy sources and conversion techniques. Without this aid, my research would never have gotten off the ground.

Last, but not least, I would like to thank my wife, Colleen, for her understanding and support through the many long nights and weekends it took to complete this project.

Table of Contents

ACKNOWLEDGMENTS .... **ii**

LIST OF FIGURES .... **vi**

LIST OF TABLES .... **vii**

LIST OF TERMS .... **viii**

ABSTRACT .... **x**

I. INTRODUCTION **1**

1.1 BACKGROUND 1

1.2 STATEMENT OF PROBLEM 2

1.3 OBJECTIVES 3

1.4 SCOPE OF ANALYSIS 3

1.5 REPORT OUTLINE 4

II. LITERATURE REVIEW **5**

2.1 GPS HISTORY 5

2.2 GPS BLOCK IIR CURRENT CAPABILITIES 7

2.3 ELECTRIC POWER SYSTEM TECHNOLOGIES 7

2.3.1 GPS Electrical Power Subsystem *8*

2.3.1.1 Solar Arrays 8

2.3.1.2 Energy Storage 10

2.3.1.3 Power Control and Distribution 10

2.3.1.4 Thermal Control 10

2.3.2 Current Power System Technologies *11*

2.3.2.1 Power Sources 11

2.3.2.2 Power Conversion Systems 12

2.3.2.3 Energy Storage Systems 13

2.3.2.4 Thermal Control 14

III. SYSTEMS ANALYSIS 16

3.1 INTRODUCTION 16

3.2 RESULTS 17

3.2.1 Problem Definition 17

3.2.1.1 Needs 18

3.2.1.2 Alterables 19

3.2.1.3 Constraints 19

3.2.1.4 Actors 20

3.2.1.5 Subjective Factors 20

3.2.1.6 System Boundaries 21

3.2.2 Value System Design 21

3.2.3 System Synthesis 23

3.2.4 System Modeling 24

3.2.5 System Evaluation 24

3.2.5.1 Dynamic Conversion Evaluation 25

3.2.5.2 Direct Conversion Evaluation 28

3.2.5.3 Nuclear Power Comments 29

IV. DESIGN 30

4.1 INTRODUCTION 31

4.2 DESIGN #1 32

4.2.1 Concentrator 32

4.2.2 Receiver/Energy Storage 34

4.2.3 Power Converter 36

4.2.4 Radiator 38

4.2.5 Summary *40*

4.3 DESIGN #2 42

4.3.1 Concentrator *43*

4.3.2 Receiver/Energy Storage *44*

4.3.3 Power Converter *44*

4.3.4 Radiator *46*

4.3.5 Summary *47*

4.4 COMPARISON WITH GPS IIR 47

4.4.1 Required AMTEC Efficiency *51*

4.5 BOOM LENGTH CALCULATIONS 54

V. CONCLUSION 58

5.1 SUMMARY 58

5.2 RECOMMENDATIONS 60

APPENDIX: POWER CONVERSION DEVICES 61

A.1 DYNAMIC CONVERSION SYSTEMS 61

A.1.1 Carnot Cycle *61*

A.1.2 Brayton Cycle *62*

A.1.3 Rankine Cycle *64*

A.1.4 Stirling Engine *65*

A.2 DIRECT ENERGY CONVERSION SYSTEMS 66

A.2.1 Thermoelectric System *66*

A.2.1.1 AMTEC 68

A.2.2 Thermionic System *70*

A.2.3 Magnetohydrodynamic System *71*

BIBLIOGRAPHY 72

VITA 74

List of Figures

FIGURE 1 - EPS SIMPLIFIED BLOCK DIAGRAM 9

FIGURE 2 - OBJECTIVE HIERARCHY FOR GPS ELECTRICAL POWER SYSTEM REDESIGN 22

FIGURE 3 - SOLAR THERMAL ELECTRIC POWER SYSTEM DIAGRAM 30

FIGURE 4 - EPS BOOM CONFIGURATION 54

FIGURE 5 - BOOM ANGLE VS. BOOM LENGTH 56

FIGURE 6 - T-S AND P-V DIAGRAMS FOR CARNOT CYCLE 62

FIGURE 7 - SCHEMATIC AND T-S DIAGRAM FOR IDEAL CLOSED BRAYTON CYCLE 63

FIGURE 8 - SCHEMATIC AND T-S DIAGRAM FOR IDEAL RANKINE CYCLE 64

FIGURE 9 - P-V AND T-S DIAGRAMS FOR IDEAL STIRLING CYCLE 65

FIGURE 10 - OPERATING PRINCIPLES OF FREE PISTON STIRLING ENGINE 67

FIGURE 11 - OPERATING PRINCIPLE OF THERMOELECTRIC CONVERTER 68

FIGURE 12 - AMTEC CYCLE DIAGRAM 69

FIGURE 13 - THERMIONIC CONVERTER, PARAMETERS, & MATERIALS 70

FIGURE 14 - SCHEMATIC OF MAGNETOHYDRODYNAMIC GENERATOR 71

List of Tables

TABLE 1 - PRELIMINARY EVALUATION OF POWER SYSTEMS 26

TABLE 2 - DYNAMIC CONVERSION EVALUATION RESULTS 27

TABLE 3 - SYSTEM EFFICIENCIES AND MULTIPLIERS FOR SOLAR DYNAMIC MODEL 33

TABLE 4 - CONCENTRATOR AREA CALCULATIONS FOR SOLAR DYNAMIC MODEL 34

TABLE 5 - CONCENTRATOR COMPONENTS AND MASSES FOR SOLAR DYNAMIC MODEL 35

TABLE 6 - RECEIVER & ENERGY STORAGE CALCULATIONS FOR SOLAR DYNAMIC MODEL 37

TABLE 7 - CALCULATED COMPONENT MASSES AND AREAS FOR SOLAR DYNAMIC MODEL DESIGN 41

TABLE 8 - SYSTEM EFFICIENCIES & MULTIPLIERS FOR SOLAR DIRECT MODEL 42

TABLE 9 - CONCENTRATOR AREA CALCULATIONS FOR SOLAR DIRECT MODEL 43

TABLE 10 - CONCENTRATOR COMPONENTS AND MASSES FOR SOLAR DIRECT MODEL 44

TABLE 11 - RECEIVER & ENERGY STORAGE CALCULATIONS FOR SOLAR DIRECT MODEL 45

TABLE 12 - CALCULATED COMPONENT MASSES AND AREAS FOR SOLAR DIRECT MODEL DESIGN 48

TABLE 13 - DESIGN MASS AND AREA COMPARISONS 49

TABLE 14 - SYSTEM MASS AND AREA SUMMARIZATION 53

TABLE 15 - BOOM LENGTHS AND ANGLES 57

List of Terms

AFMC- Air Force Materiel Command

AFSPC - Air Force Space Command

AH - Amp-Hours

AKM - Apogee Kick Motor

AMTEC - Alkali Metal Thermal-to-Electric Converter

BASE - β''-Alumina Solid Electrolyte

CBC - Closed Brayton Cycle

CMD - Command

CSS - Coarse Sun Sensor

DEC - Direct Energy Conversion

DOD - Depth of Discharge

EMI - Electromagnetic Interference

EPA - Environmental Protection Agency

EPS - Electric Power System

FPSE - Free-Piston Stirling Engine

GPS - Global Positioning System

GPS IIR - GPS Block IIR

HP FPSE - Heat Pipe FPSE

IPACS - Integrated Power and Attitude Control System

J - Joules

K - Kelvin

kg - Kilograms

kJ - Kilojoules

kW - Kilowatts

kW_e - Kilowatts Electric (Electrical Energy)

kW_t- Kilowatts Thermal (Thermal Energy)

m - Meters

MHD - Magnetohydrodynamic

NASA - National Aeronautical and Space Administration

Navstar - Navigation System using Timing and Ranging

OMB - Office of Management and Budget

PCM - Phase Change Material

PCU - Power Conversion Unit

PL FPSE - Pumped Loop FPSE

PRU - Power Regulation Unit

Q_{IN} - Heat into System

Q_{OUT} - Heat out of System

S/A - Solar Array

SAD - Solar Array Drive

SAIC - Science Applications International Corporation

SV - Space Vehicle

TCF - Thermal Control Function

T_H - High Temperature

T_L - Low Temperature

TLM - Telemetry

TT&C - Telemetry, Tracking, and Commanding

USAF - US Air Force

USSPC - US Space Command

V - Volts

W - Watts

W_{IN} - Work into System

W_{OUT} - Work out of System

AFIT/GSO/ENY/95D-03

Abstract

A systematic analysis is applied to the electrical power subsystem of the Global Positioning Satellite (GPS) system. Results determined the most appropriate power source and conversion system options. Photovoltaic solar arrays, the current power system, were not included in the analysis. The best electrical power subsystem options found in the analysis include a solar power source with either a dynamic or direct conversion technique, and a direct conversion nuclear source. The two solar options are designed, at a low level of detail, to provide the same level of power the current GPS photovoltaic solar array system provides. These two designs are then compared with the current system, stressing mass and area. Results show that the solar dynamic design has approximately 28% less mass and approximately 35% less area than the GPS IIR design. The solar direct model has approximately 38% more mass and 72% more area than the GPS design.

SYSTEMS ANALYSIS OF GPS ELECTRICAL POWER SYSTEM REDESIGN

I. Introduction

The Global Positioning System (GPS) satellites are highly sophisticated, navigational tools in orbit about the Earth. Like most other spacecraft, GPS satellites require a constant source of electrical power to provide continuous operations. This thesis presents a systems engineering analysis of the GPS power system, with a view towards placing an improved system on vehicles developed in the future. This chapter begins with some background information, contains a detailed problem statement, and concludes with an outline of the thesis.

1.1 Background

All space vehicles require light-weight, on-board, self-contained power generating equipment to energize various electronic components, scientific instruments, and other apparatus. Space vehicles have imposed stringent and sometimes opposing requirements upon space power generators, namely, light weight, long life, and high reliability (1:vii).

One can not understate the importance of electrical power to a satellite. "With few exceptions, all space vehicles without electrical power become useless debris" (2:145). As the world continues to exploit the space frontier, new technologies and advancements continuously change the state of the art. Photovoltaic solar cells with supplementary batteries have been the mainstay of electric power generation in space. Nuclear reactors, radioisotopes, and fuel cells make up the predominant exceptions.

New and exciting research in space power production is building on scientific principles proven as much as thirty years ago. Increased efficiencies, decreased masses, improved control, and other factors all combine to make power generation and control in space smaller, lighter, more efficient, and longer lived. Reducing the mass and increasing the efficiency and lifetime of a satellite system should reduce the cost of operation of the system. Lower mass means launching on a smaller booster or using less fuel, either of which would save money. Increased efficiency and lifetime equate to fewer replacements over time, which results in lower life cycle costs.

Improvements in any one of the many satellite subsystems should help keep the skyrocketing costs of operating a satellite system in check. This report describes the systems engineering analysis of the electrical power subsystem of the Navstar GPS satellite with the intent of defining a potentially improved system.

1.2 Statement of Problem

Perform a systems engineering analysis on the redesign of the electrical power subsystem of the Navstar Global Positioning System satellites emphasizing alternative power systems.

1.3 Objectives

The following objectives to the problem statement were derived.

- Compare different power system's characteristics to identify best and worst options.
- Design best options to determine vital characteristics.
- Compare designs with currently used system.

Whenever possible, during the design phase of work, the most advanced system or technique considered feasible was used. If a system or technique has undergone and passed testing and a model design has been successfully created and tested, it is considered feasible for the purposes of this study. Most of these systems have not, however, been approved for space flight yet, but do meet all space flight requirements. Hypothetical scenarios, computer models, and other underdeveloped ideas were not considered.

1.4 Scope of Analysis

In order to maximize the impact of this work, the analysis was limited to a first level comparative study. The various power sources and conversion techniques are compared to one another, determining the best and worst options. The best options are roughly designed in order to determine vital parameters such as mass and area. These options are then compared with the current Navstar system. The goal of this work is to show whether or not a new electrical power subsystem design is worthy of further investigation and design.

1.5 Report Outline

The remainder of this thesis is organized as follows. Chapter 2 briefly describes the research conducted. The first area covered is that of the Navstar GPS system, its mission, and its history. The next topic covers the current GPS design, GPS IIR, and its capabilities. Finally, current work and research in the area of satellite power systems are discussed.

Chapter 3 explains the systems analysis approach and applies this analytical technique to the problem at hand. The first section of the chapter defines the concept of a systems analysis and describes the seven steps involved. The second section specifies the results attained from applying this method.

In chapter 4, the systems identified in Chapter 3 are designed. The designs are limited to basic characteristics and primary properties in order to gain a quick but insightful description of the system. The systems designed are then compared with the current GPS electrical power system.

Chapter 5 is the final chapter of this report. It contains a summary of the research and results, and recommendations for future work in this area.

II. Literature Review

The first two sections of this chapter cover the history and workings of the Navstar Global Positioning Satellites, and the technical aspects of the satellites' electrical power subsystem, respectively. Current work in the area of power systems is discussed in the final section and constitutes the majority of the research effort for this project.

2.1 GPS History (3:195-198)

The GPS program began in 1973 to replace the 200 meter accuracy Transit navigation system by the mid 1980s. Rockwell was awarded the prime contract in 1974 for 12 Navstars (Navigation System using Timing and Ranging). The original satellites were first launched in the late 1970s and early 1980s. The seventh satellite launched was lost due to an Atlas failure. Navstar 8 & 9 were launched in the summer of 1983 and 1984, respectively. Both suffered attitude control and battery problems, but remained in service until 1993. The decision to harden the satellites and to re-designate as prototypes the eleven already launched, delayed the timeliness of the program. The new design, completed in 1982, increased power from *400 W* to *700 W*.

Navstars 13-21 were designated Block II satellites. These satellites were originally designed for launch aboard the Space Shuttle, but the Shuttle's scheduling difficulties led to the procurement of the Delta II expendable launcher. The first Block II satellite was launched in February 1989 aboard the new Delta II medium launch vehicle. The Block II, and all subsequent versions were, or will be, launched into a 55° inclined, 12 hour orbit.

The next series of satellites are known as Block IIA. These satellites have only minor differences from the Block II design. One Block IIA was launched in 1990 but it developed a problem in the circuit board of one of the solar array's control board, so only one more satellite was launched in 1991. Sixteen satellites were launched between 1992 and 1995, bringing the constellation to its full complement of 24, 21 active and 3 on-orbit spares, after shutting off some of the original satellites.

The contract for the newest design, designated Block IIR, was awarded to General Electric Astro Space, which is now part of the Lockheed/Martin corporation. The actual payload will be produced by ITT Aerospace/Communications. Twenty satellites are to be produced under this contract, to replace the original Block II and IIA satellites when they are decommissioned. The first delivery was planned for October 1995 for a launch in March 1996.

The Navstar Global Positioning System was declared operational on May 9, 1990, when USAF Space Command took control of the daily operation of the system from USAF Space Systems Division. The system was then providing two-dimensional coverage for 14 to 22 hours a day. By the time tensions broke in the Persian Gulf War in early 1991, there were fifteen satellites spaced appropriately to provide maximum coverage over the region.

2.2 GPS Block IIR Current Capabilities

The Navstar Global Positioning System is a space-based radio positioning, navigation and time transfer system providing 16 meter position accuracy, 0.1 meter per second velocity accuracy, and 0.1 microsecond time accuracy to military users. Each satellite carries four atomic clocks, two with a stability of 1 second in 300,000 years and two with approximately half that stability (3:196). By placing the satellites into 12 hour, 55° inclination orbits, using six orbital planes, a minimum of four satellites will be in view anywhere in the world. Certain geometric constrictions inherent in the location calculations will allow only about 98% of the world to receive the most accurate data at any given time.

2.3 Electric Power System Technologies

Considering the main thrust of this thesis involves the design of an electrical power system, the majority of the technological review was accomplished in this area. The first objective was to understand the electrical power system currently being used in the Block IIR satellites. The second objective was to concentrate on the state of the art technology and research in the hopes of discovering a system or combination of systems that would be able to meet the necessary power requirements with better properties such as mass, and external area.

2.3.1 GPS Electrical Power Subsystem (4)

The electrical power subsystem (EPS) of the Navstar GPS Block IIR is composed of several distinct components. Figure 1 shows a simplified block diagram of the EPS. The solar arrays collect incident sunlight and convert it into electricity. Shunt dissipators absorb excess energy from the solar arrays while the solar array drive controls the orientation of the arrays. Rechargeable batteries are used to store energy for the periods of eclipse when the solar arrays are unable to produce power. The power regulation unit controls the voltage of the spacecraft to support all the spacecraft loads and ensures the batteries are properly charged. An ordnance controller uses power to fire explosive bolts and other ordnances as commanded. The entire system mass is approximately *265 kg*.

2.3.1.1 Solar Arrays

The primary aspect of the Navstar GPS power system is the solar arrays. There are two array assemblies, each consisting of two panels. A solar array panel consists of eight isolated circuits, each having nearly *1.9 amps* output at beginning of life. Each panel is approximately *1.778 m* by *1.905 m* by *0.0254 m*. The average operational requirements placed on the power system are approximately *800 W* to *900 W*, with peak power levels of *1600 W*.

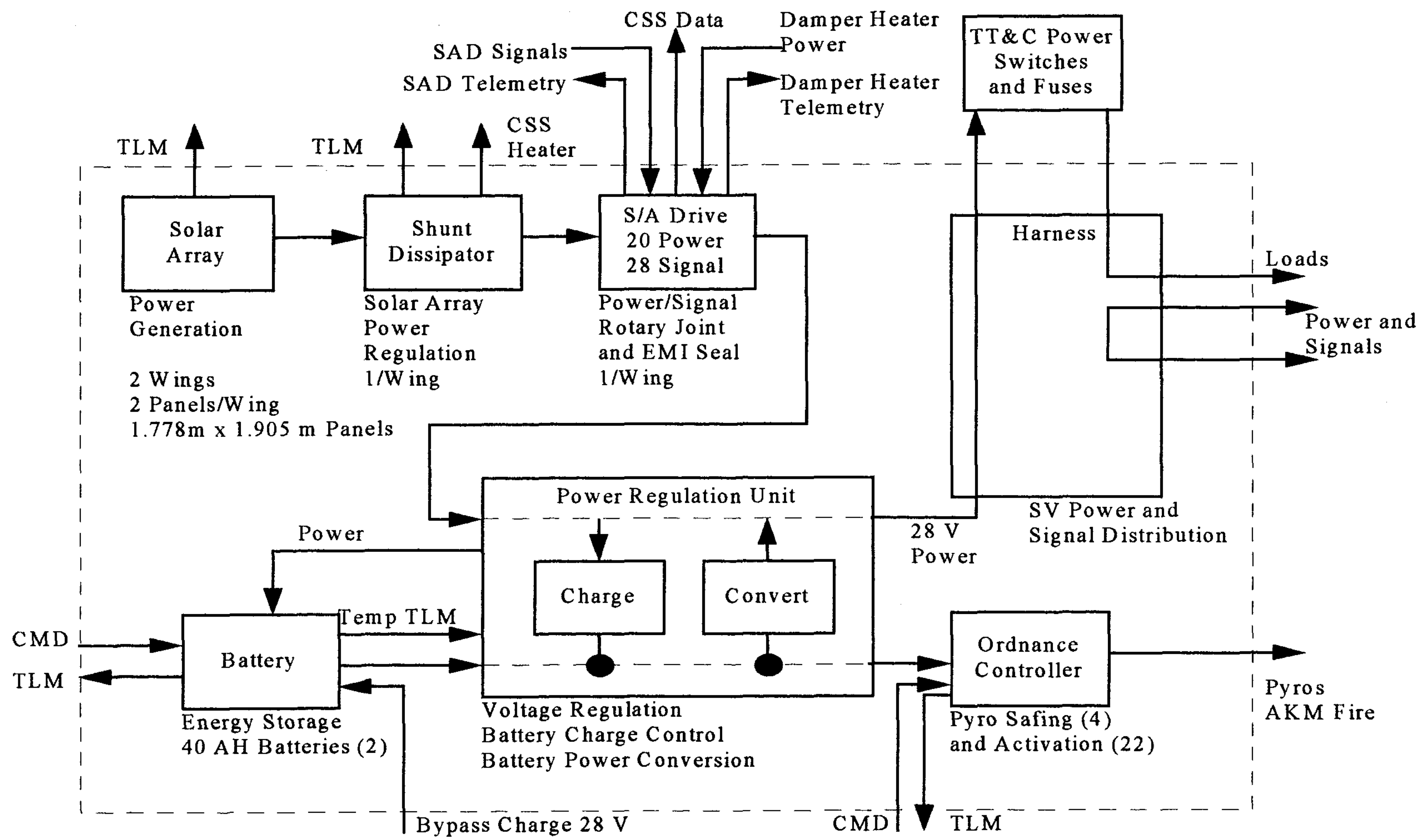


Figure 1 - EPS Simplified Block Diagram (4:4.4-3)

2.3.1.2 Energy Storage

Navstar GPS satellites are in a 12 hour, 55° orbit, which produces a maximum eclipse time of approximately 56 minutes. During this time there is no incident solar energy on the solar panels, so no power is being produced. To provide power during these eclipse periods, two nickel hydrogen (NiH_2) 40 Amp-Hour batteries are used. The maximum depth of discharge (DOD) allowed is 60% at an eclipse load of 958 Watt-Hours for a 56 minute eclipse period every 12 hours.

2.3.1.3 Power Control and Distribution

The Power Regulation Unit (PRU) regulates the 28 volt direct current power bus and converts battery voltage power to the bus voltage. The PRU is also responsible for controlling the battery charge and providing required power to certain portions of the electrical power subsystem.

2.3.1.4 Thermal Control

The GPS IIR Thermal Control Function (TCF) controls and maintains the satellite temperatures within acceptable limits during all mission phases. Passive thermal control is accomplished by such techniques as thermal coatings and insulation blankets. Heaters and thermostats provide active thermal control. Direct radiators maintained parallel to the sun line provide heat rejection. The GPS IIR vehicle does not require the use of active radiators or vented louvers. Passive radiators consisting of conductive white painted surfaces are used for the exterior surface of the satellite. Thermostatically controlled heaters maintain minimum temperatures for batteries and other components and subsystems.

2.3.2 Current Power System Technologies

When designing a power system for a satellite, several primary aspects must be considered; where the power will come from, how it will be converted into electrical power, how to store energy for eclipse or emergency use, and how to eliminate excess heat. Many new and exciting areas of research are currently being investigated in these areas, not all of which are discussed here. This report is limited to discussing those areas that proved pertinent to the analysis.

2.3.2.1 Power Sources

Most spacecrafts in orbit for more than a few hours or days rely on either solar or nuclear energy for power. Photovoltaic solar arrays, specially designed materials that release energy when impacted by photons of sunlight, have been the mainstay of solar powered spacecraft. Solar thermal power differs from solar photovoltaic in that it uses the heat from concentrated solar energy to drive a heat engine which in turn drives a generator or alternator to produce electrical power. Nuclear sources are primarily intended for requirements outside the Earth's orbit, such as interplanetary travel and lunar bases. The potential health hazards of a nuclear disaster deter its use in Earth orbit.

The primary focus in solar thermal designs is that of the solar concentrator. Other characteristics of solar thermal systems, including energy conversion and storage, are discussed in the following sections. The design and application of solar concentrators have received significant attention since the proposed use of solar thermal power on the space station to supplement photovoltaic arrays.

Numerous concentrator designs are currently being investigated for use in solar thermal power systems. For satellite use, the concentrator must be storable for launch and self-deployable on orbit. NASA Lewis Research Center and Cleveland State University have designed and built a two meter prototype deployable solar concentrator based on the proven concept of the Sunflower solar concentrator of the 1960's (5:867-873). Science Applications International Corporation (SAIC) has developed a two meter flexible membrane prototype concentrator. The membrane can be rolled up and is deployed by telescoping arms and rigidizing foam (6:875-880). These prototypes, designed to prove the level of technology required for a Space Station concentrator, are approximately the right size for a satellite's power needs.

2.3.2.2 Power Conversion Systems

Both nuclear and solar thermal systems provide thermal energy that must be converted to electrical energy. A power conversion unit is required for this task. Power conversion systems can be separated into two general categories, dynamic conversion and direct energy conversion (DEC).

Dynamic systems use the heat engine principle to provide the mechanical work necessary to generate electricity in a turboalternator assembly. Dynamic systems include the Brayton and Rankine cycles, which are rotating systems, and the Stirling engine, which is a reciprocating system (7:75). All of these systems can operate as an 'open' or 'closed' system. The working fluid of an open system is vented after it passes through the conversion unit, while a closed system recycles the working fluid throughout the system. Open systems require large fluid storage areas and the fluid venting affects attitude control. All systems considered hereafter are closed systems.

Direct energy conversion systems require no moving mechanical parts to produce electricity. DEC systems include thermoelectric, thermionic, and magnetohydrodynamic (MHD) devices. In spite of its name, the magnetohydrodynamic system has no moving mechanical parts, so is considered a direct system. Each of the power conversion systems mentioned are described in more detail in the Appendix.

2.3.2.3 Energy Storage Systems

Energy storage is necessary to provide continuous power to a solar powered satellite when in the shadow of the Earth. Rechargeable batteries have been the mainstay for energy storage for many years, but research in the areas of mechanical and thermal energy storage is continuing. New chemicals for use in batteries are also being investigated. The standard Nickel-Cadmium batteries are giving way to Nickel-Hydrogen and Sodium-Sulfur, which show significant weight reductions.

Mechanical energy storage typically takes place in a flywheel. Excess energy produced during the non-eclipse portion of the orbit is converted into angular momentum by spinning up a flywheel. When the power is needed, the flywheel is forced to do work against a load, thus returning the power. Research is also being done on an Integrated Power and Attitude Control System (IPACS). An IPACS stores excess energy kinetically in mechanical rotors with the accompanying angular momentum available for attitude control of the spacecraft (8:247-249).

Thermal energy storage is ideal for use with a solar thermal power system. The excess heat produced during the sunlit portion of the orbit is used to melt a solid. This phase change material (PCM) remains in a liquid state until the satellite enters eclipse and no further heat is present. At this point the liquid solidifies, releasing the stored energy. The energy released as a liquid solidifies is known as the Heat of Fusion and is measured in Joules per kilogram (*J/kg*). Salt mixtures with very high heats of fusion such as Calcium-Diflouride--Lithium Fluoride (CaF_2/LiF), are being investigated for thermal storage uses.

2.3.2.4 Thermal Control

The function of a satellite thermal control system is to control the temperature of individual spacecraft components so that proper operation is maintained throughout the mission. Thermal loading, caused by changes in temperature, needs to be minimized due to the fatiguing effects it has on delicate materials. Conduction and radiation are the primary means of energy exchange, since convection is impossible in the vacuum of space. Conduction drives the motion of thermal energy between adjacent components; radiation is how excess energy is removed. The amount of energy radiated by a body is directly proportional to the surface area of the radiating surface.

The efficiency of a heat engine power conversion system depends on the high and low temperatures of the working fluid. Extreme high and low temperatures provide a greater efficiency, but require extensive thermal control and large energy removal techniques. Excess thermal energy from solar radiation and mechanical heating must also be removed.

As mentioned earlier, radiation is the only form of energy exchange usable for rejecting waste heat, not including venting a heated material. Numerous radiation techniques are possible, including simple radiators, louvers, and movable appendages. The most important thing to remember about thermal control and its role in power system designs is that all waste heat must be removed to ensure proper operation of internal components. The more waste heat produced, the larger and more massive the radiating surface must be. The radiator mass must be traded off against the efficiency of the power conversion system.

III. Systems Analysis

This chapter discusses the fundamentals of the systems analysis technique to problem evaluation, defining the seven primary steps involved. This chapter also presents the results attained when this technique was applied to the redesigning of the electrical power subsystem of the Navstar GPS satellite.

3.1 Introduction

Systems analysis is a predefined, analytical approach used to define, analyze, and evaluate a problem. Systems analysis involves several distinct steps. Different authors use different steps, but most accomplish the same thing. The approach used for this thesis includes seven steps as listed below (9).

1. Problem Definition - Accurately and completely define the existing problem to be solved. Define subproblems, needs, alterables, constraints, actors, and boundaries.

2. Value System Design - Create a measurement system to use when comparing different solutions. Measurable attributes are identified and may be weighted. Subjective values such as high/low or good/bad are acceptable.

3. System Synthesis - Create as many different systems as possible, or practical, that may solve the problem.

4. System Modeling - Model systems created in synthesis step for testing. May use mathematical, computer, analytical, scale, or any other types of models as long as they accurately represent the system or a portion of the system.

5. System Evaluation - Evaluate the different systems' performances and parameters using the value system design to accurately weigh the outcomes.

6. Decision Making - Compare the evaluations and chose the best system to solve the stated problem or fill the observed need.

7. Implementation - Implement the system chosen.

Following these seven steps does not guarantee the best answer to a problem will be found, but it does allow the decision maker to have the most information when making decisions concerning the problem. The results of applying these seven steps are discussed in the following section.

3.2 Results

The following sections cover the results obtained when the systems analysis steps are applied to the electrical power subsystem redesign problem. The last two steps of the seven step process, decision analysis and implementation, are beyond the scope of this work. The information researched and presented in this report is solely intended to provide information to a decision maker who has the power to make design changes.

3.2.1 Problem Definition

To define the problem accurately and completely, numerous subtasks are accomplished. The needs, alterables, constraints, and actors are identified. The major subjective factors and system boundaries are also defined. A one line problem definition is defined that completely represents all aspects of the problem. Referring back to the Problem Statement identified in Chapter 1, the problem is defined as:

The Navstar Global Positioning System Electrical Power Subsystem must be redesigned to meet new requirements associated with future missions.

Using Athey's problem definitions, the problem has a positive deviation. This means that no problem exists with the current system, but the performance expectations for future systems have increased (10:43-48). Higher efficiency, lower mass, and smaller size are examples of the new expectations for future electrical power systems. The subtasks within the problem definition step are summarized in the following sections.

3.2.1.1 Needs

The following needs are identified for the redesigned electrical power subsystem.

- Must provide worst case scenario power requirements for all subsystems at all times plus additional power for energy storage purposes, if needed.
- Must minimize mass, volume in bus, complexity, and requirements placed on other subsystems, such as strict pointing accuracy requirements for attitude control.
- Must maximize efficiency.
- Must minimize degradation, risk of failure, and environmental and human health hazards.

3.2.1.2 Alterables

The following areas are identified as potential alterable aspects of the electrical power subsystem.

- Energy source and conversion technique
- Size, weight, and volume
- Energy storage type and size
- Redundancy, reliability, and timeliness
- Power regulation and distribution technique
- Thermal management system

3.2.1.3 Constraints

The attributes listed below are some of the more important constraining features addressed while designing the electrical power subsystem. This does not mean that these aspects can not be changed at all, only that their values have a great impact on the design. In most cases, some upper or lower limit exists and a design outside these limits would immediately be eliminated from consideration.

- Orbital parameters
- Expense
- Minimum power requirements
- Current technology
- Size, weight, and volume
- Environmental and health factors
- Compatibility with other subsystems

3.2.1.4 Actors

The following organizations are identified as major players in the design, manufacture, launch, control, and use of the Navstar GPS satellite and its electrical power subsystem.

- Air Force Space Command (AFSPC)
- U. S. Space Command (USSPC)
- Air Force Materiel Command (AFMC)
- Environmental Protection Agency (EPA)
- Office of Management and Budget (OMB)
- Contractors, users, and payload managers

3.2.1.5 Subjective Factors

Some of the major areas of the design that are open to subjective evaluation are:

- Redundancy
- Reliability
- Timeliness
- Safety factors
- Maintainability
- Availability

3.2.1.6 System Boundaries

The following boundaries limit the capabilities of the designed system.

- Current technology
- Subsystem compatibility
- GPS Request for Proposal minimum standards
- Funding level

3.2.2 Value System Design

The Value System Design involves determining the system's objectives and the parameters that are used to measure the objectives. This design typically takes the form of an objective hierarchy. An objective hierarchy is a graphical representation of the system's objectives. The hierarchy shows the breakdown of objectives into sub-objectives and sub-sub-objectives, until a measurable parameter is determined. The objective hierarchy for this project is shown in Figure 2. Because this is an independent thesis versus a group project, only some of the measurables could be evaluated. Working for an academic institution, the measurables of greatest interest are those of a scientific nature. These measurables are highlighted in bold in Figure 2.

A list of the measurables is included after the diagram. Some of the items listed are not used during the evaluation, but are listed here for future reference. Additional measurables not listed may become apparent in future research on this problem. No weightings are applied at this time.

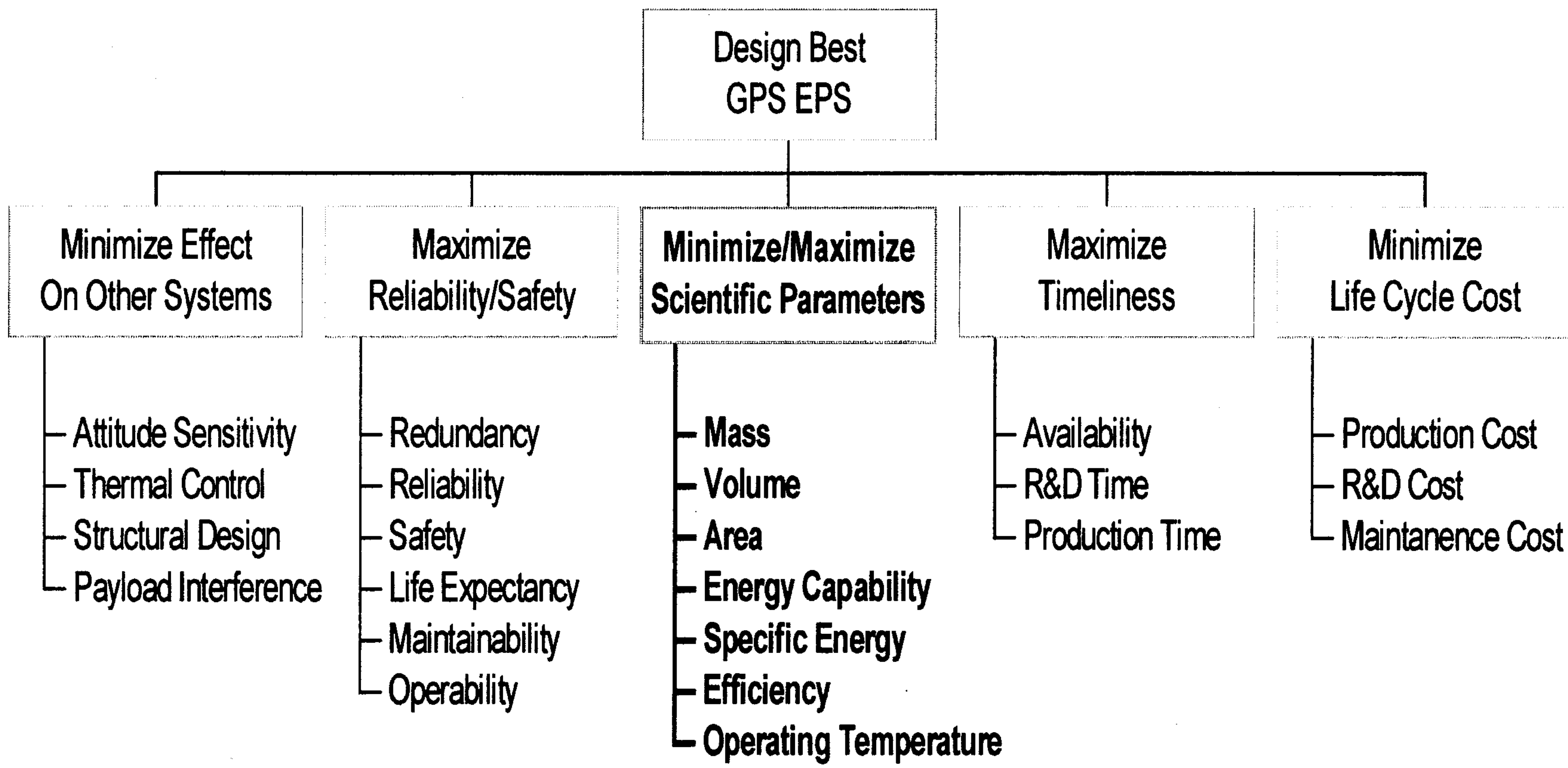


Figure 2 - Objective Hierarchy for GPS Electrical Power System Redesign

- Cost ($)
- Mass (*kg*)
- Energy Capacity (*W*)
- Specific Energy (*W/kg*)
- Efficiency (%)
- Surface Area (m^2)
- Life Expectancy (yrs)
- Reliability (% or subjective)
- Redundancy Level (subj)
- Timeliness (subj)
- Maintainability (subj)
- Availability (subj)
- Ground Support Requirements (subj)

3.2.3 System Synthesis (11:351)

The various space power sources, with a power converter if needed, are listed below.

- Primary Batteries
- Secondary Batteries (with other recharging source)
- Fuel Cell
- Regenerative Fuel Cell
- Chemical Dynamic
- Nuclear (with direct or dynamic conversion)
- Radioisotope (with direct or dynamic conversion)
- Photovoltaic
- Solar (with direct or dynamic conversion)
- Ground-based Transmitter

3.2.4 System Modeling

The system modeling step involves designing the electrical power subsystem using the power sources listed above. In order to avoid designing ten different power subsystems, (more if the different conversion possibilities are included) a preliminary evaluation is accomplished on the power sources and conversion techniques. This initial evaluation is used to identify the best options and eliminate the worst. Only the best systems identified in this evaluation are modeled in more detail. The models are evaluated by comparing them with the GPS IIR design. The designs and evaluations are covered in Chapter 4.

3.2.5 System Evaluation

This section covers the preliminary evaluation of the power sources and conversion techniques identified in the system synthesis step. Those power sources requiring a conversion system to change thermal energy into electrical energy are listed with two conversion techniques, dynamic and direct. Evaluation of specific conversion techniques is covered in later subsections.

The evaluation approach involves listing the available power sources and the major measurables of the electric power system as identified in the System Synthesis and Value System Design steps. Each power source is ranked for its appropriateness with each measurable. The most appropriate system for a specific measurable is not always the one with the largest or smallest measurable value, but is the system whose measurable is most suited to the satellite, orbit, and mission requirements.

The eleven systems evaluated are ranked from one to eleven for each measurable. One represents the most inappropriate system and eleven, the most appropriate. The values are then multiplied by the measurables' weighting factors and added for each system. The system with the highest overall value is the most appropriate system for the GPS redesign.

Table 1 shows the results of this evaluation. The top three options are: solar power with dynamic conversion, solar power with direct conversion, and nuclear power with direct conversion.

3.2.5.1 Dynamic Conversion Evaluation

Within the dynamic conversion category there are three primary techniques to be considered; Closed Brayton Cycle (CBC), Rankine Cycle, and Stirling engine. The latest design in Stirling engines is the Free Piston Stirling Engine (FPSE), which will be the design used throughout this report. See the Appendix for more detailed information about these systems. Two different styles of FPSEs are evaluated. The first uses heat pipes to transport the thermal energy to the Power Conversion Unit (PCU) while the second uses a separate liquid metal pumped loop.

The Rankine cycle is incapable of achieving the high thermodynamic efficiencies of the CBC or Stirling. A single-stage potassium Rankine cycle has an efficiency range of 15% to 20% for the temperature range used in this thesis. CBC and Stirling cycles have efficiencies in the 35% to 45% range. For a solar dynamic power system, most of the mass accrues from the concentrator, receiver, and radiator. These masses are roughly

Table 1 - Preliminary Evaluation of Power Systems

			Power Sources										
				Fuel Cell		Chemical		Nuclear		Radioisotope		Solar	
Measurable	units	wt	Batteries	Standard	Regenerative	Dynamic	Direct	Dynamic	Direct	Dynamic	Direct	Dynamic	Direct
Energy	W	1	1	4	5	2	3	8	9	6	7	10	11
Mass	kg	1.5	1	2	3	4	5	6	7	8	9	10	11
Volume	m^3	1	5	3	4	1	2	8	9	6	7	11	10
Lifetime	yr	1.1	1	2	5	3	4	10	11	8	9	6	7
Safety	subj	1.1	9	7	8	5	6	1	2	3	4	10	11
Timeliness	subj	1	11	10	9	8	1	7	4	6	3	5	2
Attitude Sensitivity	subj	1.25	11	10	9	3	5	6	8	4	7	2	1
Specific Cost	$/kg	1.25	11	10	9	8	3	7	4	1	2	6	5
Total			57	54.9	59.3	39.55	34.5	60.35	61.8	48.35	56.05	68.6	66.8

Note 1: Solar direct conversion system does not include solar cells
Note 2: Energy rankings according to appropriateness of medium for providing level of energy required (app. 2kW)
Note 3: Dynamic systems include Brayton, Rankine, and Stirling engines
Note 4: Direct conversion systems include thermoelectric, thermionic, and magnetohydrodynamic

BEST OPTIONS: #1 -- Solar Dynamic
#2 -- Solar Direct
#3 -- Nuclear Direct

proportional to the inverse of the efficiency. As a result, Rankine solar dynamic power system masses are not competitive with the other two cycles, and the Rankine cycle will no longer be considered a candidate for design (12:62).

Detailed model designs for the two FPSEs and the CBC were created, tested, analyzed, and compared in a NASA definition study (12). Five power systems evaluation criteria were identified and weighted for the power levels and mission types involved. All of the criteria used were identified as measurables in the Value System Design. The low power, high altitude study mission closely resembles the GPS requirements and will be used as the basis for evaluations and efficiencies throughout this report. Quantitative differences between systems were determined, where possible. Qualitative differences were measured as being similar, better, worse, much better, or much worse. Nonlinear scaling was established to reduce the comparative results to numeric rankings for each subcriterion. Subcriterion rankings were then averaged and multiplied by the weighting to determine the overall criterion rankings as shown in Table 2. The results show that the Heat Pipe Stirling Engine is the most appropriate system for the requirements.

Table 2 - Dynamic Conversion Evaluation Results (12:151)

Criteria	*Weight*	*Brayton*	*PL FPSE*	*HP FPSE*
Reliability/Safety	50.0	0.0	-6.2	6.2
Technology Readiness	14.0	2.2	-0.8	-1.4
Performance	14.0	0.0	9.8	14.0
Operability	14.0	0.0	0.1	-0.3
Life Cycle Cost	8.0	0.0	0.5	1.0
Total	100.0	2.2	3.4	19.5

3.2.5.2 Direct Conversion Evaluation

The direct conversion techniques include; thermoelectric converters, thermionic converters, and magnetohydrodynamic generators. See the Appendix for more information regarding these conversion techniques.

Magnetohydrodynamics show good conversion efficiencies when analyzed in a terrestrial environment. Unfortunately this design is unacceptable for space applications due to the extremely high temperatures involved (2000 to 3200 K), the power losses required to produce the magnetic field, and the need for a highly ionized gas (13:309-311).

Thermionic energy conversion is another direct conversion technique being researched today. Some application deficiencies have yet to be resolved before this technique can be considered more appropriate than thermoelectrics or dynamic systems. The space charge in the atmosphere between the cathode and anode reduces the efficiency of the system. This can be overcome by reducing the distance between the plates, but then radiative heat transfer between the plates also keeps the efficiency down (7:95).

Another point of concern is the fact that the efficiency depends on the work function and temperature of the materials used for the collector plates. The work function is the potential barrier that must be overcome by electrons leaving either electrode. The operating temperatures required to produce a reasonable current, given the work function of the materials currently used, are too high and the materials are evaporating (7:95).

Eliminating magnetohydrodynamics and thermionics only leaves thermoelectric power conversion within the direct conversion category. The Alkali Metal Thermal-to-Electric Converter (AMTEC) shows mass and cost savings due to efficiencies significantly higher than other direct conversion systems operating within the same temperature limits (14:861). More information on AMTEC systems can be found in the Appendix. An AMTEC will be used in the second design, described in Chapter 4.

3.2.5.3 Nuclear Power Comments

The third option identified in the preliminary evaluation involves a nuclear power source. Putting a nuclear reactor into Earth orbit is a highly political task. The enormous impact a launch failure or other accident may have on the space environment, the atmosphere, the ecosystem, and human health make orbiting a nuclear reactor a discouraged act.

Nuclear systems emit large amounts of gamma rays and other subatomic particles. Shielding electronic equipment from this radioactive bombardment would significantly raise the mass of the system. These concerns, along with end of life re-entry problems, manufacturing requirements, and safety testing, make a nuclear power system inappropriate for this design. No further analysis in this area will be done.

IV. Design

This chapter describes the electrical power subsystems modeled for step four of the seven step process. The primary components of the subsystems are designed to produce the same amount of usable power as the current GPS IIR photovoltaic solar array design, *1.6* kW_e (4:27). Overall system mass and external area are the primary parameters of interest, but other concerns will be mentioned where appropriate. The two solar thermal designs, a solar dynamic and a solar direct, are then compared with the current GPS IIR system. A simple diagram of the designs is shown below in Figure 3.

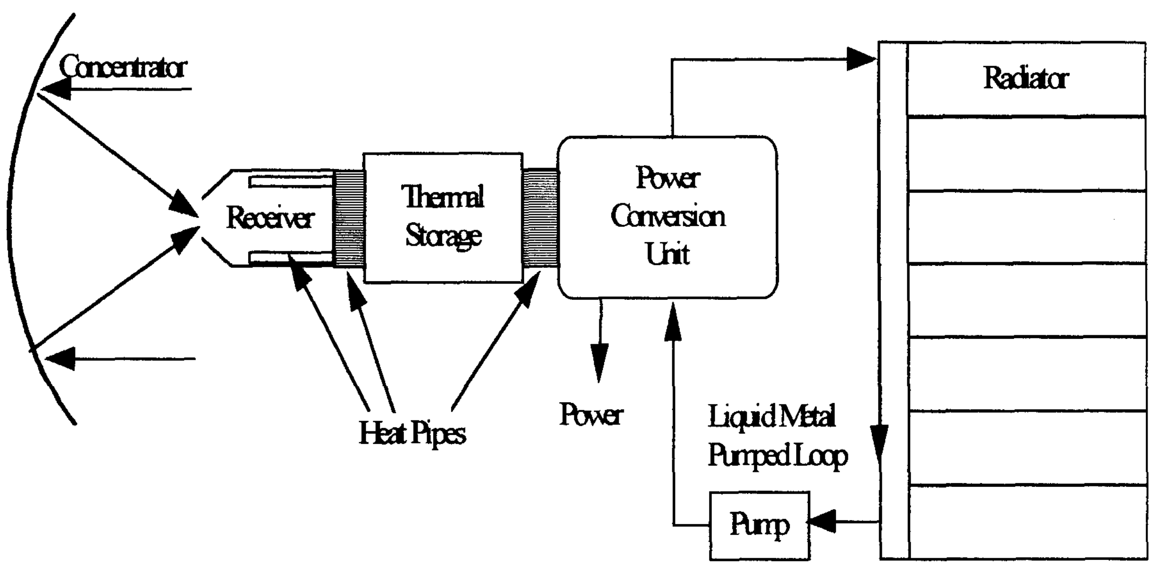


Figure 3 - Solar Thermal Electric Power System Diagram (12:107)

4.1 Introduction

Two electrical power systems are modeled in this chapter. Both are solar thermal systems with essentially the same components. A solar concentrator collects the solar thermal energy and focuses it into the receiver aperture. Within the receiver the thermal energy heats the working fluid of the converter. Heat pipes transfer the working fluid through a solid phase-change material (PCM), usually a salt mixture, to the power conversion unit (PCU). The PCM stores some of the thermal energy by melting. When the satellite enters eclipse and no solar energy is incident on the concentrator, the salt will release its stored energy to the working fluid by freezing back into a solid.

The PCU converts the thermal energy of the working fluid into electrical energy. A dynamic system first converts the thermal energy into mechanical energy via a thermal cycle that drives a turbine. The turbine drives an alternator or generator converting the mechanical energy into electrical energy. A direct system converts the thermal energy directly into electrical energy.

Excess energy lost due to the inefficiency of the PCU must be rejected to maintain the proper operating temperatures. The excess energy heats a separate liquid metal loop that is pumped to the radiator. The liquid metal heats the radiator's working fluid, usually ammonia, before returning to the PCU. The radiator distributes the heat and releases it through the radiator fins. The radiators modeled for this study are designed only to radiate the waste heat from the PCU. Electronic component and bus cooling are accomplished through the existing GPS IIR design.

The primary components discussed above are designed for both models. Many similarities exist between the two designs. This is done intentionally to increase the accuracy of the comparison of systems. The primary difference between the two systems modeled lies in the PCU. The first design uses a dynamic system with a relatively high efficiency and mass; the second uses a direct conversion device that has a lower mass but a lower efficiency as well. The lower efficiency increases the radiator, concentrator and energy storage masses and areas.

4.2 Design #1

The first design is for the solar dynamic system. As discussed in Chapter 3, the Free-Piston Stirling Engine (FPSE) is the most appropriate dynamic conversion unit. The efficiencies for the various components of the system are based on a NASA sponsored study (12:102). These efficiencies are listed in Table 3. The Solar Multiple indicates how much extra energy is required during the shortest sunlit portion of the orbit, in order to provide the desired power throughout the associated maximum eclipse period. Dividing the Solar Multiple by the product of the efficiencies determines the ratio of input power required at the concentrator to the desired output power from the power conversion unit.

4.2.1 Concentrator

Using the total input to output power ratio, and knowing that $1.6\ kW_e$ is the desired steady-state power output of the system, the necessary input energy is determined. This

Table 3 - System Efficiencies and Multipliers for Solar Dynamic Model (12:102)

Component	*Efficiency*
Power Conversion Unit plus Alternator	0.408
Receiver and Thermal Energy Storage	0.905
Receiver Interception	0.975
Concentrator	0.802
Solar Multiple	1.084
Input-Output Ratio	3.754

required input energy determines the size of the concentrator, using the fact that the average amount of solar energy at Earth's orbit is $1.353\ W/m^2$. Over the lifetime of the concentrator space dust, radiation, and oxidation degrade the concentrator. A 10% degradation factor to compensate for lifetime degradation is applied to achieve a final concentrator area. Details and calculations for the sizing of the concentrator are summarized in Table 4.

The concentrator modeled in this design is based on a Science Applications International Corporation (SAIC) flexible membrane concentrator study (6:875-880). The concentrator is composed of four primary segments; the membrane, circumferential annular rib, radial support ribs, and rigidizing foam. The mass of the membrane is a

Table 4 - Concentrator Area Calculations for Solar Dynamic Model

Parameter	*Value*
Output Power	1.60 kW_e
Input-Output Ratio	3.754
Input Power	6.01 kW_t
Solar Constant	1.353 kW/m^2
Required Area	4.44 m^2
Degradation Area	0.44 m^2
Total Area	4.88 m^2

function of its area, while the other components all depend on the radius of the concentrator. Using the separate component masses from the SAIC design and multiplying by a ratio of areas or radii, where appropriate, the component masses for the modeled concentrator are determined. Table 5 summarizes the SAIC design masses and coordinating model masses.

4.2.2 Receiver/Energy Storage

The primary tasks of the receiver are to collect the concentrated solar energy, heat the working fluid, and store excess heat for eclipse periods. The working fluid is contained in

Table 5 - Concentrator Components and Masses for Solar Dynamic Model (6:877)

Component	*SAIC mass (kg)*	*SAIC area(m^2) or radius (m)*	*Model area(m^2) or radius (m)*	*Model mass (kg)*
Membrane	266.0	254.5 m^2	4.88 m^2	5.10
Annular Rib	44.0	9.0 m	1.25 m	6.10
Radial Ribs	22.0	9.0 m	1.25 m	3.05
Rigid Foam	198.0	9.0 m	1.25 m	27.41
Total	530.0			41.66

tubes that pass through the centers of cylinders of PCM. The tubes circulate the working fluid from the aperture end of the receiver, where the fluid is heated by the concentrated solar energy, to the power conversion unit. The ideal PCM for minimizing the mass, while working with the operating temperatures involved, is Lithium-Fluoride (LiF). The latent heat of fusion of LiF, the amount of energy involved in the freezing or melting of the substance, is approximately *1040 kJ/kg* (12:14-15).

The amount of PCM needed depends on the desired output power, the efficiencies of certain components of the system, and the latent heat of fusion of the PCM. Starting with a $1.6\ kW_e$ output from the PCU, applying the efficiencies of the PCU and receiver determines the power capacity of the receiver. The PCM must be able to provide this power for the duration of a maximum eclipse period, 56 minutes or 3360 seconds. Multiplying the power and the time determines the energy required of the PCM. The latent heat of fusion defines the amount of energy per unit mass of PCM. Knowing the energy requirement and the latent heat of fusion, the mass required is determined. An additional 10% of the mass is added for redundancy and lifetime degradation effects.

The mass of the receiver structure, working fluid tubes, PCM canisters, and other materials typically measure two to four times the mass of the PCM (12:14). In an attempt to keep this design realistic, a multiplier of three was used to determine the mass of the rest of the receiver package. Table 6 displays the results of the above discussed Receiver/PCM calculations

4.2.3 Power Converter

The power conversion unit (PCU) for this model is composed of a Free Piston Stirling Engine (FPSE) and a linear alternator. The FPSE has a high operating temperature of *1033 K* and a temperature ratio, high to low, of 2.9. From the NASA design, this FPSE has a mass-to-power ratio of $6.7\ kg/kW_e$ (12:93).

The maximum output of the PCU occurs when there is no eclipse. The initial energy input at the concentrator is affected only by the efficiencies, not by the solar multiple. With no degradation or eclipse, the oversized concentrator collects $6.60\ kW_e$, which produces $1.91\ kW_e$ after the various efficiencies are considered. Using the mass-to-power ratio defined above and the maximum energy output, the mass of the FPSE is *12.78 kg*.

Table 6 - Receiver & Energy Storage Calculations for Solar Dynamic Model

Parameter	*Value*
Output Power	1.60 kW_e
PCU Efficiency	0.408
Receiver Efficiency	0.905
Receiver Power Capacity	4.33 kW_t
Maximum Eclipse Time	3360.00 s
PCM Required Energy	14559.64 kJ
Latent Heat of Fusion of LiF	1040.00 kJ/kg
Mass of PCM	14.00 kg
Additional 10% Mass	1.40 kg
Total PCM Mass	15.40 kg
Receiver Mass (3x PCM mass)	46.20 kg
Total Mass	61.60 kg

The FPSE converts thermal energy into mechanical energy. The linear alternator mentioned above is required to convert the mechanical energy into electrical energy. The alternator mass is approximated as one-third the engine mass (12:93). This means the alternator mass is approximately *4.26 kg* and the total PCU mass is *17.04 kg*.

4.2.4 Radiator

The radiator design modeled here is responsible only for the rejection of the waste heat from the PCU. Excess satellite and electronic component heating will be controlled by the existing thermal control system for the GPS IIR. The material used for the construction of the radiator modeled will be the same material used in the NASA study: titanium pipes with aluminum fins (12:96). Equation (1) is the basic equation used to size a radiating surface (7:103):

$$P / A = \varepsilon \sigma T^4 \tag{1}$$

where

P = Waste Heat (W_t)
A = Radiator Surface Area (m^2)
ε = Radiator Emissivity
σ = Stefan-Boltzmann Constant = 5.67E-8 $Wm^{-2}K^{-4}$
T = Radiator Temperature (K)

Many details are involved in the thermal control of a satellite in orbit besides removing waste heat produced by the PCU. The Earth reflects some of the sun's energy toward the satellite, radiates its own energy toward the satellite, and absorbs the radiated energy from the satellite. The relative positions of the Earth and the satellite, as well as their orientation in relation to one another, all affect the thermal control of the satellite.

The Earth's reflected and radiated energies, along with other cosmic energy sources, provide a small amount of energy when compared to the large amount of waste heat from the PCU, but they should not be dismissed. To address these issues, this design uses Equation (1) to solve for an equivalent emissivity. By inserting the operating temperature, radiating area, and waste heat from the NASA study, an emissivity for the material can be determined. This emissivity incorporates all of the extra energy sources that need to be radiated as well as the waste heat. Using this equivalent emissivity, the area required to radiate the waste heat from the modeled design can be determined.

The NASA study design radiates a maximum of 22.5 kW_t through a minimum of $29.39\ m^2$ of radiating surface operating at a temperature of $398\ K$ (12:100, 102, 127). Inserting these values into Equation (1) and solving for ε gives an equivalent emissivity of 0.538.

The modeled design produces a maximum output of $1.91\ kW_e$ at 0.408 efficiency. This means $4.68\ kW_t$ must enter the PCU to produce $1.91\ kW_e$, leaving $2.77\ kW_t$ as waste heat to be rejected. Using the equivalent emissivity and waste heat determined above, with the same operating temperature, the area required for the modeled design is calculated as $3.62\ m^2$. Adding a redundancy factor of ten percent to compensate for degradation over the lifetime causes the area to grow to $3.98\ m^2$.

The NASA study radiator that the modeled design is based upon has a radiant surface area of $33.8\ m^2$ and a mass of $259\ kg$ (12:101). This leads to a radiator density of approximately $7.66\ kg/m^2$. Because the modeled design uses the same materials in an identical design, but with different dimensions, as the NASA-sponsored design, the density is maintained. A radiating surface area of $3.98\ m^2$, with a density of $7.66\ kg/m^2$, produces a radiator mass of *30.47 kg*.

The waste heat is transferred to the radiator via an external liquid metal loop. The excess energy from the PCU heats the liquid metal, which is pumped to the radiator by electromagnetic pumps. The smallest pump design examined by the NASA study for this application was an Annular Linear Induction Electromagnetic Pump (ALIP) with a mass of *22.68 kg* (12:74). Because this thesis is concentrating on the primary components, research into smaller, less massive pumps for radiator liquid metal loops was not accomplished. The best pump determined by the NASA study is used for this design. The final mass of the complete radiator subsystem, which includes the radiator and the pump, is *53.15 kg*.

4.2.5 Summary

The modeled solar dynamic design consists of four primary components; the concentrator, the receiver and thermal storage, the PCU and alternator, and the radiator. These components were designed to provide the required power over the lifetime of the satellite with the minimum mass and area, satisfying the objectives defined in the Value System Design. Calculated masses and areas for this design are summarized in Table 7.

Table 7 - Calculated Component Masses and Areas for Solar Dynamic Model Design

Component	*Mass (kg)*	*Area (m^2)*
Concentrator	41.66	4.88
Receiver/Thermal Storage	61.60	N/A
PCU/Alternator	17.04	N/A
Radiator/Pump	53.15	3.98
Sub-Total	173.45	8.86
Additional 10% Mass	17.34	N/A
Total	190.79	8.86

The additional ten percent mass is to compensate for wiring and such items as regulators and converters. Wertz and Larson recommend adding an additional one to four percent of the total spacecraft mass for these items (15:319). Wertz and Larson also estimate the GPS power system mass as 30% of the total mass. The added ten percent of the power system mass is equal to three percent of the total mass (15:806).

This completes one of the designs modeled for the System Modeling step of the systems analysis. A second design is modeled and evaluated in the following section. The final system evaluations are covered in Section 4.4, where the two modeled designs are compared with the current GPS IIR Electric Power System.

4.3 Design #2

The second design is for the solar direct system. In Chapter 3 the preliminary evaluation of different power conversion systems indicated that a thermoelectric system, particularly the Alkali Metal Thermal-to-Electric Converter (AMTEC) is the most appropriate direct conversion unit. The efficiencies for the various components of the system are maintained from the first design wherever possible, to allow better comparison of designs. The primary difference lies in the efficiency of the AMTEC. Efficiencies reported in design studies range from 15% to 25% (14:861). An efficiency of 18% is assumed for this design. The component efficiencies and multipliers for the solar direct model are listed in Table 8, along with the Input-Output power ratio.

Table 8 - System Efficiencies & Multipliers for Solar Direct Model (12:102; 15:864)

Component	*Efficiency*
AMTEC	0.180
Receiver and Thermal Energy Storage	0.905
Receiver Interception	0.975
Concentrator	0.802
Solar Multiple	1.084
Input-Output Ratio	8.510

4.3.1 Concentrator

The same process of sizing the concentrator is applied for this design as for the dynamic design, except using the new efficiencies. The output power and efficiencies determine the input power at the concentrator. The input power and the solar constant determine the area of the concentrator required. Once the area is known the mass can be approximated in a similar fashion as the dynamic concentrator. Results of the area calculations are presented in Table 9. Mass calculations are summarized in Table 10.

Table 9 - Concentrator Area Calculations for Solar Direct Model

Parameter	*Value*
Output Power	1.60 kW_e
Input-Output Ratio	8.51
Input Power	13.62 kW_t
Solar Constant	1.353 kW/m^2
Required Area	10.06 m^2
Additional 10%	1.01 m^2
Total Area	11.07 m^2

Table 10 - Concentrator Components and Masses for Solar Direct Model (6:877)

Component	*SAIC mass (kg)*	*SAIC area/radius*	*Model area/radius*	*Model mass (kg)*
Membrane	266.0	254.5 m^2	11.07. m^2	11.57
Annular Rib	44.0	9.0 m	1.88 m	9.18
Radial Ribs	22.0	9.0 m	1.88 m	4.59
Rigid Foam	198.0	9.0 m	1.88 m	41.30
Total	530.0			66.64

4.3.2 Receiver/Energy Storage

The modeling of the receiver and thermal energy storage components of the direct conversion design are accomplished following the same steps as the dynamic design, but again using the new efficiency of the PCU. Results are displayed in Table 11.

4.3.3 Power Converter

The power conversion unit (PCU) used for this design model is an Alkali-Metal Thermal-to-Electric Converter (AMTEC). The power density of this direct conversion device ranges between 185 and 230 watts per kilogram (14:864). No additional alternator or generator is required for this design. In an effort to minimize the mass of the system,

Table 11 - Receiver & Energy Storage Calculations for Solar Direct Model

Parameter	*Value*
Output Power	1.60 kW_e
PCU Efficiency	0.180
Receiver Efficiency	0.905
Receiver Power Capacity	9.82 kW_t
Maximum Eclipse Time	3360.00 s
PCM Required Energy	33001.84 kJ
Latent Heat of Fusion of LiF	1040.00 kJ/kg
Mass of PCM	31.73 kg
Additional 10% Mass	3.17 kg
Total PCM Mass	34.91 kg
Receiver Mass (3x PCM mass)	104.72 kg
Total Mass	139.62 kg

the best density within this range is used. The maximum output of the PCU occurs during a fully sunlit orbit, providing *1.91 kWe* as determined earlier. Using this maximum power value and a power density of *230 W/kg* drives the PCU mass to *8.29 kg*.

4.3.4 Radiator

The design of the radiator for the direct conversion design follows the same procedure as for the dynamic. The same materials are used, so the equivalent emissivity is still 0.538. The operating temperatures of the AMTEC are slightly different from the FPSE, however. The high end temperatures range from *900 K* to *1200 K*. Low temperatures range between *400 K* and *800 K*. Because the AMTEC is limited by the Carnot efficiency, the best efficiencies correspond with the largest temperature ranges. Keeping this in mind, the lower temperature is assumed to be approximately *400 K*.

The AMTEC PCU produces a maximum output of $1.91\ kW_e$ at 0.18 efficiency. This means $10.60\ kW_t$ must enter the PCU to produce $1.91\ kW_e$, leaving $8.69\ kW_t$ as waste heat to be rejected. Inserting the equivalent emissivity and waste heat determined above, an operating temperature of *400 K*, and the Stefan-Boltzmann constant into Equation (1), the area required for the modeled design is found to be $11.13\ m^2$. Adding a redundancy factor of ten percent to compensate for lifetime degradation causes the area to grow to $12.24\ m^2$.

Because the same material is being used for this radiator design as was used in the dynamic design, the density remains the same at $7.66\ kg/m^2$. A radiator with this density and an area equal to $12.24\ m^2$ will have a mass of *93.78 kg*. An electromagnetic pump is again needed to circulate the liquid metal between the PCU and the radiator. The pump will be the same as used in the dynamic model. Its mass is *22.68 kg*. This raises the total mass of the radiator system to *116.46 kg*.

4.3.5 Summary

The solar direct design consists of four primary components: the concentrator, the receiver and thermal storage, the PCU, and the radiator. These components were designed to provide the required power over the lifetime of the satellite with the minimum mass and area, satisfying the objectives defined in the Value System Design. Table 12 summarizes the calculated masses and areas for this design. An additional ten percent is added to the total mass to compensate for wiring, converters, regulators, and other items.

This concludes the Systems Modeling step of the Systems Analysis. The comparison of the two modeled designs with the current GPS IIR Electric Power System will complete the Systems Evaluation step. As mentioned earlier, the Decision Making and Implementation steps of the Systems Analysis process are beyond the authority of this project. The information contained within this report is intended solely to aid and inform potential decision makers responsible for the Electric Power System of the Navstar Global Positioning System satellites.

4.4 Comparison with GPS IIR

To complete the Systems Evaluation step of the Systems Analysis process, the two designs modeled in the previous sections are compared to one another and the current photovoltaic system used on the GPS IIR satellites. To meet the primary objective defined in the Value System Design, to *design the best GPS Electric Power System*, the design model with the minimum mass and area is desired.

Table 12 - Calculated Component Masses and Areas for Solar Direct Model Design

Component	*Mass (kg)*	*Area (m^2)*
Concentrator	66.64	11.07
Receiver/Thermal Storage	139.62	N/A
AMTEC PCU	8.29	N/A
Radiator/Pump	116.46	12.24
Sub-Total	331.01	23.31
Additional 10% Mass	33.10	N/A
Total	364.11	23.31

The masses and external surface areas of the designs are summarized in Table 13. The solar thermal design areas include the concentrator and radiator, whereas the GPS design area is composed of only the solar array panels. Recall that the values for the modeled designs are only approximate within some range, whereas the GPS values listed can be considered exact.

It is apparent from these results that the solar dynamic design model provides the required power output with the least mass and external area. The solar dynamic design mass is approximately 28% smaller than the GPS IIR mass. The solar dynamic area is

Table 13 - Design Mass and Area Comparisons (4:1)

System	*Mass (kg)*	*Δ %*	*Area (m^2)*	*Δ %*
GPS Photovoltaic	263.95	N/A	13.55	N/A
Solar Dynamic	190.78	-27.72	8.86	-34.61
Solar Direct	364.11	37.95	23.31	72.03

approximately 35% smaller than the GPS IIR area. The solar direct model has approximately 38% more mass and 72 % more area than the GPS design.

Calculations to determine the AMTEC efficiency required to make the solar direct model compatible with the GPS and solar dynamic designs are summarized in section 4.4.1. Calculations for the receiver/concentrator boom length required to prevent shading effects are discussed in section 4.5.

The mass and the external area are both very important aspects of the design of any satellite or satellite subsystem, but they do not completely describe the worthiness of a design. This project used the mass and area as measures to determine whether further research into this design is warranted. Other aspects such as the reliability, life expectancy, and life cycle cost play a major role in choosing a design.

While the solar thermal dynamic power system shows improvement in mass and area over photovoltaics, it has some drawbacks as well. The concentrator requires a high degree of pointing accuracy to focus the energy properly into the relatively small receiver aperture. This will have an effect on the design of the attitude control system and might require better sun sensors than currently used with photovoltaic solar arrays. If the radiator for the waste energy is located away from the concentrator, to avoid shading the concentrator, it may have a heating effect on the electronic components. The radiator would also have to have an edge face the sun to minimize the solar heating of the radiator. This would mean having a sun sensor on the radiator as well as the concentrator.

The lack of batteries inside the bus would, however, either make the main body of the satellite smaller or would make room for a secondary payload to be placed on-board. A smaller bus may also mean launching the satellites on smaller boosters or launching two at once. This could result in significant launch cost savings.

All of the above mentioned conditions, and others, need to be considered when deciding whether to pursue the new design modeled here. Other measurables defined in the Value System Design that were not included in these models need to be addressed and the interrelations between all of the measurables must be defined.

As mentioned earlier, this report does not attempt to make any decisions regarding the design of the electrical power system, it simply presents information to help a decision maker decide. The information does, however, show promise and further research into this area may be warranted.

4.4.1 Required AMTEC Efficiency

The solar direct modeled design using an 18% efficient AMTEC as a PCU was unable to produce the required energy without excessive mass and area values. This section looks at the efficiency of the AMTEC and determines the value required to keep the solar direct system from having a larger mass and area than the other designs.

Leaving the PCU efficiency as an unknown in the equations used throughout chapter four, the masses and areas of the various components are found as a function of the efficiency. The various equations, in their original and combined forms, used to determine the mass and area are given below as Equation (2) through Equation (10). Once the equations for total mass and area are known as functions of the PCU efficiency, they are set equal to the desired values, the GPS IIR or solar dynamic values, and solved for the required efficiency.

Setting Equation (9) equal to the area of the GPS IIR solar arrays, *13.55* m^2, and solving for the efficiency gives a value of 0.288. This means that if the AMTEC efficiency can be raised from 18% to 29%, the area of the solar concentrator and receiver would equal the area of the photovoltaic solar arrays. The estimated AMTEC efficiencies are between 15% and 25% (14:861), so 29% is not very far out of the projected range. The required efficiency to make the solar direct mass equal to the GPS IIR EPS mass, *263.95 kg*, is 0.256. Using the higher efficiency to ensure both the area and mass are no larger than the GPS values, the required AMTEC efficiency is 29%. Inserting this efficiency into Equations (9) and (10) gives an area of *13.55* m^2 and a mass of *237.35 kg*.

$$IO_{ratio} = 1.084 / [(0.905)(0.975)(0.802)(E_{PCU})] \tag{2}$$

$$A_{rad} = (1.1)^2 (1.084)(1 / E_{PCU} - 1)(P_{out}) / (\varepsilon \sigma T^4) \tag{3}$$

$$M_{rad} = A_{rad}(7.66kg / m^2) + 22.68kg \tag{4}$$

$$A_{conc} = 1.1 P_{out} IO_{ratio} / (1.353kW / m^2) \tag{5}$$

$$R_{conc} = (A_{conc} / \pi)^{1/2} \tag{6}$$

$$M_{conc} = A_{conc} 266kg / 254.47m^2 + R_{conc} 264kg / 9m \tag{7}$$

$$M_{rcvr} = 4.4[(P_{out} / 0.905 E_{PCU})t] / (1075kJ / kg) \tag{8}$$

$$\begin{aligned} A_{tot} &= A_{rad} + A_{conc} \\ &= [(4.68 / E_{PCU}) - 2.69]m^2 \end{aligned} \tag{9}$$

$$\begin{aligned} M_{tot} &= 1.1(M_{rad} + M_{conc} + M_{rcvr} + 8.29kg) \\ &= [(51.68 / E_{PCU}) + (25.697 / E_{PCU}{}^{1/2}) + 11.423]kg \end{aligned} \tag{10}$$

where

IO_{ratio} = Input to Output Power Ratio

E_{PCU} = PCU Efficiency

A_{rad} = Radiator Area (m^2)

P_{out} = Power Output = *1.6 kW*$_e$

ε = Radiator Emissivity

σ = Stefan-Boltzmann Constant

T = Radiator Temperature (K)

M_{rad} = Radiator Mass (kg)

A_{conc} = Concentrator Area (m^2)

R_{conc} = Concentrator Radius (m)

M_{conc} = Concentrator Mass (kg)

M_{rcvr} = Receiver/PCM Mass (kg)

t = Eclipse Time = *3360 s*

A_{tot} = Total Area (m^2)

M_{tot} = Total Mass (kg)

Setting Equation (10) equal to the solar dynamic mass *190.78 kg*, and solving for the efficiency gives a value of 0.376. This means that the AMTEC efficiency must be raised from 18% to 38% for the mass of the solar direct system to equal the solar dynamic mass. The required efficiency to make the solar direct area equal the solar dynamic area, $8.86\ m^2$, is 0.405. Because the required efficiencies are well outside the expected range, the lower efficiency, 38%, will be used. Inserting this efficiency into Equations (9) and (10) gives a mass of *190.78 kg* and an area of 9.76 m^2.

If the efficiency of the AMTEC PCU can be raised to 29%, which is within the projected range of efficiencies, the solar direct EPS modeled in this thesis would have a better mass and area than the current GPS photovoltaic EPS. An efficiency of 38% would be needed to make the direct EPS competitive with the dynamic system. The various masses and areas for the systems discussed, including the more efficient AMTECs, are summarized in Table 14.

Table 14 - System Mass and Area Summarization

System	*Mass (kg)*	*Area (m^2)*
GPS IIR	263.95	13.55
Solar Dynamic	190.78	8.86
Solar Direct (18%)	364.11	23.31
Solar Direct (28.8%)	237.35	13.55
Solar Direct (37.6%)	190.78	9.76

4.5 Boom Length Calculations

A boom is required to allow the receiver and concentrator to have the capability of tracking the sun throughout the satellite's orbit. The receiver is placed at the end of the boom, with the concentrator off the receiver. Figure 4 shows a schematic of the boom, receiver, and concentrator designed. The boom, similar to the solar arrays, is able to rotate 360 degrees about its own axis. The receiver is connected to the boom through a gimbaled joint that allows two degrees of freedom. The receiver can not move away from the joint, but can rotate to any angle, as long as the concentrator does not intersect the boom.

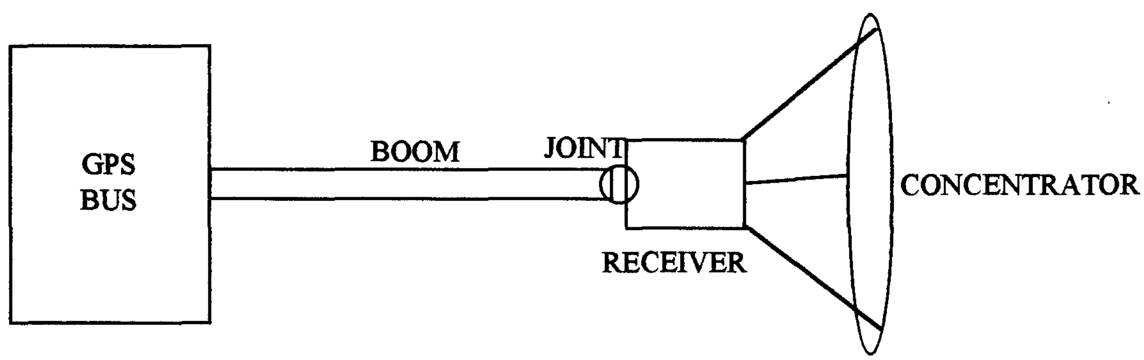


Figure 4 - EPS Boom Configuration (Not to Scale)

As can be seen in Figure 4, the satellite may prevent the sunlight from illuminating the concentrator, creating an artificial eclipse. To prevent this happening, the boom is made to rotate through a certain angle away from the satellite normal. The angle required depends on the size of the satellite, the concentrator radius, and the boom length. The greater the angle required, the more complex the boom design and the greater the impact on other subsystems, particularly the attitude control.

The satellite can be approximated as a square with side length of *1.30 m*, for shading effects (4:4.4-7). This includes the various payload antennas, propulsion elements, and other protrusions from the main satellite body. Using the solar dynamic design, the concentrator radius is *1.25 m*. To allow the full radius to clear the edge of the satellite, the end of the boom must be *2.55 m* from the center of the satellite.

Using simple geometry, the relationship between the boom length and the angle away from the satellite normal is determined to be:

$$\sin\alpha = 2.55 / L \tag{11}$$

where

α = Boom Angle away from normal (*rad*)
L = Boom Length (*m*)

Plotting the length versus the angle (Equation 11) gives the graph shown in Figure 5. In an effort to minimize both the angle and the boom length, the point closest to the origin should be chosen. This would seem to indicate a boom length of approximately *12 m* and an angle of *0.21 radians* or *12.3 degrees*.

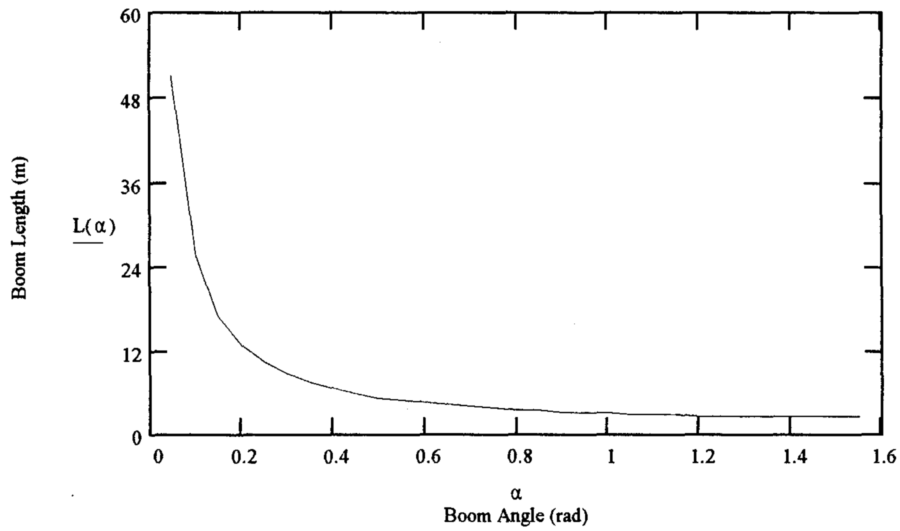


Figure 5 - Boom Angle vs. Boom Length

Analyzing these values, it becomes apparent that more weight should be placed on minimizing the length of the boom, twelve meters is much too long. A summary of boom lengths and their corresponding angles is included below as Table 15.

A greater angle corresponds with a shorter length, until the point where the angle is π *rad* and the boom length is *2.55 m*. If we arbitrarily decide that the boom angle should not exceed $\pi/2$ *rad* to allow proper connections with the satellite, then the length required to allow the concentrator to see around the satellite body is *3.6 m*.

The requirement of having a boom to prevent the satellite from shadowing the concentrator will affect the mass of the overall power system. Many different lightweight but sturdy materials are available for boom production. The choice of materials and

Table 15 - Boom Lengths and Angles

Boom Length (m)	*Boom Angle (deg)*
12.0	12.27
9.0	16.46
6.0	25.15
5.1	30.00
3.6	45.00
3.0	58.21

construction design for the boom are not addressed in this thesis. Further systems analysis on the structural system of the satellite would be required, therefore no further work in this area is done.

V. Conclusion

5.1 Summary

This research project investigated the redesign of the electrical power system of the Global Positioning System satellites. A Systems Analysis procedure was applied to the various power sources and conversion processes available, not including photovoltaic arrays. The most appropriate combinations of power source and conversion process were identified for the satellite, orbit, and power needs being investigated. Solar thermal designs, with either a dynamic or direct conversion process, topped the list. Further analysis identified the Free-Piston Stirling Engine as the best dynamic conversion cycle, and the Alkali Metal Thermal-to-Electric Converter as the best direct converter to work with the solar thermal power source.

Designs for the two power systems were modeled to provide the same steady-state power output as the current GPS IIR system. To meet the primary objective defined in the Value System Design, to *design the best GPS Electric Power System*, the design model with the minimum mass and area for the same power output is desired. The solar dynamic model showed a 28% improvement in mass and a 35% improvement in area over the GPS IIR design, while the solar direct model had worse mass and area values than the GPS design. All designs produced $1.6\ kW_e$ steady state power.

The relationship between the solar direct model's mass and area values and the efficiency of the AMTEC PCU was determined. The efficiencies required to make the solar direct system competitive with the GPS IIR and solar dynamic systems were determined.

An AMTEC efficiency of 29% would be needed to make the solar direct system comparable to the GPS IIR system. This is approximately 11% greater than the designed efficiency. The external area would be equal and the mass would be approximately 10% less than the GPS IIR design. The required efficiency to compare with the solar dynamic model was found to be 38%. This efficiency would make the masses equal but the external area of the solar direct system would be approximately 10% greater than the solar dynamic system.

A quick analysis of the boom required to prevent shadowing of the concentrator was performed. Assuming a maximum boom angle of $\pi/2$ *rad* from the satellite normal, the boom length would need to be *3.6 m*.

Other aspects of the designs were addressed, including the impact on other satellite subsystems. The results, however, indicate that a solar dynamic electrical power system for GPS satellites may prove beneficial, and further investigation into this area is warranted and highly recommended.

5.2 Recommendations

This design study shows the potential for a solar dynamic electrical power system for GPS satellites. Further research into the following areas relating to this area is highly recommended.

- Integrated Power and Attitude Control System (IPACS) effect on mass and volume of satellite.
- Smaller, less massive liquid metal pumps or heat pipe configurations for heat transport to the radiator.
- Life-cycle cost analysis of redesigning the satellite and implementing a solar dynamic electrical power system.
- Reliability, maintainability, and safety of solar dynamic power system on GPS.
- Solar dynamic power system's effect on all other subsystems of GPS.
- Actual schematic design and interface analysis of new GPS satellite with solar dynamic power system.
- Effects of power system redesign on ground support and users.

Appendix: Power Conversion Devices

This appendix contains detailed descriptions of the various power conversion devices examined for use in the systems analysis. The first section covers dynamic systems, primarily heat cycle engines. The second section covers direct energy converters.

A.1 Dynamic Conversion Systems (7:75-88)

A dynamic conversion system uses a thermodynamic cycle, expanding and compressing a fluid, to provide the mechanical work necessary to generate electricity in a generator or alternator. Dynamic systems include Brayton and Rankine cycles, and Stirling Engines. Before any detailed discussion of thermodynamic cycles can be accomplished, the ideal, or Carnot, cycle must first be introduced.

A.1.1 Carnot Cycle

Before a discussion of heat cycle engines can be properly addressed, an ideal cycle must first be defined. The Carnot cycle is a fundamental theoretical concept because the thermal efficiency of this cycle is the maximum possible for any heat engine operating between the same two temperature limits. In this idealized cycle, a working fluid experiences: reversible, isothermal heat addition from state 1 to state 2; reversible adiabatic expansion from 2 to 3 during which work is extracted; reversible isothermal heat rejection from state 3 to 4; and finally a reversible, adiabatic compression process from 4 to 1, in which work is required from the surroundings. Note that a reversible, adiabatic process is defined as isentropic. Figure 6 shows the pressure-volume and temperature-entropy diagrams for the Carnot cycle.

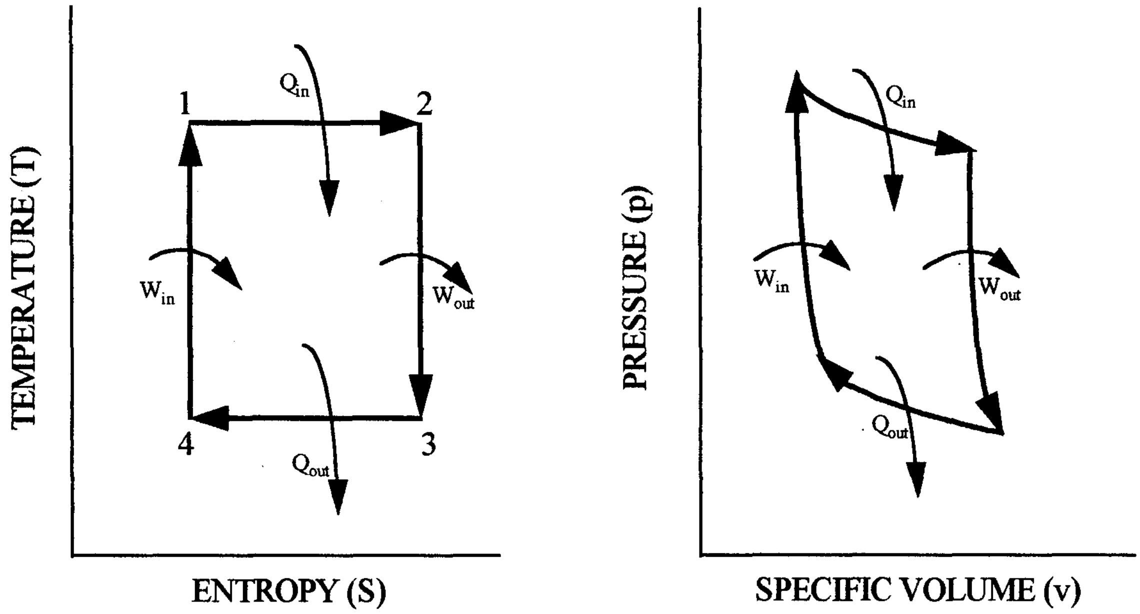


Figure 6 - T-S and P-V Diagrams for Carnot Cycle (7:76)

All of the cycles discussed here are considered ideal for the purpose of their explanation. A truly ideal cycle does not exist. Friction, material imperfections, and other losses reduce the overall efficiency of the cycle.

A.1.2 Brayton Cycle

The stages of an ideal Brayton cycle are: constant pressure heat addition from state 1 to 2; isentropic expansion in the turbine from 2 to 3; constant pressure heat rejection from state 3 to 4; and isentropic compression from 4 to 1 (Figure 7). The fluid expansion in the turbine drives the turbine shaft which in turn drives a generator for producing electricity. To improve the thermal efficiency of the basic closed Brayton cycle, a regenerator may be used. The regenerator uses the heat from the turbine (state 3) to preheat the working fluid before it enters the heat source (state 1), thus requiring less heat input and reducing the amount of heat rejected.

The working fluid of a closed Brayton cycle is typically an inert gas to reduce the erosion of turbine blades caused by liquids and reactive gases. Materials with lower molecular weights have better thermal energy transport properties, but require more turbine stages, than those with higher weights. A mixture of helium and xenon, with a molecular weight of approximately 40, appears to be the favorite working fluid for closed Brayton cycles. A schematic and the temperature-entropy diagram for an ideal closed Brayton cycle are shown in Figure 7.

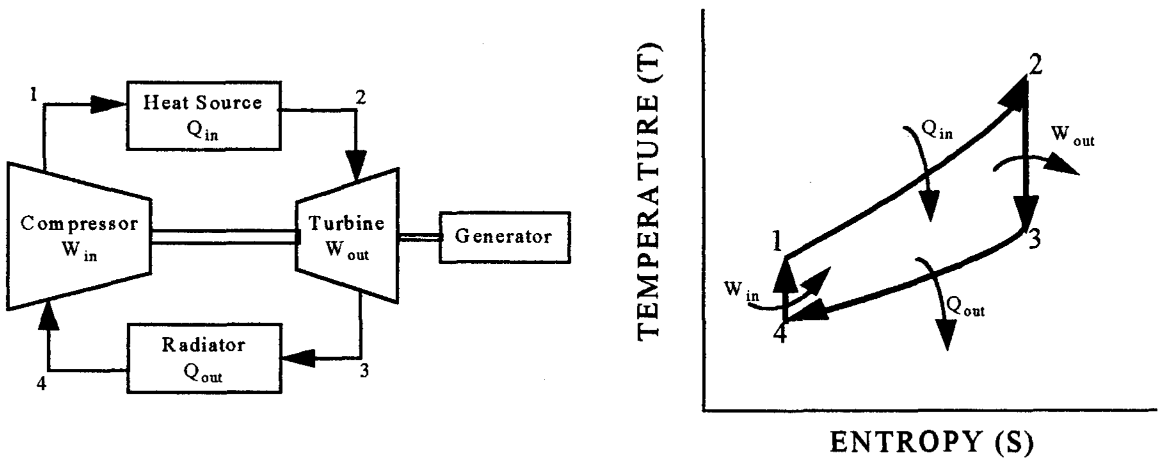


Figure 7 - Schematic and T-S Diagram for Ideal Closed Brayton Cycle (7:82)

A.1.3 Rankine Cycle

The Rankine cycle (Figure 8) differs from the Brayton cycle in that the working fluid, at times, is in the liquid state. Also, the heat rejection and addition are done at constant temperature, not constant pressure. The stages of the ideal Rankine cycle are: isentropic expansion in the turbine from state 1 to 2; isothermal heat rejection in the condensing radiator (where the vapor becomes a liquid) from 2 to 3; isentropic compression in a pump from state 3 to 4; and constant pressure heat addition from 4 to 1. At state 4′, the liquid under goes a phase change to a vapor, so the additional heat does not increase the temperature or pressure of the fluid, just changes the phase. The fluid expansion in the turbine again drives the turbine shaft which in turn drives a generator for producing electricity.

A schematic and the temperature-entropy diagram for an ideal basic Rankine cycle are shown in Figure 8.

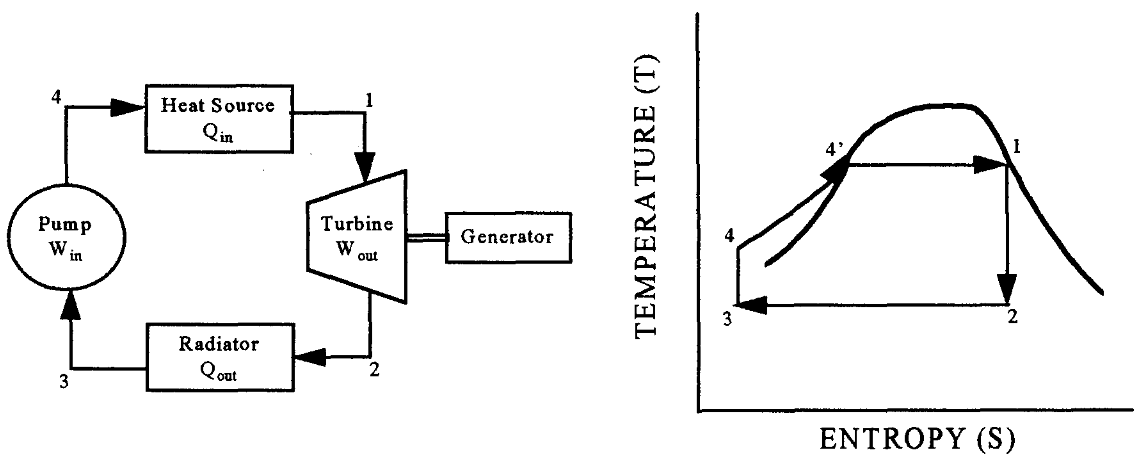


Figure 8 - Schematic and T-S Diagram for Ideal Rankine Cycle (7:79)

A.1.4 Stirling Engine

A reversible heat engine requires a means to transfer, isothermally and reversibly, all thermal energy to and from the system. The working fluid of an ideal Stirling engine imparts its thermal energy to a regenerator while going from high to low temperature. When the fluid returns to the regenerator at the low temperature, it regains its energy and returns to the original high temperature.

The pressure-volume and temperature-entropy diagrams for a basic ideal Stirling engine are shown in Figure 9. The stages of the cycle are: reversible, constant volume heating from the low temperature limit, state 1, to the high limit, state 2, through the regenerator; reversible heat addition (from heat source) and isothermal expansion from 2 to 3; constant volume cooling in the regenerator from the high temperature state 3 to the low temperature state 4; isothermal compression and heat rejection from 4 to 1.

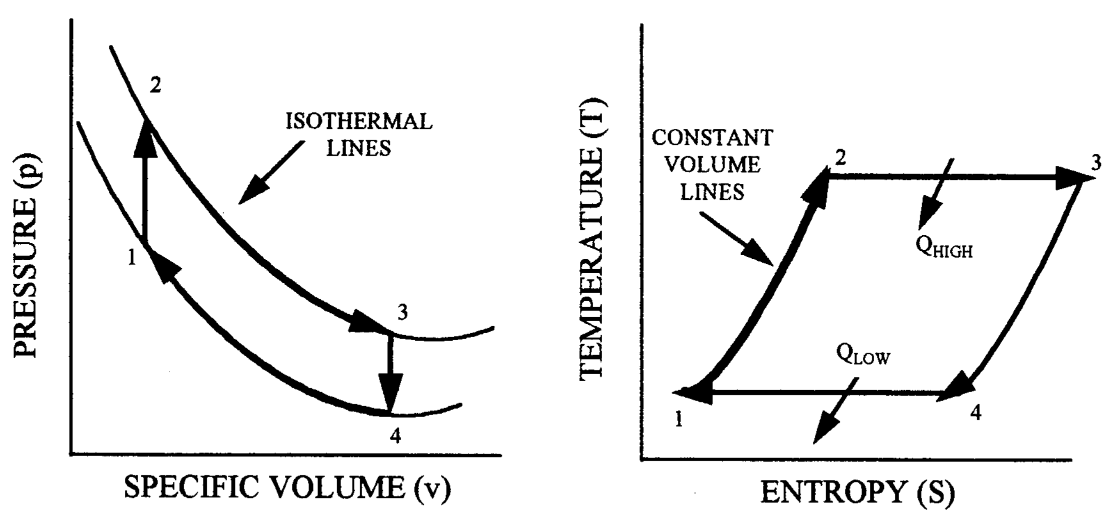


Figure 9 - P-V and T-S Diagrams for Ideal Stirling Cycle (7:86)

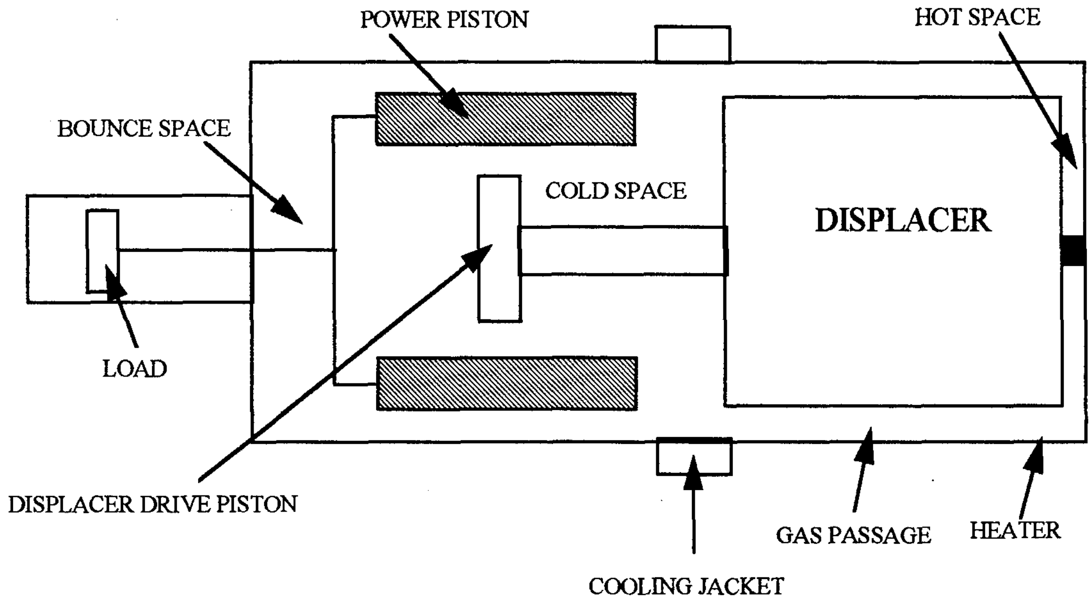
The free piston Stirling engine (FPSE) is a thermally driven mechanical oscillator using gas pressure rather than mechanics to produce motion. This type of engine operates at the highest device efficiency of all known heat engines and is ideal for coupling to a linear generator. The FPSE consists of three basic components: a heavy power piston, a low mass displacer piston, and a sealed cylinder. The operating principles of a Stirling engine are described in Figure 10.

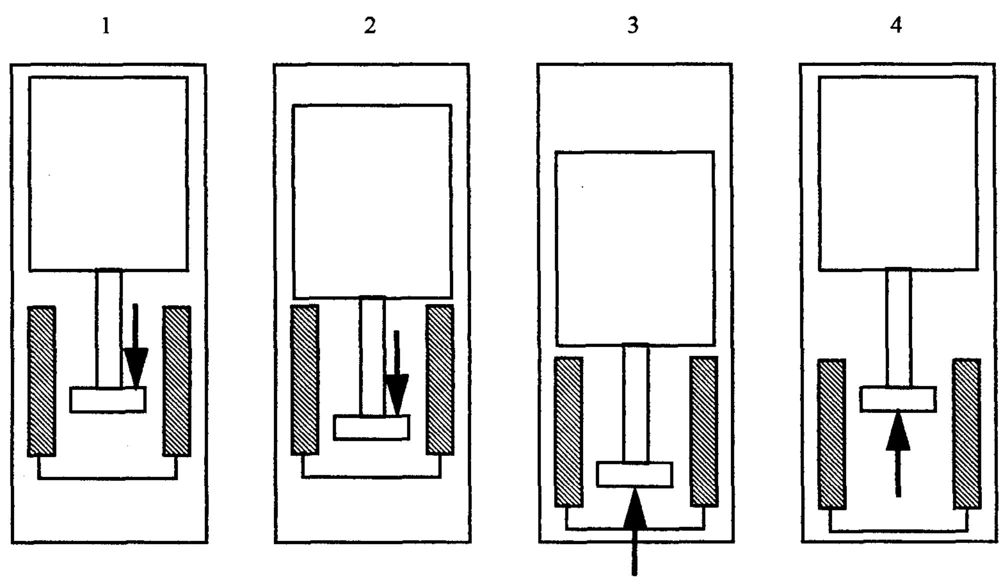
A.2 Direct Energy Conversion Systems

Direct Energy Conversion (DEC) systems use thermophysical principles to convert heat into electricity with no moving mechanical parts (7:75). Typical DECs include thermoelectric, thermionic and magnetohydrodynamic systems.

A.2.1 Thermoelectric System (7:88-93)

A typical thermoelectric converter consists of two semiconductor legs that are bonded to two heat transfer surfaces, called the hot and cold shoes or junctions. One of the semiconductor materials is a *p*-type material, the other is an *n*-type. The temperature gradient between the hot and cold shoes drives electrons in the *n*-type material and positively charged holes in the *p*-type material toward the cold end. The thermally driven flow of electrons and holes creates a voltage across the cold shoe plates. Connecting an external load across the two cold shoe plates causes a current to flow through the external circuit (Figure 11). The power flowing through the external circuit is maximized when the load resistance is matched to the internal thermoelectric converter resistance. Numerous





1-2 Displacer driven toward cold space by pressure difference, working space pressure higher than bounce space pressure.

2-3 Piston expands working gas; displacer on piston.

3-4 Displacer driven toward hot space by pressure difference, bounce space pressure greater than working space.

4-1 Piston driven into working space by higher bounce space pressure.

Figure 10 - Operating Principles of Free Piston Stirling Engine (7:87)

external thermoelectric couples can then be connected in an external series-parallel circuit to provide redundancy against open circuit failures. Figure 11 illustrates the basic operating principles of a thermoelectric conversion device.

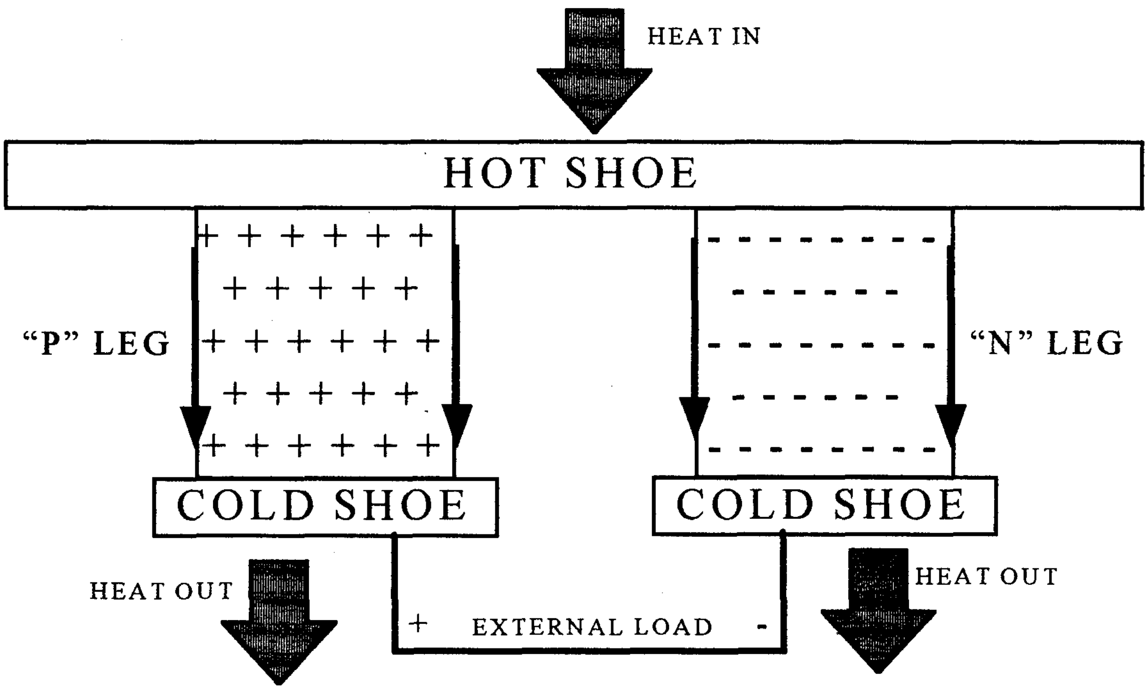


Figure 11 - Operating Principle of Thermoelectric Converter (7:89)

A.2.1.1 AMTEC (16:855-856)

An alternative thermoelectric design is an Alkali Metal Thermoelectric Converter or AMTEC (Figure 12). An AMTEC utilizes β''-alumina solid electrolyte (BASE) as a conductor of sodium ions and an insulator for electrons. The BASE is wedged between an anode and cathode, so that the sodium working fluid contacts the anode first. The front side of the BASE oxidizes the sodium and diverts the electrons. The sodium ions pass through the BASE, then are reduced by the diverted electrons, returning to their original neutral state. The sodium releases heat through the radiator, is pumped to the heat source, gains heat and energy, then repeats the cycle. Electrical power is produced when an external load is applied across the electrodes. Figure 12 shows the basic operation of an AMTEC. Typical high temperatures range from *900 K* to *1200 K* with heat rejection at *400 K* to *800 K*.

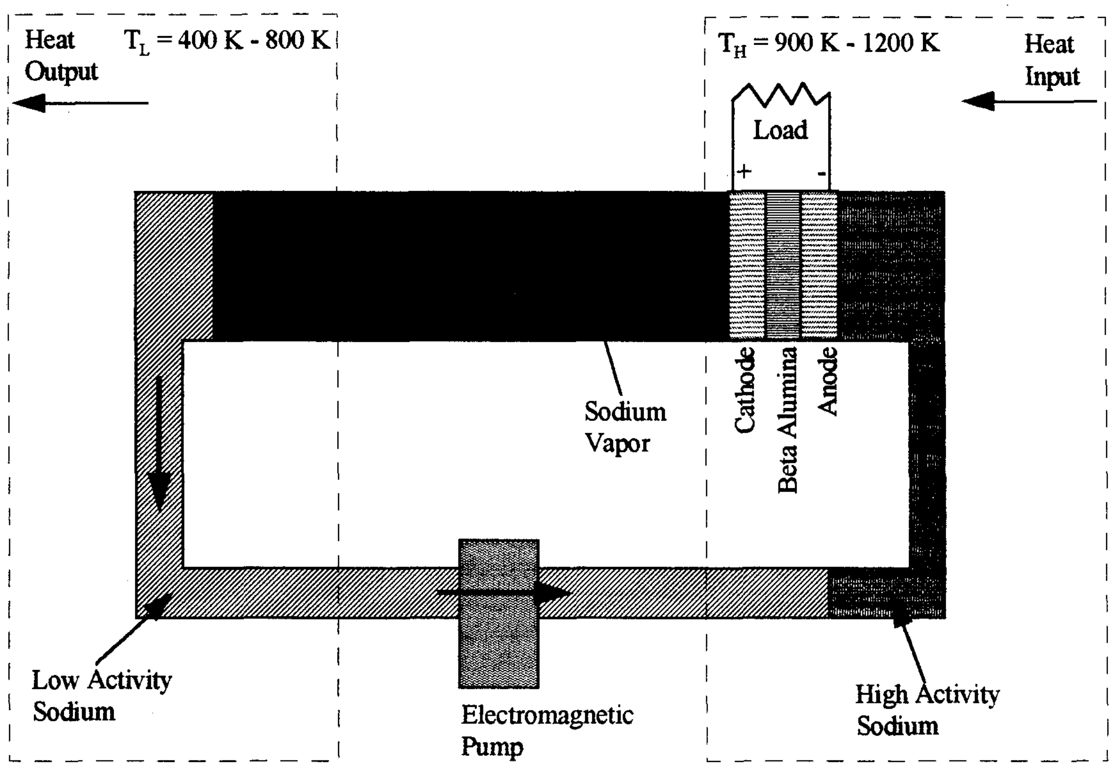
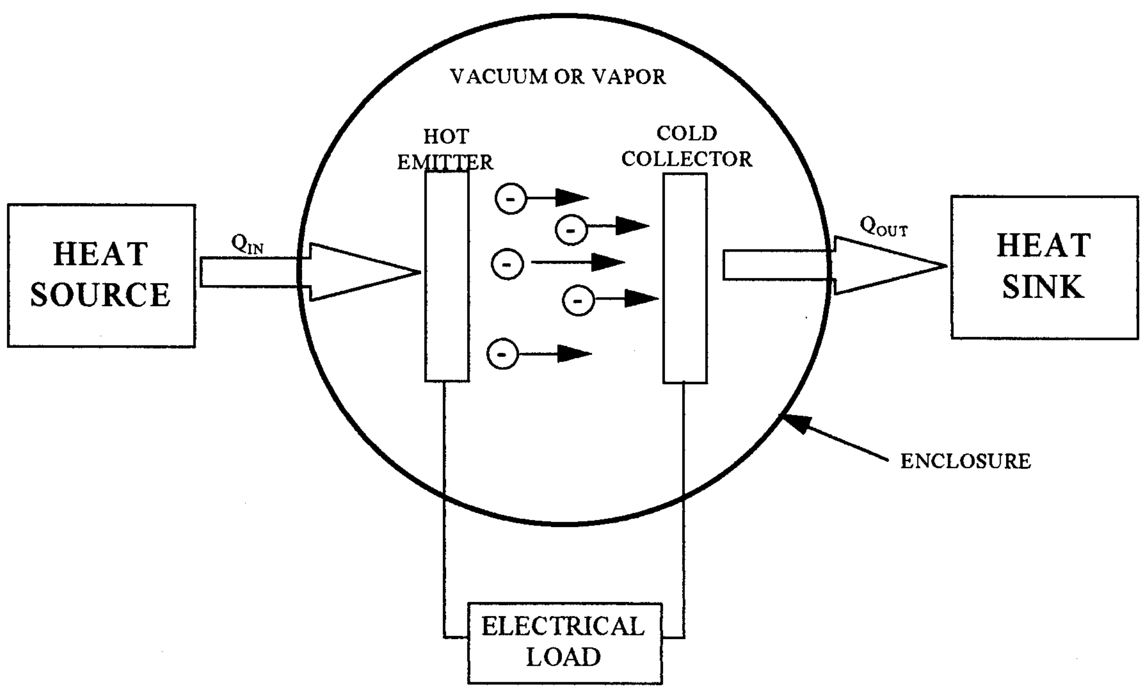


Figure 12 - AMTEC Cycle Diagram (16:855)

A.2.2 Thermionic System (7:93-96)

A thermionic converter operates by transferring electrons from a hot emitter surface, across a very small interelectrode gap, to a cooler collecting surface. Typical temperatures for the emitter and collector are approximately *1800 K* and *1000 K*, respectively. Connecting an external load across the voltage potential between the emitter and collector, or cathode and anode, will produce a current. Figure 13 depicts the operating principles, typical operating parameters, and potential component materials.



TYPICAL OPERATING REGIME		MATERIALS	
EMITTER TEMPERATURE:	1600 - 2000 K	EMITTER MATERIALS:	W, Re, Mo
COLLECTOR TEMPERATURE:	800 - 1100 K	COLLECTOR MATERIALS:	Nb, Mo
ELECTRODE EFFICIENCY:	UP TO 20%	INSULATOR MATERIALS:	Al_2O_3, Al_2O_3/Nb CERMET
POWER DENSITY:	1 - 10 W/cm^2	ELECTRODE ATMOSPHERE:	Cs at 1 Torr

Figure 13 - Thermionic Converter, Parameters, & Materials (7:94)

A.2.3 Magnetohydrodynamic System (17:17-18)

The principle of a magnetohydrodynamic (MHD) system is based on Faraday's discovery that an electromotive force is induced in a circuit when the flux of induction through it is changed. The basic process of a MHD generator, like most electromagnetic generators, involves passing a conductor through a magnetic field. Unlike most conventional generators, though, a MHD generator uses an ionized gas as the conductor. When this gas is passed through a magnetic field, an emf is produced mutually perpendicular to the magnetic field direction and the direction of gas flow. If electrodes are placed in suitable positions and an external load is placed across the electrodes, a current will be produced. A very simple schematic of a MHD generator is shown in Figure 14, below.

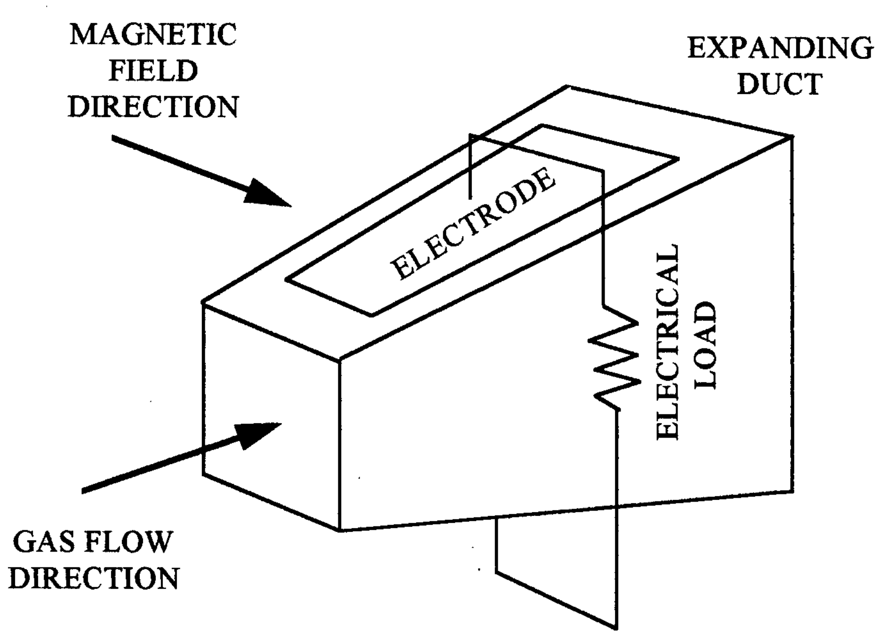


Figure 14 - Schematic of Magnetohydrodynamic Generator (17:18)

Bibliography

1. Snyder, Nathan. Space Power Systems, Progress in Astronautics and Rocketry (Vol. 4). New York: Academic Press, 1961.

2. Air University. Space Handbook, An Analysts Guide (Vol. 2). AU-18. Maxwell Air Force Base, AL: Air University Press, 1993.

3. Wilson, Andrew. Jane's Space Directory. Alexandria, VA: Jane's Information Group, 1993.

4. Martin Marietta Corporation. Navstar Global Positioning System (GPS) Replenishment Satellites (Block IIR) Phase II - Orbital Operations Handbook (OOH) (Vol. 1, Sections 4.3 and 4.4). Philadelphia: 1994.

5. Pintz, Castle, Reimer, and Naujokas. "Design of an Auto-Deployable Technology Demonstration Solar Concentrator." Proceedings of the 24th Intersociety Energy Conversion Engineering Conference (Vol. 2). 867-873. New York: IEEE Press, 1989.

6. Beninga, Kelly and Davenport, Roger. "An Advanced Concentrator for Solar Dynamic Power Systems in Space." Proceedings of the 24th Intersociety Energy Conversion Engineering Conference (Vol. 2). 875-880. New York: IEEE Press, 1989.

7. Angelo, Joseph A, Jr. and Buden, David. Space Nuclear Power. Malabar, FL: Orbit Book Company, 1985.

8. Gondhalekar, Downer, Eisenhaure, Hockney, and Johnson. "Low Noise Spacecraft Attitude Control Systems." Proceedings of the 26th Intersociety Energy Conversion Engineering Conference (Vol. 4). 244-249. New York: IEEE Press, 1991.

9. Kramer, Stuart. Class lectures and handouts, SENG 520, Systems Analysis for Design. School of Engineering, Air Force Institute of Technology, Wright-Patterson AFB OH, Fall Quarter 1994.

10. Athey, Thomas H. Systematic Systems Approach - An Integrated Method for Solving Systems Problems. Englewood Cliffs NJ: Prentice-Hall, 1982.

11. Pisacane, Vincent L. and Moore, Robert C. Fundamentals of Space Systems. New York: Oxford University Press, 1994.

12. Wallin, Wayne E. and Friefeld, Jerry M. Solar Dynamic Power System Definition Study. Contract N88-20361. Cleveland, OH: NASA Lewis Research Center, March 1988.

13. Angrist, Stanley W. Direct Energy Conversion (Fourth Edition). Boston: Allyn and Bacon, 1982.

14. Ivanenok, Joseph F. "High Power Density AMTEC Systems." Proceedings of the 28th Intersociety Energy Conversion Engineering Conference (Vol. 1). 861-865. New York: IEEE Press, 1993.

15. Larson, Wiley J. and Wertz, James R. Space Mission Analysis and Design (Second Edition). Torrance, CA: Microcosm, 1995.

16. Underwood, Suitor, Williams, Ryan, Jeffries-Nakamura, and O'Connor. "A Five Volt AMTEC Multicell." Proceedings of the 28th Intersociety Energy Conversion Engineering Conference (Vol. 1). 855-859. New York: IEEE Press, 1993.

17. Coombe, R. A. Magnetohydrodynamic Generation of Electrical Power. New York: Reinhold Publishing, 1964.

Vita

Captain Kevin J. Walker [redacted] He graduated in the top 1% at Hillcrest High School in 1986 and entered undergraduate studies and the Reserve Officers Training Corps (ROTC) at the University of Illinois in Champaign-Urbana. Captain Walker graduated with a Bachelor of Science degree in Aeronautical/Astronautical Engineering on 13 May 1990 and received his reserve commission on 14 May 1990. His first assignment was to Onizuka Air Station in Sunnyvale, California as a Space Operational Plans Officer for the 750th Operations Support Squadron and later a Flight Director for the 5th Space Operations Squadron. In May 1994, Captain Walker entered the Air Force Institute of Technology's School of Engineering at Wright-Patterson AFB. Upon completion of his Masters Degree in Space Operations - Space Systems Engineering, Captain Walker will be assigned to Headquarters, Air Force Space Command at Peterson AFB, Colorado.

REPORT DOCUMENTATION PAGE

Form Approved OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*

2. REPORT DATE
December 1995

3. REPORT TYPE AND DATES COVERED
Master's Thesis

4. TITLE AND SUBTITLE
SYSTEMS ANALYSIS OF GPS ELECTRICAL POWER SYSTEM REDESIGN

5. FUNDING NUMBERS

6. AUTHOR(S)
Kevin J. Walker, Capt, USAF

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
Air Force Institute of Technology
WPAFB OH 45433-6583

8. PERFORMING ORGANIZATION REPORT NUMBER
AFIT/GSO/ENY/95D-03

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
Capt Rick Koon
HQ AFSPC/DRFN
150 Vandenberg St Ste 1105
Peterson AFB CO 80914-4590

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Approved for public release; distribution unlimited

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

A systematic analysis is applied to the electrical power subsystem of the Global Positioning Satellite (GPS) system. Results determined the most appropriate power source and conversion system options. Photovoltaic solar arrays, the current GPS power system, were not included in the analysis. The best electrical power subsystem options found in the analysis include a solar power source with either a dynamic or direct conversion technique, and a direct conversion nuclear source. The two solar options are designed, at a low level of detail, to provide the same level of power the current GPS photovoltaic solar array system provides. These two designs are then compared with the current system, stressing mass and area. Results show that the solar dynamic design has approximately 28% less mass and approximately 35% less area than the GPS IIR design. The solar direct model has approximately 38% more mass and 72% more area than the GPS design.

14. SUBJECT TERMS
systems analysis, systems engineering, space power systems, power generation, thermal energy storage, solar concentrator

15. NUMBER OF PAGES
86

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	UL

GENERAL INSTRUCTIONS FOR COMPLETING SF 298

The Report Documentation Page (RDP) is used in announcing and cataloging reports. It is important that this information be consistent with the rest of the report, particularly the cover and title page. Instructions for filling in each block of the form follow. It is important to ***stay within the lines*** to meet ***optical scanning requirements***.

Block 1. Agency Use Only *(Leave blank)*.

Block 2. Report Date. Full publication date including day, month, and year, if available (e.g. 1 Jan 88). Must cite at least the year.

Block 3. Type of Report and Dates Covered. State whether report is interim, final, etc. If applicable, enter inclusive report dates (e.g. 10 Jun 87 - 30 Jun 88).

Block 4. Title and Subtitle. A title is taken from the part of the report that provides the most meaningful and complete information. When a report is prepared in more than one volume, repeat the primary title, add volume number, and include subtitle for the specific volume. On classified documents enter the title classification in parentheses.

Block 5. Funding Numbers. To include contract and grant numbers; may include program element number(s), project number(s), task number(s), and work unit number(s). Use the following labels:

C	- Contract	**PR**	- Project
G	- Grant	**TA**	- Task
PE	- Program Element	**WU**	- Work Unit Accession No.

Block 6. Author(s). Name(s) of person(s) responsible for writing the report, performing the research, or credited with the content of the report. If editor or compiler, this should follow the name(s).

Block 7. Performing Organization Name(s) and Address(es). Self-explanatory.

Block 8. Performing Organization Report Number. Enter the unique alphanumeric report number(s) assigned by the organization performing the report.

Block 9. Sponsoring/Monitoring Agency Name(s) and Address(es). Self-explanatory.

Block 10. Sponsoring/Monitoring Agency Report Number. ***(If known)***

Block 11. Supplementary Notes. Enter information not included elsewhere such as: Prepared in cooperation with...; Trans. of...; To be published in.... When a report is revised, include a statement whether the new report supersedes or supplements the older report.

Block 12a. Distribution/Availability Statement. Denotes public availability or limitations. Cite any availability to the public. Enter additional limitations or special markings in all capitals (e.g. NOFORN, REL, ITAR).

DOD - See DoDD 5230.24, "Distribution Statements on Technical Documents."
DOE - See authorities.
NASA - See Handbook NHB 2200.2.
NTIS - Leave blank.

Block 12b. Distribution Code.

DOD - Leave blank.
DOE - Enter DOE distribution categories from the Standard Distribution for Unclassified Scientific and Technical Reports.
NASA - Leave blank.
NTIS - Leave blank.

Block 13. Abstract. Include a brief *(Maximum 200 words)* factual summary of the most significant information contained in the report.

Block 14. Subject Terms. Keywords or phrases identifying major subjects in the report.

Block 15. Number of Pages. Enter the total number of pages.

Block 16. Price Code. Enter appropriate price code *(NTIS only)*.

Blocks 17. - 19. Security Classifications. Self-explanatory. Enter U.S. Security Classification in accordance with U.S. Security Regulations (i.e., UNCLASSIFIED). If form contains classified information, stamp classification on the top and bottom of the page.

Block 20. Limitation of Abstract. This block must be completed to assign a limitation to the abstract. Enter either UL (unlimited) or SAR (same as report). An entry in this block is necessary if the abstract is to be limited. If blank, the abstract is assumed to be unlimited.