Spring 1996

A Total Systems analysis Method for the Conceptual design of Spacecraft: An application to Remote Sensing Imager Systems

Knut I. Oxnevad
Old Dominion University

A TOTAL SYSTEMS ANALYSIS METHOD
FOR
THE CONCEPTUAL DESIGN OF SPACECRAFT:
AN APPLICATION TO REMOTE SENSING IMAGER SYSTEMS

by

Knut I. Øxnevad
Siviløkonom, December 1984, Norges Handelshøyskole

A Dissertation submitted to the Faculty of Old Dominion University in Partial Fulfillment of the Requirement for the Degree of

DOCTOR OF PHILOSOPHY

ENGINEERING MANAGEMENT

OLD DOMINION UNIVERSITY
May 1996

Approved by:

Laurence D. Richards (Director)

Griffith McRee (Member)

Derya A. Jacobs (Member)

Frederick Steier (Member)

Abstract

A TOTAL SYSTEMS ANALYSIS METHOD
FOR THE CONCEPTUAL DESIGN OF SPACECRAFT:
AN APPLICATION TO REMOTE SENSING IMAGER SYSTEMS

Knut I. Øxnevad
Old Dominion University, 1996
Director: Dr. Laurence D. Richards

Increased emphasis is being placed on improving the performance of space projects, within tighter budgets and shorter development times. This has led to a need for more efficient space system design methods. The research described here represents an effort to develop and evaluate such a method.

Systems engineering and concurrent engineering together provide the theoretical foundation for the method. The method, derived from both this theoretical foundation and ideas from experts in the space industry, emphasizes a **total systems analysis approach**, taking into account given mission requirements, and the **mathematical modeling of interactions** between system variables and between subsystems. The emphasis makes it possible to apply the method for effectively sizing and configuring the full space project, its subsystems, and its variables.

Size and configuration issues are especially important in the early conceptual design stages. The focus of this research and the developed method was, therefore, put on facilitating the design decisions taking place during those design stages. Mass, as a proxy for cost, was selected as the evaluation and optimization criterion. To make the method practical, LabVIEW was selected for developing the total systems analysis model.

LabVIEW is a graphical programming language that is easy to learn, program, modify, and run: and, it has a good user interface. These characteristics make it well suited

for rapid model development and for performing the large number of analysis runs required in the early conceptual design stages. The method was demonstrated for a V/IR (Visual/Infrared) space based Earth observation system. The mathematical model describing the interactions in this system was developed in close cooperation with subsystem specialists, primarily at NASA Langley Research Center, making it as realistic as possible. The model includes some 300 variables and 130 equations, and uses 1.7 MB of code.

The demonstration, focusing on size and configuration issues, showed how the method and model could be used for better understanding of model dynamics, for evaluating alternative technologies, for detecting technology limits, for performing inter-subsystem analyses, and for suggesting new technology developments.

It is hoped that this research will encourage engineers and project managers in the space sector to apply the developed design method to other types of space projects.

It is with great pride that I am dedicating this work to my parents and to the memory of a missed and very special physics and science teacher, Berit Skailand. To my parents for encouraging my independent and inquisitive mind and for supporting my many endeavors. To Berit Skailand for being a great teacher and a friend and for nurturing my interests in physics and the sciences. Together they helped me build the foundation from which this research grew.

ACKNOWLEDGMENTS

A number of people and institutions contributed at different levels to making this research a reality. I am deeply grateful to them all: Dr. Pål G. Bergan at Veritas Research for pointing me in the right direction; Bjørn Landmark and Georg Rosenberg at the Norwegian Space Center for sponsoring my involvement with the International Space University (ISU); Alumni and faculty of ISU for being bottomless sources of inspiration; Barney Roberts at the NASA Johnson Spaceflight Center/Futron for supporting my ideas from their early stages; Gary Price and Lee Rich at the NASA Langley Research Center for having the courage to help provide funding for this research. Without their unbending support this research would not have been possible; Eric L. Dahlstrom for providing invaluable input, feedback, and support throughout my research; Edwin B. Dean at the NASA Langley Research Center for encouraging and supporting this research; Jim Johnsen, Dr. Steven S. Katzberg, and George Ganoe at the NASA Langley Research Center for spending numerous hours discussing remote sensing satellite systems; Dr. Larry D. Richards for being my advisor and for spending countless hours discussing, defining, and working through all aspects of this research; Dr. Griffith McRee for his many ideas and for always having an open door and for being interested in my work; and last but not least Dr. Derya A. Jacobs and Dr. Frederick Steier for their many suggestions.

TABLE OF CONTENTS

LIST OF FIGURES

1. INTRODUCTION

1.1. Background, Objectives, and Limitations

As space technology moves from being experimental to mature, technology focus shifts from **technology development** to **technology utilization**. As this happens, space mission emphasis shifts from performance only, to a combination of performance, cost, and development time. To facilitate these shifts, the design process has to move from a subsystems orientation to a total systems orientation, and from being a sequential process to being a parallel one.

In an environment where the emphasis is on developing new technologies and concepts, the complications within each subsystem force the designers to concentrate their efforts within their own subsystem with less attention being paid to the interactions between the subsystems. Typically, subsystems, in this environment, are designed independently, and often in a sequential manner. Consequently, a total systems analysis for system sizing and configuration can only be performed at the later stages of the design process. At those stages, any required subsystem changes will require significant and time-consuming modifications. However, when performance of the final product is the major concern, these time consuming iterations are acceptable.

Using this approach, only a limited number of system designs can be analyzed. The approach is, therefore, often termed a point design approach. But again, this is acceptable when the focus is on making the technologies work rather than on utilizing them in an optimal manner to achieve a set of mission objectives. This changes as technology utilization, development time, and total cost become the main concerns. At this stage, the technologies have developed to a higher level of maturity and are expected to perform to given specifications. The main challenge of the designers, therefore, is to utilize the available technologies through **sizing and configuring the subsystems and the total system to meet mission objectives**. These mission objectives have to be met in the best possible way on time and within budget.

Time consuming re-designs of the complete system at the final stages of the design process, therefore, have to be replaced by continuous design iterations and minor design changes, starting from the early conceptual design stages, that gradually and rapidly move the design through its continuously decreasing design space closer to an "optimal" point. The aim of the research reported here has been to develop a practical method for doing this. The specific objectives of the research have been:

1. To develop a method based on a solid theoretical foundation, encapsulating central issues highlighted by experts in the space industry and emphasizing a total systems modeling approach for evaluating and optimizing space system designs. The method should focus on the requirements of the early conceptual stages of the design process, and it should include a total systems evaluation and optimization criterion. Through implementing such a total systems analysis criterion from the early conceptual design stages, the method seeks to facilitate the consistent sizing and configuring of the total system and its subsystems throughout the design process.
2. To develop in cooperation with subsystem experts a valid, and highly realistic, total systems model, based on this method, for an Earth observation imager system, to demonstrate the benefits of utilizing a total systems analysis approach for sizing and configuring space systems in the early conceptual design stages.
3. To use the developed model to investigate size and configuration issues regarded by subsystem experts and the literature as central to the conceptual design stages.
4. To develop the total systems model using a common computer tool that is simple to program, modify, and use. The intent is to show that total systems models can be

developed using a common tool. Through using a common tool with the mentioned characteristics, the method seeks to integrate the subsystem specialists into the total systems modeling process, to promote multi-disciplinary team efforts and a parallel rather than a sequential design process.

This research should be seen as a first step towards the implementation of a total systems analysis approach to the design of space systems. Focus in this research was therefore placed on the development of a method, and on showing that total systems models can be built, using a common programming tool, and used for sizing and configuring space systems in the early conceptual stages of the design process. Issues related to the implementation of the method and modeling approach are, therefore, not dealt with directly. However, investigations focusing on these issues are included in the list of suggestions for future research.

1.2. Research Problem and Hypotheses

This research seeks to deal with the problems associated with the development of large, total systems analysis models. The research problem arises as a consequence of building models that account for the many and complicated interactions between subsystems and system variables, a problem that is particularly troublesome when subsystem models (equations) have been developed independently of each other. First, there is the issue of complexity. Total systems models will span a number of subsystems, and may include hundreds of variables. Understanding, and being able to mathematically model, the relationships between the subsystems and between the many variables, therefore, represent challenges in themselves. Second, there is the issue of consistency. Each subsystem area is likely to derive its knowledge base from its own sources of literature and its own experts. Great care must, consequently, be taken to assure that variables, equations, and assumptions are defined in the same way in all subsystems.

Third, there is the issue of computability. The impact of a weakly defined subsystem variable or equation can cascade and even be amplified in large total systems models. Using equations based on solid mathematical, physics, and engineering theory to describe as many model relationships as possible might alleviate this problem.

The hypotheses for this research are defined as follows: (1a) It is feasible to develop a total systems analysis method for the design of space systems, with focus on the conceptual stages of the design process. (1b) The method can be demonstrated by applying it to the design of Earth remote sensing systems. (1c) The demonstration of a total systems analysis approach generates insight into issues emphasized by space system experts and literature. (2) Common programming tools are available that can be used for developing and analyzing these total systems analysis models. (3) A total systems analysis forces the consolidation of information on subsystem and system level relationships that has value in future design projects.

Verification and validation of these large total systems models are difficult. Ideally, model validation would involve comparing results from the model with real-world data. For total systems models, real-world data covering the full system are quite often not available, as is the case with complex satellites. Without adequate system-wide data, model verification may also be problematic. In this research, an alternative verification and validation approach is therefore taken. First, an effort is made to use equations that were already validated in their own fields, or backed by expert opinions. Second, the programming language that is used makes a strong verification of the programmatic relationships between subsystems possible. Third, preliminary feedback on the accuracy and utility of the model is provided by various space system experts. The demonstrations in Chapter 6. DEMONSTRATING THE MODEL, served to provide extensive computational experience with this total systems model, including the analysis of numerous design issues.

1.3. Overview of Chapters

The development, validation, verification, and demonstration of the method are discussed in the seven chapters of this report. Brief summaries of these chapters are provided here.

Chapter 2, **Theoretical Foundation:** The theoretical literature dealing with systems analysis, evaluation, optimization, and design is vast. To narrow the number of relevant sources to a manageable level, emphasis has been placed on literature that has a practical orientation, that deals with complex systems, and that focuses on total and integrated systems modeling. The literature review deals primarily with the fields of **systems theory**, **concurrent engineering,** and **systems engineering**. Other related fields, such as systems analysis, multi-criteria decision making, and utility analysis are mentioned only briefly.

Chapter 3, **Defining the Research:** The rationale and focus for the suggested research is discussed and developed in this chapter. The discussion includes a description of previous and present efforts at applying methods, similar to the one suggested here, to the design of complex space projects; and, it highlights the main new features of the method suggested in this research. The procedure for evaluating the method is also discussed.

Chapter 4, **Developing, Verifying, and Validating the Model:** This chapter deals with the development, verification, and validation of the mathematical model developed for this research. The chapter includes an overview of the modeling tool, LabVIEW, and how it was applied to this specific modeling problem, and a discussion of the equations describing the interactions in the model. There are two groups of equations: those based on theory from physics, engineering, and mathematics, and those derived from empirical/expert data gathered specifically for this research. The development and validation of the equations are dealt with differently in the two groups.

Chapter 5, **Describing the Technical Systems Model:** In this chapter, the specific interactions modeled between system variables, and between subsystems, are discussed. The discussion includes a general description of the model equations, and a specific description of every variable and its value interval. A complete list of all model equations is provided in Appendix A: EQUATIONS.

Chapter 6, **Demonstrating the Model:** The purpose of this chapter is to demonstrate how the developed model, and thereby the developed method, can benefit the analyses required in the early conceptual design stages, for system and subsystem size and configuration. The analyses included in this demonstration are technology selection analyses, detection and analyses of technology limits and bottlenecks, and trade-offs between inter-subsystem variables.

Chapter 7, **Feedback from Potential Users:** Feedback from potential users of the developed method and modeling approach is discussed in this chapter.

Chapter 8, **Conclusions and Recommendations:** The development of the proposed method and the demonstration of how it can be applied in the design process of V/IR Earth remote sensing projects represent the main contributions of this research. These contributions together with suggestions for future research are discussed in this chapter.

2. THEORETICAL FOUNDATION

The importance of making the design process for space projects more efficient is just starting to take a foothold within the space industry. Academic literature dealing specifically with these issues is, therefore, quite limited. In this review, three relevant areas of the literature are discussed. They are systems theory, concurrent engineering, and systems engineering.

Aguilar says, in his book about **system theory**: " Physically a system is composed of a large number of interacting components, each of which may or may not serve a different function, but all of which contribute to a common purpose." He goes on to define system analysis, system design, and system synthesis. His system analysis includes the "process of separating or breaking up a whole system into its fundamental elements or component parts" and "detailed examination of the system...to determine its essential features." In his system design definition, he focuses on "the process of selecting the components...steps, and procedures for producing a system that will optimally satisfy the stated goals." He, therefore, sees system design as forming the basis for "anticipating and solving problems" during the planning, engineering, architectural, and construction stages. He defines system synthesis as the "process of putting together...elements to form a whole system...to ensure optimal system performance." With respect to optimal performance, he defines an optimization model, and says that it consists of a "conceptual model...sufficiently analogous to the real problem, but...simple enough to...be amenable to quantitative analysis." About implementing the systems approach, he stresses the importance of having decisions being made through teams.[1] Systems theory captures a number of the general issues that are dealt with in this research. However, for more information on specific issues, such as alternative optimization criteria, the time phases in a project, and alternative system design approaches, other areas of the academic literature are better suited.

Concurrent engineering or **CE** is one such field. The concept of concurrency has been defined by one author as a "systematic approach to the integrated, concurrent design of products and related processes, including manufacturing and support."[2] Another author says: "CE is intended to cause designers, from the very beginning of a design activity, to consider all elements of the product cycle, from product concept through design, manufacture, service, and even disposal."[3] Some authors, within the CE community, have termed this approach "life cycle design."[4] Life cycle design focuses on developing designs that are producible, assemblable, testable, serviceable, and transportable.

The term "very early in the design process" corresponds closely in time to the conceptual design stage as defined, for example, by the aerospace industry which includes the following phases in the design process: mission requirements, conceptual design, conceptual baseline, preliminary design, allocated baseline, detailed design, production baseline, and production and support. The conceptual design phase should include both an optimization analysis and a parametric analysis.[5]

In the CE process, the intention is to evaluate design performance and economics concurrently. Trade-off analyses can therefore be conducted during the design process to guide the designers towards an optimal design that effectively balances performance issues, such as quality and maintainability, against a project's economics. This approach is often named "design justification" in the CE literature. Another similar approach "techno-economics" has been introduced by Brownfield.[6]

Economics takes an important place in CE. One reason for this is that economics is seen as the "only real common language between all of the diverse elements of an organization." In such an environment, it would be only logical to base design decisions on "quantitative economic trade-offs." This concept is regarded as one of the CE "commandments."[7] Noble sees economics as the primary motivation for doing CE, and emphasizes the need for developing mathematical tools that integrate economic issues within the CE approach. Such tools would ensure that economics, including cost considerations, would have a direct impact on design decisions during the design process. "Traditionally economic evaluations were not conducted until the design had been completed."[8]

The CE literature discusses different performance criteria for optimizing designs. Some of these are design for maintainability, design for reliability, design for cost, and design for supportability. Design for maintainability is discussed extensively in MIL-STD-721C, and the importance of formalizing reliability and maintainability as integral parameters of the design process has been emphasized by the Air Force in their "R&M 2000," the Navy in their "Best Practices Approach," and the Army in their "Reliability Initiatives." Software tools for integrating repair and maintenance considerations into the design process have been developed by, for example, the GD Convair Division. Their program is called RAMCAD.[9]

The problem with these performance criteria is that they are not general, in the sense that optimizing for supportability does not mean that the system would also be optimized for reparability and maintainability, or for cost for that matter. A general optimization criterion, that takes all these different performance criteria into account simultaneously, should therefore be preferred.

CE offers a framework in which the principles of project and product design can be discussed. Issues, therefore, tend to be discussed at a principle level rather than at a detailed level. Detailed discussions are left to the many related fields. Such fields are costing, economics, management, and systems engineering. The strong link between CE and systems engineering is well recognized in the CE literature. Both tend to deal with similar issues, but with a different focus. Systems engineering focuses on issues related to the design and operations of complex projects. CE, on the other hand, was developed to deal primarily with issues related to products and manufacturing processes. CE was built on a systems engineering foundation, and authors of CE say about this relationship: "CE equals the old systems engineering process (SEP) plus a new computing and networking environment, plus borrowed Japanese quality engineering methods," and "the first of ten characteristics of CE state that a comprehensive systems engineering process using a top-down design approach is required, and that this process is almost a requirement for implementing the 9 other characteristics of CE."[10]

Many of the ideas that today make up **systems engineering** were pioneered by Bell Labs. During the 1950's and 1960's, systems engineering was successfully implemented in a number of spacecraft, and civilian and military aircraft projects. The US Department of Defense (DoD) took an early interest in systems engineering, and they have been regarded as leaders in this field for more than 20 years. Their handbooks and standards provide central source material for students of systems engineering.[11]

Systems engineering can be regarded as "both a technical and management process, and to successfully complete the development of a complex system, both aspects must be applied."[12] Systems engineering can be defined as "a process that starts with the detection of a problem and continues through problem definition, planning, design of a system, manufacturing or other implementing action, its use, and finally on to its obsolence."[13] Some authors talk about this as the systems engineering process,[14] but the basic idea is the same.

Systems engineering deals with complex systems, and various aspects of the modeling of these complex systems are discussed, extensively, in the literature. "A model should represent the dynamics of the system configuration being evaluated, and incorporate provisions for ease of modification, and/or expansion to permit evaluation of additional factors as required."[15] There are different types of models. A descriptive model should reveal the structure of a complex system and demonstrate how elements interact with other elements; the primary purpose being to learn more about the system. Most descriptive models are quantitative models. Quantitative models are "mathematical models whose behavior is completely determined by assumptions used in constructing the model."[16]

There are a number of advantages to using a mathematical modeling approach: (1) Interrelated elements can be integrated as a system rather than being treated on an individual basis; (2) all major variables of a problem can be dealt with and considered on a simultaneous basis; (3) a comparison of many possible solutions is possible and can aid in selecting the best among them rapidly and efficiently; and (4) relations between various aspects of a problem which might not be apparent in a verbal description can be exposed.[17]

The level of complexity in some of these descriptive mathematical models can become quite overwhelming, and sometimes "just finding a systematic way of handling all the variables and their interactions can become the most important problem."[18]

Other decision making, evaluation and optimization approaches, such as robust decision making,[19,20,21] multi-criteria/objective decision making,[22,23] satisificing,[24,25] and the utility function approaches[26] are also discussed in the literature. However, the multi-criteria decision making approach and the utility function approach bring into the process a type of unwanted subjectivity, and all of the above approaches are limited in their ability to deal with the complexity and number of variables inherent in most engineering design processes. These approaches, therefore, tend to be better suited for a management environment than an engineering design environment. As such, these approaches will not be pursued any further here.

From this discussion, it was decided that systems engineering and concurrent engineering together should make up the theoretical foundation for this research.

---

[1]Aguilar, R.J., *Systems Analysis and Design in Engineering, Architecture, Construction, and Planning*, Englewood Cliffs. NJ. Prentice-Hall, 1973, 3-7.
[2]Schrage, D.P., Concurrent Design: A Case Study, in Kusiak, A. (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993, 535.
[3]Turino, J. L., *Managing Concurrent Engineering: Buying Time to Market*, New York, Van Nostrand Reinhold, 1992, 8.

[4]Gerschenson, J., Ishii, K., Life Cycle Serviceability Design, in Kusiak, A., (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993, 363.
[5]Schrage, D.P., 536.
[6]Noble, J.S., Tanachoco, J.M., Design for Economics, in Kusiak, A. (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993, 405-415.
[7]Turino, J. L., 130.
[8]Noble, J.S., Economic Design in Concurrent Engineering, in Parasei, H.R., Sullivan, W.G. (Editors), *Concurrent Engineering: Contemporary Issues and Modern Design Tools*, 1st. ed., London, New York, Chapman and Hall, 1993, 352-354.
[9]Klement, M.A., Design for Maintainability, in Kusiak, A. (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993, 385-389.
[10]Schrage, D.P., 540-549.
[11]Aslaksen, E., Belcher, R., *Systems Engineering*, Englewood Cliffs, NJ, Prentice-Hall , 1992, x.
[12]Schrage, D.P., 538.
[13]Hall, A.D., Systems Engineering from an Engineering Viewpoint, in Sage, A.P. (Editor), *Systems Engineering: Methodology and Applications*, New York, IEEE Press, 1977, 13.
[14]Schrage, D.P., 540-549.
[15]Blanchard, B.S., Fabrycky, W.J., *Systems Engineering and Analysis*, Englewood Cliffs, NJ, Prentice-Hall, 1981, 270.
[16]Sage, A.P. (Editor), *Systems Engineering: Methodology and Applications*, New York, IEEE Press, 1977, 7.
[17]Blanchard, B.S., 270.
[18]Aslaksen, E., 30.
[19]Friend, J.K., Jessop, W.N., *Local Government and Strategic Choice*, 2nd ed., London, Pergamon Press, 1977, 163-237.
[20]Rosenhead, J., Elton, M.E., Gupta, S.K., Robustness and Optimality as Criteria for Strategic Decisions. *Operational Research Quarterly*, 1972, 23, 4, 413-431.
[21]Rosenhead, J., Gupta, S.K., Robustness in Sequential Investment Decisions, *Management Science*, 1968, 15, 2, B18-B29.
[22]Dyer, J.S., Fishburn, P.C., Steuer, R.E., Wallenius, J., Multiple Criteria Decision Making, Multiattribute Utility Theory: The Next Ten Years, *Management Science*, 1992, 38, 5, 645-645.
[23]Schenkerman, S., Use and Abuse of Weights in Multiple Objective Decision Support Models, *Decision Sciences*, 1991, 22, 369-378.
[24]Ballou, D.P., Pazer, H.L., A Framework for the Analysis of Error in Conjunctive, Multicriteria, Satisificing Decision Processes, *Decision Sciences*, 1990, 21, 752-771.
[25]Kaufman, B.E., A New Theory for Satisificing, *The Journal of Behavioral Economics*, 1990, 19, 1, 35-51.
[26]Colton, J.S., An Intelligent Evaluation of Design for Manufacturing, in Kusiak, A. (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993, 457.

3. DEFINING THE RESEARCH

3.1. *Rationale for the Method*

This chapter discusses how issues such as the modeling approach (system vs. subsystem approach), modeling of interactions, programming approach, and design decision criteria are being dealt with in the space industry and in other relevant industries. Through this discussion, a rationale and focus for the proposed method is developed.

Most players in the space industry seem to deal with complexity by breaking projects down into subsystems at the expense of the relationships between them. This has led to a focus on evaluation and optimization at the subsystem level rather than at the system level. A number of advanced mathematical models have been developed describing different subsystems such as mining and processing systems,[1,2] life support systems, propulsion systems, and, perhaps more relevant for this research, power systems, imaging systems, communication systems, and guidance, navigation and control (GN&C) systems. Unfortunately, a number of these models are complicated and time consuming to run and tend to be used at the end of the design cycle, rather than being integrated into the dynamic design process from the beginning.

There are, however, people in the space sector who emphasize the importance of **total systems models**. For example, Dr. Eileen Stansbery emphasizes the need for using a total system model for comparing operational approaches for accomplishing mission objectives, and ultimately for optimizing top-level performance parameters for major system concepts. She further focuses on a mathematical model's ability to capture the dynamic interactions between system performance variables.[3] Similar ideas are voiced in the "First Lunar Outpost System Effectiveness Report." With a concentration on

interactions between elements, it says, "Defining the interactions between major elements of the system through measurable parameters allows us to understand how changes in the performance of one major system affect the performance of another major system or the overall accomplishment of mission objectives."[4]

Some efforts at developing mathematical systems models have been made. One such is the Lunar Base Model developed by the University of Texas, Austin, by Bell and Bilby. It was developed for the Johnson Space Center (JSC) to evaluate, study, and simulate different Lunar base concepts, with emphasis on (Lunar) surface equipment. The mathematical model is simple, and the equations describing relationships are for the most part linear and of the one variable type. This limits its usefulness. Still, the model demonstrates how descriptive mathematical models can be used to increase the understanding of a system. In the previous chapter, Chapter 2, the term "to learn more about the system" was used.[5]

Even more interesting are the Figure of Merit (FoM) approach developed at the University of Arizona[6] and the space station optimization model tool developed by Chamberlain, Fox, and Duquette at Jet Propulsion Laboratory (JPL).[7] In the FoM approach, the authors focus on the overall mission architecture, but still capture the importance of details through the use of accurate technical equations describing relationships. System level optimization, rather than isolated component (subsystem) optimization, together with model flexibility are being emphasized. Flexibility ensures that design changes, occurring during the design process, can be integrated into the model. FoM has been demonstrated on a Mars sample return mission. The space station optimization model tool also captures the technical relationships between subsystems and variables, but its complexity has made its usefulness limited.

TECHSAT uses a different approach. It was developed to be used as an aid in the design process for Earth observation satellites. TECHSAT is built up as a database, and focuses on the technical details of the various subsystems rather than on the interrelations between them. Still, simple mathematical relationships are included in the database, giving TECHSAT some systems analysis capabilities. Time consuming analyses have limited its usefulness.[8]

Modeling tools have also been developed for other sectors such as the chemical process industry[9] and the offshore oil and gas industry. The process industry seems to have placed focus primarily on project evaluation approaches with less emphasis on the technical modeling aspect. The offshore industry with its emphasis on large and complex technology developments for extreme environments has, on the other hand, developed sophisticated tools integrating both technical and economic considerations into the design process from the early conceptual design stages.[10]

Dan Goldin, head of NASA, through his exhortation for cheaper, faster and better, has emphasized the need for making space missions do more through smaller and lighter platforms, at a **lower total cost**, and through a **shorter development period**. To meet this challenge, NASA centers such as Goddard, Johnson, and Langley, are trying to integrate existing systems and tools with the aim of making a total systems analysis possible throughout the design process.

The Jet Propulsion Laboratory has gone a step further and opened a center dedicated to looking into ways of improving the design process. The center called the Project Design Center (PDC) was opened last year. Projects such as the LIGHTSAR, Pluto Express, Mars'98, solar Probe, and New Millennium are currently using the PDC facilities and tools. Some of the computer tools available at the center are Multidisciplinary Integrated Design Assistant for Spacecraft (MIDAS), Project Trades Model (PTM), QUICK (a conceptual design tool). These tools are integrated through an interface built up around Excel spreadsheets and Visual Basic.

The ambition of JPL is to use the PDC to reengineer their design process to facilitate the design of the next generation of JPL space missions. These missions will cost less, hundreds of millions as opposed to the billions that were spent on missions such as the Viking and Voyager missions of the seventies and eighties, have a lower mass, and be developed and launched in months or years, rather than decades. The **new design process being reengineered** at JPL will focus on concurrent engineering (a parallel, rather than a sequential design process); on multidisciplinary team efforts, through a flat rather than a hierarchical organization; and on integrating existing and newly developed computer tools making a total analysis of any project possible.[11,12,13]

At the PDC, JPL has made it possible to design concurrently "the major elements in the design process - science mission, spacecraft, and operations." This concurrent engineering of mission, spacecraft, and operations through mathematically integrating these tools represents a total systems analysis approach, similar to the one suggested in this research. The major difference is that JPL uses a number of different programs describing different parts of the systems, while this research suggests using a common programming language for the programming of all subsystems. The advantages coming from the JPL approach should therefore also be applicable to the method developed for this research.

Some of the advantages of the JPL approach are that "designers can try out ideas, construct models, and observe the effects of various solutions as the results of the proposed action propagate through all the functional areas. These multiple 'iterations' can be carried out rapidly and evaluated in real time." By using this total systems analysis approach, "the effects of changing requirements or capabilities among the major elements can be quickly assessed..." and "...be readily understood." JPL documents emphasize that such a concurrent engineering process, including the use of multifunctional teams, should be implemented into the design process "from the very beginning" of a project.

The realization of these advantages is partly attributed by JPL to "today's information-system technology," and JPL experts say that through the use of this technology it has become "...possible to radically change the traditional project design process, making it faster, more efficient, and more cost effective."

An advantage of using the approach suggested in this research is that all subsystems are developed using a common programming environment. At JPL, the analysis tools are connected together through a common interface (Visual Basic and Excel), but the tools themselves are developed in different languages, primarily in FORTRAN-type languages. Programming and modifications are therefore likely to be more time consuming than what would be the case for similar modifications carried out in LabVIEW.

To JPL, the opening of the PDC represents "a revolutionary development in the Laboratory's project design capabilities," and they see the PDC as providing an environment "where the new reengineering design processes can be validated and implemented for use on future missions." [14]

Most programs used for modeling and analyzing space systems have been developed in line-by-line code such as FORTRAN and C, both powerful programming languages. However, line-by-line coding is time consuming, and the developed programs are relatively difficult to modify. As development time gets more critical, other alternatives should be sought. LabVIEW, a graphical programming language, represents one such alternative. The programming language is extensively used for developing so-called Virtual Instruments (vi's) and data-analysis tools for data acquisition systems. LabVIEW is now also starting to be used in the design of real instruments and testing systems such as submarine sonar test systems,[15] and radar simulators.[16] A developer of sonar test systems claimed that "using LabVIEW rather than traditional line-by-line coding, reduced development time from 16 months to four months." This represents a significant 75% time savings.

LabVIEW, being a graphical programming language, is easy to use, program, and modify. These characteristics give additional benefits. Wise, R.M., Department of Surgery, Thoracic and Cardiovascular Division, School of Medicine, University of Maryland, says about "easy to program and modify": "...finished programs could not be modified by residents; any alterations, which often proved painstaking and time consuming, were performed by software-engineers. Today, using LabVIEW, we can bypass the software engineers and get closer to our data. Residents not only perform complex data analysis on their own. They can also rapidly alter [modify] the experiments as needed."[17] The idea of bypassing the software engineer (or programmer) and letting the user also be the programmer might be termed: making the programmer and engineer or scientist "one."

In a design environment, making the programmer and user (engineer or scientist) "one" has a number of advantages. It makes the engineer and scientist (subsystem specialists) an integral part of the design process, and makes them "feel a higher sense of ownership in the [developed] system and a sense of control over the way it operates."[18] The process of defining and developing these systems should facilitate communication between subsystem specialists and enhance a multidisciplinary team approach.

Various design decision criteria have been discussed and used for evaluating and optimizing space projects. Launch cost computed as a function of mass launched led designers to focus on minimizing system mass. Mass, it was argued, could serve as a good test for project feasibility in a situation with limited project data available. The mass payback ratio (MPR) is based on these same ideas. The MPR was developed for evaluating and optimizing early space resource utilization schemes. Optimizing for MPR meant that designs were sought that would maximize the ratio between extracted resources and launched mass. MPR has been applied to the FoM developed at the University of Arizona.[19] Other criteria such as minimum Δv (change in velocity) have been used in designing interplanetary missions. Propellant is mass, and any velocity change, Δv, will use propellant. Designs minimizing Δv, therefore, would also minimize total launched mass.[20] Cutler would call these physical economic criteria, as opposed to monetary economic criteria.[21]

Monetary economic criteria have been used in some space projects. In the design of the space shuttle and the space station, minimum life cycle cost was used. Simonds used the present value (PV) approach in his study of potential investments in new technology developments.[22,23] For the TECHSAT model the PV approach was applied in calculating life cycle cost. ROI was used by Woodcock for evaluating and comparing different interplanetary mission scenarios.[24] Fox and Chamberlain of JPL used the PV idea in developing an evaluation and optimization model for the space station.[25]

To conclude, in the design process of space systems, emphasis is slowly starting to be put on the **development of complex technical total systems models** describing subsystem interactions. The efforts at the JPL PDC are especially interesting, but even they are in the experimental phase. Using **graphical programming languages** as tools for modeling and analyzing spacecraft designs has the potential of significant savings in development time.

Based on these findings, it was decided that the emphasis of this research and the developed method should be placed on **total systems analysis models** and on **mathematically modeling the interactions** between system variables and subsystems. These development efforts target the **early conceptual stages** of the design process, and LabVIEW is used for developing the total systems models. Mass as a proxy for cost is used as the evaluation and optimization criterion.

This emphasis is not new compared to the efforts being made at the JPL PDC. On the other hand, since these efforts are only in their experimental stages it could be argued that the research described in this report, rather than duplicating the efforts at the JPL PDC, works in parallel with them -- parallel, in the sense that using a graphical tool like LabVIEW in modeling and analyzing space design projects represents something new. The same goes for the use of mathematically derived total systems models in the design of remote sensing systems.

3.2. *Focus of the Method*

For design projects where the emphasis is on technology utilization, low cost, and a short development cycle, it is required that these concerns are integrated into the design cycle from the early conceptual design stages. It has been estimated that some 70% of the accumulated total life cycle costs are determined through the decisions made during the conceptual design stage.[26] In this research, focus has been placed on developing an approach, a process and a model especially tuned to the major issues of these early conceptual design stages.

The method emphasizes total systems analysis models and the mathematical modeling of interactions between system variables and subsystems. These concepts are discussed in the systems engineering and concurrent engineering literature and emphasized in the "First Lunar Outpost System Effectiveness Report," and by JPL and Dr. Eileen Stansbery. The aim of this research is to translate these concepts into a practical and applied methodology that can be used in the early conceptual design stages. The model describing the system and subsystem interactions will be built up as a descriptive mathematical model. It will be quantitative rather than qualitative. Using a total systems analysis approach, as opposed to a subsystem analysis approach, makes it possible to deal with major decision variables on a simultaneous basis. These characteristics make the mathematical model well suited for quick and effective evaluation and ultimately optimization of different alternatives, especially in the early conceptual design stages.

3.2.1. Size and Configuration

Two major issues of the early conceptual design stages are size and configuration. All subsystems and their components must be sized so that they, and the resulting total system can meet set mission requirements. In the same manner, all subsystems must be configured with the right technologies and components to meet these mission requirements.

Demands on the payload generated from the mission requirements represent the main drivers for these size and configuration decisions.

3.2.2. Subsystem Development within the Total Systems Model

Once initial subsystem sizes and configurations have been determined, each of the subsystem specialists can start developing their own subsystems to a greater level of detail. As these subsystems are sized and configured to be part of the total systems model, these individual subsystem efforts are kept within the frame of the total system. Using this design approach should avoid later, major and time consuming re-designs due to incompatibility between subsystems.

3.2.3. Analysis Capabilities

Optimal size and configuration demands that a large number of mission scenarios, technology alternatives and variable values can be analyzed easily and within a short time frame. The developed model includes sophisticated trade-off analyses between almost any variables, both intra- as well as inter-subsystem. Extensive post analyses were made possible through the generation of spreadsheet files containing trade-off analysis data.

3.2.4. Variable and Subsystem Interactions

Size and configuration decisions require a comprehensive understanding of the mathematical interactions between system variables, and between subsystems. The developed total systems model focuses on mathematically describing these interactions. Mission scenarios or mission requirements are defined by the user and integrated into the model. So are operational phase considerations.

3.2.5. Evaluation and Optimization Criterion

In both the systems engineering and the CE literature, it is being emphasized that economic parameters should be used for guiding designers towards an optimal design that effectively balances performance against a project's economics. In this research, mass is used as a proxy for the economic parameter cost, such that the design alternative giving the lower total mass would be preferred. However, the developed model can easily be modified to accommodate cost and value relationships. A project's value would be a function of its output.

3.2.6. Graphical Programming Language

To provide a strong and common programming environment with sophisticated, and effective analysis capabilities, the graphical programming language of LabVIEW was selected for this research. The LabVIEW graphical programming language is easy to learn and easy to program and modify, and it creates a clear and easy to use user-interface.

3.3. Evaluating the Method

Method evaluation was performed through the development and running of the model. For demonstrating and evaluating the method, a model of a space based Earth observation V/IR imager system was developed. LabVIEW was used for developing and analyzing the model. The level of detail incorporated into this model made it possible to perform realistic design analysis. A full discussion of model subsystems, variables, and interactions is given in Chapter 5.

The developed model was used for performing trade-off analysis, focusing on two of the major issues of the conceptual design stages -- size and configuration. Great care was taken to use relevant mission scenarios. The aim of these analyses was to demonstrate how the method, model, and selected modeling tools can be used for facilitating design decisions in the early conceptual design stages.

These evaluation efforts were complemented by feedback from potential users of the method, regarding this specific method, the model development, and the results gained from running the model. Potential users include subsystem specialists and systems engineers involved in the design of spacecraft systems. Concepts from systems engineering and concurrent engineering apply to this group. Further details are provided in Chapter 7.

---

[1]Waldron, R.D., Processing Options for Lunar Propellant and By-Products, Annual Invitational Symposium, Space Mining and Manufacturing, UA/NASA Space Engineering Research Center for Utilization of Local Planetary Resources, Tucson, Arizona, 1989.
[2]Keller, R., Production of Oxygen from Lunar Soil by Molten Salt Electrolysis, Annual Invitational Symposium, Space Mining and Manufacturing, UA/NASA Space Engineering Research Center for Utilization of Local Planetary Resources, Tucson, Arizona, 1989.
[3]Stansbery, E., *Architecture Effectiveness Model Summary*, Lunar and Mars Exploration Program Office, NASA - JSC, April 18, 1991.
[4]First Lunar Outpost System Effectiveness Report, EXPO Internal Document No. XE-92-005, Exploration Programs Office, NASA-JSC, May 1992.
[5]Bell, L., Bilby, C., The Lunar Base Model and its Role in Systems Optimization, Proceedings of the Annual Invitational Symposium, Space Mining and Manufacturing, UA/NASA Space Engineering Research Center for Utilization of Local Planetary Resources, Tucson, Arizona, 1989, v19-v20.
[6]Ramohalli, K., Kirsch, T., A "Figure of Merit" Approach to Extraterrestrial Resource Utilization, Proceedings of the Annual Invitational Symposium, Space Mining and Manufacturing, UA/NASA Space Engineering Research Center for Utilization of Local Planetary Resources, Tucson, Arizona, 1989, v1-v11.
[7]Chamberlain, R.G., Fox, G., Duquette, W.H., *A Design Optimization Process for Space Station Freedom*, JPL Publication 90-23, NASA, June 15, 1990.
[8]Princeton Synergetics, Inc., *Technical Description of TECHSAT, A Computer Aided Spacecraft Design System for Earth Observing Satellites*, NASA Contract No. NAS23-26313, Cleveland, August 5, 1992, 19-27.
[9]Valle-Riestra, J.F., *Project Evaluation in the Chemical Process Industries*, New York, McGraw-Hill, 1983, 1.
[10]Oxnevad, K.I., Space and Offshore, *Norwegian Oil Review (NOR)*, August 1992, No.7, 101-114, 169-170, 176-178.
[11] Marino, K., *The New Millennium Program*, JPL Universe, February 10, 1995.
[12] Project Design Center, Jet Propulsion Center, Publication JPL 400-533, July 1994.
[13] Casani, K.E., Thomas, N.W. The Flight System Testbed, SPIE-94, Conference.
[14] Project Design Center.
[15] Simmons, K.P., SEA CORP., LabVIEW Slashes Development Time of Submarine Sonar Test System, Instrumentation Newsletter, *Technical News from National Instruments*, Autumn 1995, Vol. 7, Special Academic Edition, 12.
[16] Durand, R.L., Technology Services Corporation, LabVIEW for Windows Cuts Development Time for Radar Simulator, Instrumentation Newsletter, *Technical News from National Instruments*, Summer 1995, Vol. 7, Number 2, 13.
[17] Wise, R.M., Department of Surgery, Thoracic and Cardiovascular Division, School of Medicine, University of Maryland, LabVIEW our Choice for Cardiothoracic Research, Instrumentation Newsletter. *Technical News from National Instruments*, Autumn 1995, Vol. 7, Special Academic Edition, A-4.
[18] Clarkson, M., Tools on the Fly: Virtual Instruments Take Off, *Desktop Engineering*, September/October 1995, 1, 1, 20-28.
[19]Ramohalli, K..
[20]Sackheim, R.L., Wolf, R.S., Space Propulsion Systems, in Wertz, J.R., Wiley, J.L. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 579-582.

[21]Cutler, A.H., *Space Manufacturing 5, Engineering with Lunar and Asteroidal Materials,* Proceedings of the Seventh Princeton/AIAA/SSI Conference, Princeton, New Jersey, May 8-11, 1985.
[22]Simonds, C.H., *Decision Analysis for Space Systems: Overview and Summary of Pilot Study,* Presentation as part of subcontract to Bionetics Corp., Subcontract: 906-317, Prime Contract NAS 2-13, 1993, 273.
[23]Simonds, C.H., Ballin, M., Schlater, N., NASA TECHNOLOGY INVESTMENT: An Application of Decision Analysis, Presented to the Exploration Programs Office, NASA-JSC, November 17, 1992.
[24]Woodcock, G., *Space Transfer Concepts and Analysis for Exploration Missions,* Contract NAS8-37857, Executive Summary, Phase 1, Boeing Defense and Space Group, Huntsville, February 1991.
[25]Chamberlain, R.G., 1-10.
[26] Fertig, K.W., Gonda, M., *Computer Environments for Design and Analysis Design Sheet: An Engineer's Spreadsheet,* Rockwell International Science Center, Palo Alto Laboratory, December 8, 1993.

4. DEVELOPING, VALIDATING, AND VERIFYING THE MODEL

4.1. *The Modeling Tool*

4.1.1. LabVIEW Basics

LabVIEW is a cross-platform (UNIX, PC, Mac, and PowerPC) program development tool similar to, for example, C. It uses a graphical programming language called G rather than a text-based programming language for creating code. A LabVIEW program will appear in a block diagram form. The boxes in the block diagram can represent user defined programs, subroutines, or subsystems, or built in functions and subroutines. These boxes are recognized through their icons. For user-defined subroutines, the icons are drawn by the user. Functions or subroutines are, in the LabVIEW vocabulary, called virtual instruments (VIs), because they imitate the appearance and operations of an actual instrument.

A VI includes a front panel and a (block) diagram. The front panel is the VI's interactive user interface. In the **front panel**, the model developer will define all input (**controls**) and output (**indicator**) variables. The relationships between these input and output variables are defined in the **block diagram**.

Front panel controls and indicators are represented, mainly, through knobs, push buttons, and graphs. These controls and indicators may represent numerical, Boolean or string (alphanumeric) data as single data points, or as tables, arrays, clusters, charts, or graphs. Buttons and knobs and other manipulative controls are operated through point and click. Alphanumerical values are entered through the keyboard. By clicking on a knob, button or other control, the user may change the value of a single variable or a number of variables, or be able to select between equations or subroutines used for calculating one or many variable values. A control can be used for manipulating variables, equations, etc., within its own VI, and within any other VI to which this VI may be connected. Some front panel layouts are provided in Appendix E: THE LABVIEW MODEL.

The relationships between the controls and the indicators are defined through connecting functions; structures, such as FOR loops, WHILE loops, and case and sequence structures; formula nodes; and user defined subroutines connected together through "wires" to their nodes. Data (numerical, string, and Boolean) are passed through the diagram through these "wires." A node represents either an output variable (indicator) or an input variable (control). For user defined subroutines, the user defines these nodes. These user defined subroutines may be built in C making it possible to integrate, for example, MatLab routines converted into C, in a LabVIEW program.

To run a LabVIEW program, the user points and clicks on the run button, depicted as an arrow, located on the front panel and on the block diagram. Any program or subprogram can be run independently. When a program is run, all its subprograms will also run. This modular feature makes it possible to build up and test each subsystem independently before integrating it into the full total systems model. Results from a "run" can be studied directly in the window showing the front panel, and/or they can be written to a text or spreadsheet file, making it possible to perform sophisticated post-analyses and to easily share results with other users.

Reference to sources and documentation, explanation or derivation of variable values, and equations are made simple in LabVIEW through user defined on-line help windows. Some examples are provided in Appendix E: THE LABVIEW MODEL. The help windows are available for any control, indicator, structure, formula node, and user defined subsystem, and are accessed simply by moving the mouse over any of them. (The help function needs to be activated first.) Printouts of front panels, diagrams, and connector panes, showing a subsystem's connection to other subsystems, can be used to document directly the model development throughout the design process.[1]

4.1.2. Utilizing LabVIEW

The LabVIEW Earth Observation Model developed for this research includes the Propulsion System, the Sensor System (V/IR Imager), the Data Storage And Processing System, the Communication System, the Guidance Navigation & Control System (GN&C), and the Power System. Selected (block) diagrams for these subsystems are shown in Appendix E: THE LABVIEW MODEL. Together, they form the **Satellite System Analysis** part of the model. The other part, **Orbital Analysis**, includes these subroutines: Two Body Motion in Circular Orbits, Angular Displacement (degree and km), Spacecraft Horizon and Swath Width, Communication Time, and Eclipse Time. For an in-depth discussion of these subsystems and subroutines, refer to Chapter 5, DESCRIBING THE TECHNICAL SYSTEMS MODEL.

In the developed model, buttons, enumerators, menus, etc., now collectively called **switches**, in combination with case structures, are used for selecting between imager sensor modes, data processor technologies, data storage device technologies, communication antenna types, momentum dumping thruster technologies, solar cell types and solar array technologies, and battery types. Some of these are illustrated in Appendix E: THE LABVIEW MODEL.

Loops are used in the model for calculating total propellant required for reboosting and for momentum dumping during the spacecraft lifetime. These calculations describe the interactions taking place between variables over time and might as such be considered as simulations.

The developed total systems model can be run in two modes. In the first mode, calculations are performed throughout the whole model, displayed in the open front panels. and written to a history file containing data from every analysis run. The history file includes data from 200 selected input and output variables and a unique timestamp for each run. The second mode, the data analysis mode, is identical to the first, except that selected data and their variable names are written to two separate spreadsheet files: a data file and header file. Having the program automatically generate a header file using the variable names defined in the model saves time for the analyst, and ensures full consistency between variable names in the model and in any post-analysis efforts. The data and header files are automatically named based on the variable names of the two first variables included in the files and an alphabetical analysis index(A...Z). This analysis index makes it possible to run multiple analyses with the same set of variables. The user defines which seven variables are to be included in the generated files and how many runs should be included in the data spreadsheet file. The seven variables are selected through pull-down menus containing the same 200 variables that are included in the history file. The features included in the analysis mode were designed to make it possible to run any number of trade-off analyses in minutes rather than hours, days, or even weeks.

4.2. Developing the Model

4.2.1. Defining the Total Systems Model

The modeling efforts for this research started with an initial total systems model, including the subsystem input and output nodes, as defined through a number of discussions between the researcher and representatives from NASA LaRC. These initial discussions provided only a starting frame. During the modeling process, the model was continuously modified as more subsystem and system knowledge became available. The initial model included subroutines describing the interactions between the subsystems, and between these subsystems and their environment.

4.2.2. Subsystem Modeling

In this research, the author modeled all subsystems as well as the required system level routines. The subsystem models were developed in close cooperation with subsystem specialists, primarily at NASA LaRC. For a real life design project, subsystem modeling would be performed by the respective subsystem specialists.

The researcher started with limited knowledge of the spacecraft subsystems and their environmental interactions. Initial insight was gained through researching various theoretical and technical texts and relevant Earth observation projects. The environmental subroutines, such as the calculations of the orbital parameters, atmospheric friction, and eclipse time, were developed in full from these texts. For the subsystem models, these research efforts created the initial set up of the equations describing the various subsystems and their interactions with the other subsystems. These equations formed the basis for subsequent discussions with selected subsystem specialists. By including the subsystem specialists, it was possible to integrate some of their knowledge and practical experience into the mathematical equations and the developed model, bringing the modeling process as close to a real life situation as possible. The use of both established theory and expert opinions has been suggested in the literature as central elements in the development of valid descriptive and quantitative models.[2] The level of detail in the model was developed after studying the relevant literature and holding discussions with experts, focusing on the level of detail required for investigating the size and configuration issues important in the early conceptual design stages.[3] Some specific issues included: selecting the right technology (type of solar cell), configuring subsystems (on-board processing vs. no processing of sensor data), checking for technology limits and bottlenecks, and sizing the individual subsystems. How the model was used for investigating these issues is discussed and illustrated in Chapter 6, DEMONSTRATING THE MODEL.

The final version of the 130 developed equations describing the mathematical interactions between the 300 hundred plus variables included in the model were compiled into a list. The list, provided as Appendix A: EQUATIONS, includes around 170 footnotes, giving reference to every text and data source used. Such an extensive list of system level equations for remote sensing systems has not been derived and compiled before. This list, therefore, represents a contribution in itself.

Most of the equations describe **physical processes** that are either time dependent or time independent. Equations describing a time dependent physical process might be said to describe a simulation process. In the model, the equations used for calculating the propellant mass required for reboosting and for momentum dumping would fall into this category. Equations for calculating sensor signal to noise ratio, power produced by a solar array, atmospheric friction, etc., describe time independent processes. Any design model should include both types of equation.

Equations for calculating the mass of the sensor optics, communication transmitter/receiver unit, and antenna system, and the mass and power of the data processor and data storage system, were derived from empirical data. The unavailability of equations describing these mass and power interactions can be explained by the fact that current design processes tend to focus on point designs dictated by physical components. In this point design environment, only mass and power of the specific component is of interest. Often these components are sized and configured based on what is available rather than on what might be optimal for the total system of which they are part. Finding the optimal size and configuration of any subsystem requires an understanding of the relationships between its capacity and its power and mass, and of the interactions between the subsystem and other subsystems.

The list of equations was used for creating the subsystem models, the system routines, and the environmental models, which together form the total systems model. Text and data resources are extensively documented and made available for users of the model through the on-line help function.[4]

During the modeling process, the total systems model and the various subsystem models went through continuous modifications as more data became available, and as more insight was gained into the various processes of the model. The researcher experienced that insight was gained simply by setting up and structuring the model.

Modifications of the model included internal modifications only impacting the subsystems being modified, as well as modifications also impacting other subsystems, through, for example, changes in the input and output nodes of the modified subsystems. The internal subsystem modifications of equations and structures were not complicated to execute. In a real life situation, these modifications would be carried out by the involved subsystem specialist. The structure of the model would remain unchanged. The researcher also found modifications requiring the deletion or addition of subsystem nodes to be uncomplicated. These modifications would include the rewiring of some or all of the connections between the involved subsystems. For subsystems with a high number of nodes, this procedure requires accuracy and can be time consuming, but is robust and relatively uncomplicated. An on-line help function shows which variables are represented by which nodes. These types of modification would in a real life situation include the subsystem specialists for all of the involved subsystems. During the whole modeling process the researcher worked closely with a number of such experts.

4.3. Validation and Verification

These two concepts are defined in the literature.[5] Validation is concerned with whether the conceptual model, "as opposed to the computer program," is an "accurate representation of the system" being modeled. A valid model should produce results similar to the real system it is representing. Model verification seeks to establish whether the developed computer model "performs as intended." The verification process, therefore, deals with checking the translation of the conceptual model "into a correctly working program."

It would have been preferable to verify the developed model by using empirical data[6] from an existing Earth observation V/IR Imager system, and compare the results generated by the model to those generated by the design team of this existing system. In this manner, issues such as required data processing power, data storage capacity, communication data rates, required power, and the suggested sizing of the various subsystems could be compared. Few projects are available that can provide data with the required level of detail.

An alternative **verification and validation approach** was therefore taken. In this approach, it is assumed that the full model, as discussed in this section, Section 4.3, can be partially validated by validating its equations and subsystem models and verifying its total system routines.[7] For this assumption to hold, the connections between subsystems need to be solid and easy to verify. The simple flow diagram approach, connecting subsystems together through visual wires applied in the LabVIEW programming environment, makes this possible. Further details on LabVIEW are provided in Section 4.1.1, LabVIEW Basics. Further model validation was provided through feedback from space system experts. This feedback is discussed in Chapter 7, FEEDBACK FROM POTENTIAL USERS.

4.3.1. Validation Procedures

Physics, Engineering, and Mathematical Equations: 118 out of the 130 equations used in the mathematical model are securely anchored in equations developed and validated within each relevant field of physics, engineering, and mathematics. These equations were, therefore, considered solid. The validation process was, consequently, limited to verifying that these equations are programmed correctly in LabVIEW. Equations included in this group are those for calculating spacecraft velocity, orbital time, altitude, angular displacement, spacecraft horizon and swath width, communication time, eclipse time, propellant required for reboosting, sensor signal-to-noise ratio, sensor aperture diameter, sensor scanner power, sensor horizon and swath width, sensor data rate, processing power (IPS), storage capacity, communication dumprate, communication transmission power, torque from atmospheric drag, reaction wheel power, propellant for momentum dumping, area of solar array, and required battery energy. A number of these equations were developed together with subsystem specialists.

Approximately 50 hours were spent with subsystem specialists, discussing, developing, and deriving subsystem equations and models, both for these equations and for those discussed in the next paragraphs.

Empirically and Expert Derived Equations: The remaining 12 equations were derived from empirical data or based on suggestions from subsystem specialists, primarily at NASA LaRC. For these equations, the validation procedure includes both equation validation and LabVIEW programming verification. Equations included in this group are those for calculating sensor optics mass, power and mass of the data processing system, power and mass of the data storage system, mass of the communication transmitter/receiver unit, mass of the antenna system, mass of the solar array, and mass of the battery.

The equation for sensor optics mass as a function of sensor aperture diameter was derived through the line-fit method, based on data from a provider of spacecraft sensor optics.[8] The resulting equation gave a R^2=0.9734. R^2 is the correlation coefficient squared, and R^2=1 indicates a perfect fit between data points and the derived equation.[9]

Power and mass of the data processing system are defined as direct proportional functions of processing capacity, measured in IPS. These functions were derived through a combination of regression analysis of processing system mass and power and expert opinions. The mass and power constants are dependent on processing technology.[10]

Two sets of equations were used for calculating the *power and mass of the data storage system*, one set for the solid state data recorders and one set for tape based data recorders. In close cooperation with a subsystem specialist, the equations for the calculations of mass and power for the solid state data recorders were set up as incremental functions of data storage capacity. Equations for calculating mass and power for tape based data recorders were derived through the line-fit method. The mass equation gave an R^2=0.9962, and the power equation achieved an R^2=0.9818.[11]

The *absolute mass of the antenna system*, excluding the downlink parabola, varies with communication band and was derived from a suggested generic communication system. The constant defining the relationship between transmitter/receiver mass and transmitter power was derived from the same generic system. The constant defining the relationship between mass of the downlink parabola antenna and its diameter was derived from available data on antenna masses and diameters. The derived constant is dependent on communication band.[12]

The constant defining the relationship between solar array area and solar array mass was derived from data provided, by a manufacturer, for a specific solar array design.[13] The constant is dependent on solar cell technology.

Battery mass is defined as proportional to required battery energy. The constant defining this relationship was given and is dependent on battery technology.[14]

4.3.2. Testing the Model

The model was tested through running individual subsystems, and through running the whole model. In total, the whole model was run some 900 times during the development, validation, and verification periods.

Physics, Engineering, and Mathematical Equations: Given that all of these equations are validated in their own fields, only the major ones and those derived in cooperation with subsystem experts were tested.

Spacecraft velocity, orbital time, and altitude results generated from the Two Body Motion.vi (vi=virtual instrument) were compared to given examples.[15,16] The same approach was taken for verifying angular displacement results generated by the Angular

Displacement.vi;[17] spacecraft horizon and swath width results generated by the SC Horizon & Swath.vi:[18] communication time results generated by the Communication Time.vi;[19] and eclipse time results generated by the Eclipse Time.vi.[20] The results from the sensor signal-to-noise ratio and sensor aperture diameter equations generated by the Apert.Diam:pnl;dnl.vi were verified against results from other analyses of sensor systems. These results were provided by the subsystem specialist who had been involved in deriving and developing these equations. The verification effort led to modifications of the equations. The results given by the torque (from atmospheric drag) equations generated by the Atmosph.DragEff.vi were verified against available examples.[21] These tests were carried out primarily to verify that the Atmosph.Drag;Atm.Dens.vi, with its high number of data points, was giving the right results. VI titles are spelled out in full in Appendix G: VI TITLES.

Empirically and Expert Derived Equations: For the equations derived directly from empirical data, through a line-fit method, results were verified by running the model, and comparing results to those given in the empirical data sets. This approach was taken for verifying the equations for the calculation of sensor optics mass, the power and mass of the data storage system, and the mass of the solar array. A similar approach was taken for verifying the power and mass equations in the Data Processing.vi, and the parabola antenna mass equations in the Comm.Mass.vi. The equations for calculating communication transmitter/receiver unit mass were verified to see that the Comm.Mass.vi generated mass results were equal to those of the given generic communications system for any defined communication wavelength. Diagrams for some of these VI's are provided in Appendix E: THE LABVIEW MODEL. VI titles are spelled out in full in Appendix G: VI TITLES.

The empirical data used in this research should not be considered perfect, but rather considered as the best that was available at the time of this research. The resulting mass and power estimates should be considered accordingly.

Developing the model and defining its detail level and purpose through both established theory and expert opinions ensured that validity was built into the model during the model development stages. The various subsystem models were verified against results from analysis of similar systems. Total systems verification proved difficult, but the model can still be regarded as verified, due to the simple total systems verification procedure possible in LabVIEW. Lack of empirical data made validation of model output difficult.

---

[1] LabVIEW for Macintosh, *User Manual*, National Instruments, December 1993 Edition, Part Number 320591-01.
[2] Law, A.M., Kelton, W.D., *Simulation Modeling and Analysis*, 2nd ed., New York, McGraw-Hill, Inc., 1991, 300 & 308.
[3] Ibid., 300-301.
[4] 4.1.1. LabVIEW Basics, 25.
[5] Law, A.M., 299.
[6] Ibid., 311.
[7] Ibid., 306.
[8] Telecon with F. Kingsley, Space Optics Research Labs, July 24, 1995.
[9] Appendix C: EMPIRICALLY DERIVED EQUATIONS, 151.
[10] 5.2.3.1. Data Processing, 48.
[11] Appendix C: EMPIRICALLY DERIVED EQUATIONS, 151.
[12] Ford, J., Communications, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 11-22, 341.
[13] Telecon with representative for AEC-Able Engineering, Data for their PUMA rigid array, August 7, 1995.
[14] McDermott, J.K., Power, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 11-35, 362; 364.
[15] Brown, C.D., *Spacecraft Mission Design*, AIAA Education Series, American Institute of Aeronautics and Astronautics, Washington, D.C., 1992, 5-8.
[16] Stark, J. P. W., Celestial Mechanics, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 59-81.
[17] Stark, J. P. W., Mission Analysis, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 98-100.
[18] Brown, C.D., 73-74.
[19] Ibid., 77.
[20] Chetty, P. R. K., *Satellite Technology and its Applications*, 2nd ed., Blue Ridge Summit, PA, Tab Books, 1991, 157-158.
[21] Zermuehlen, R.O., Zimbelman, H.F., Guidance, Navigation, and Control, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 11-7b, 316.

5. DESCRIBING THE TECHNICAL SYSTEMS MODEL

The subsystems included in the model built for this research are shown in Figure 5-1, and some of the major interactions happening in the model are shown in Figure 5-2. The model is divided into two main parts, the Orbital Analysis part and the Satellite System Analysis part. These, their subsystems, their variables, and the interactions between them are discussed in detail below. The equations discussed in this chapter are listed in Appendix A: EQUATIONS. Specific reference will be given to the relevant sections of that Appendix. Abbreviated units of measure are discussed in Appendix H: UNITS OF MEASURE, and the main VI titles are listed in Appendix G: VI TITLES.

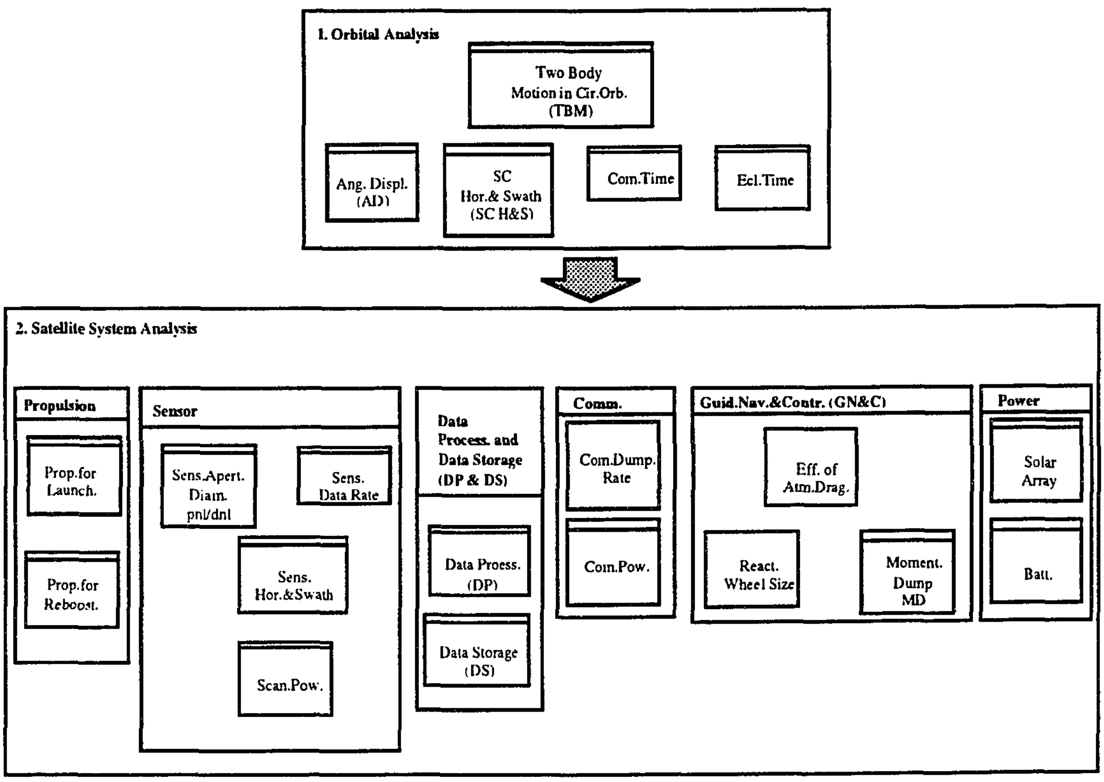


Figure 5-1 Model Hierarchy

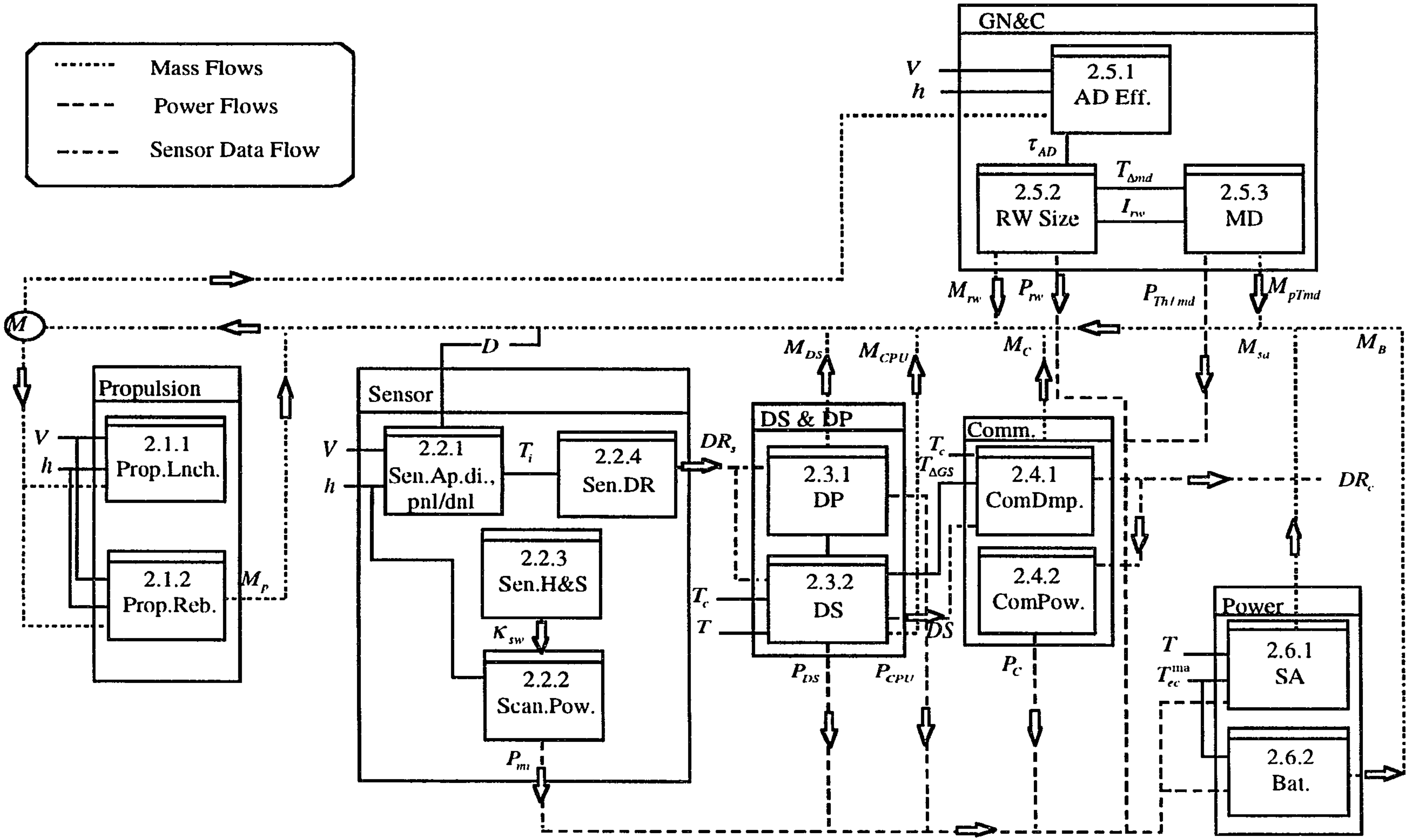


Figure 5-2 Data, Mass and Power Flows in the Model

5.1. Orbital Analysis

5.1.1. Two Body Motion

Newton's Second Law of Motion and his Law of Universal Gravitation form the mathematical basis for all orbital analysis.[1,2,3] Using the mathematical relationships derived through those Laws and assuming circular orbits (eccentricity equals 1) the variables: spacecraft altitude (h), orbital period (T), velocity (V), and angular velocity (ω) can be calculated. The number of orbits the spacecraft should do per day (Q) was chosen as input for these calculations. These calculated variables are used as input for the calculations of other orbital analysis variables such as angular displacement (AD_e & AD_{La}), communication time (T_c), eclipse time (T_{ec}^{max}), and swath width (S_w). All of the variable values calculated in the orbital analysis are fed into the satellite system analysis and used as basis for those calculations.

Appendix A: EQUATIONS, Section 1.1.

5.1.2. Angular Displacement

The ground spot at the equator for a spacecraft in orbit will change westward every orbit by a longitudinal angle of $\Delta\phi$. This movement is called angular displacement. For circular orbits,[4] the angular displacement is determined by two factors, Earth's spin around its own axis and the regression of nodes caused by Earth's equatorial bulge. This bulge leads to vectors of the gravitational force being out of the spacecraft orbital plane causing the orbital plane to precess gyroscopically. Earth's rotation under the spacecraft orbit causes a westward change of the spacecraft ground spot as the nodal regression causes a change counter to the direction of the spacecraft velocity vector. The net change is always westward.[5,6] Angular displacement is calculated both in rad and in km, and angular displacement in km is calculated for spacecraft ground spots both at equator (AD_e) and at a given latitude (AD_{La}). Angular displacement in km changes with cos(La.)

Appendix A: EQUATIONS, Section 1.2.

5.1.3. Spacecraft Horizon and Swath

Maximum spacecraft swath width (S_w), central angle (α_h) and nadir angle (β_h) to the horizon are calculated as functions of both spacecraft altitude (h) and the altitude of Earth position above sea-level.[7] The nadir angle (β_h) is used as input for calculating effective communication time.

Appendix A: EQUATIONS, Section 1.3.

5.1.4. Communication Time

Communication time (T_c) is calculated as a function of the nadir angle to the horizon (β_h), corrected for an angle ε_c. This angle is deducted from the spacecraft nadir angle to the horizon (β_h) to take into account unsatisfactory communication conditions at the edges of the spacecraft horizon. The angle ε_c is typically set at values between 3° and 5°.[8,9] Communication time (T_c) serves as input for calculations of data storage requirements (DS) and required communication dump data (DR_c).

Appendix A: EQUATIONS, Section 1.4.

5.1.5. Eclipse Time

In these calculations, based on the spacecraft's relative distance from the Earth's center and the spacecraft orbital period, maximum (T_{ec}^{max}), minimum (T_{ec}^{min}), and effective eclipse times (T_{ec}^{ef}) are calculated. Eclipse time represents the time a given spacecraft at a given altitude (h) would be in the Earth's shadow relative to the Sun. The calculation of minimum eclipse time (T_{ec}^{min}) is not valid for Sun synchronous orbits.[10] In later calculations where eclipse time is used, eclipse time is, therefore, set equal to maximum eclipse time (T_{ec}^{max}). Eclipse time is used as input for calculating solar array area (A_{sa}) and battery capacity (C_B).

Appendix A: EQUATIONS, Section 1.5.

5.2. Satellite System Analysis

5.2.1. Propulsion

The calculations in the propulsion subsystem are divided into two groups. In the first group, propellant mass for placing the spacecraft in orbit (M_p) is calculated, and, in the second group, propellant mass for on-orbit re-boosting (M_{pTb}) and related hardware mass (M_{rbps}) are calculated. The two groups of calculations are discussed separately in sections 5.2.1.1, Propulsion System, and 5.2.1.2, Propellant for Reboosting.

In both groups of calculations, it is assumed that a Hohman transfer is being used for transferring the spacecraft from a lower to a higher orbit. For this type of orbit transfer, the spacecraft transfers from the lower to the higher orbit following an elliptical transfer orbit with a perigee radius equal to that of the lower orbit and an apogee radius equal to that of the higher orbit (r). It takes half an orbit ($T/2$) to complete a Hohman transfer.[11,12]

Appendix A: EQUATIONS, Section 2.1.

5.2.1.1. Propulsion System

Propellant mass required for placing the spacecraft in orbit (M_p) is only used as an approximation for calculating launch cost. This propellant mass would be consumed by the launcher placing the spacecraft in orbit rather than the spacecraft itself, and it is, therefore, not included in the calculations of spacecraft mass. The orbital velocity of the launcher carrying the spacecraft at the launch position is a function of the Earth's velocity vector, the latitude of the launch site, and the orbital inclination of the launcher. This assumes that the inclination of the launcher orbital plane and that of the spacecraft are the same.

Specific impulse for launcher propulsion systems (Isp) range from 150 to 450 s. Most of these propulsion systems produce thrust up in the 10^6 N range.[13]

Appendix A: EQUATIONS, Section 2.1.1.

5.2.1.2. *Propellant for Reboosting*

The re-boosting propulsion subsystem hardware, as defined in this model, is sized to provide the user specified on-orbit altitude re-boosting capabilities. It includes one propellant tank, the propellant management system, and one liquid propellant re-boosting engine. For a typical space propulsion system, the propellant tank mass (M_{rbp}) is 5-15% of total propellant mass consumed, and the propellant management system mass (M_{rbpm}) represents 20-30% of the propellant tank mass (M_{rbp}).[14] The mass of the re-boosting liquid propellant engine (M_{rbE}) is set as an independent variable. Its mass ranges from approximately 3.76-7.26 kg.[15]

Re-boosting maneuvers are necessary due to the decrease in spacecraft altitude and velocity resulting from atmospheric friction. Atmospheric friction is a function of atmospheric density (ρ) which decreases with altitude (h). This relationship is included in the calculations of the required propellant for re-boosting (M_{pTb}) and in the calculations of required propellant for momentum dumping (M_{pTmd}). Momentum dumping is discussed separately in section 5.2.5, Guidance, Navigation and Control.

In the calculations of required change in velocity (ΔV_c) to re-boost the spacecraft to its original altitude (h), it is assumed that a Hohman transfer is being used. The calculated change in velocity (ΔV_c) to re-boost the spacecraft to its original altitude (h) together with the specific impulse of the selected re-boosting engine (Isp_b) form the input for calculating propellant mass required for each orbital re-boosting (M_{pi}). Specific impulse for these liquid propellant engines (Isp_b) are in the 302-314 s range. Their thrust ranges from $4.45*10^2$ to $4.00*10^3$N.[16] Maximum required engine thrust force (F_E^{Th}) is calculated and used as an indicator for determining mass of the re-boosting engine (M_{rbE}).

Propellant mass per re-boost (M_{pi}) varies with spacecraft mass (M) and remaining propellant mass such that M_{pi} decreases over time. This decrease was taken into account in the calculations of total propellant mass consumed for re-boosting (M_{pTb}) over the spacecraft lifetime (T_{lt}).

Number of re-boosts over the spacecraft lifetime (T_{lt}) is calculated as a function of spacecraft altitudinal descent ($T_{\Delta h}$), and ascent time (T_{Hc}). The calculations of descent time, also called time between re-boosts ($T_{\Delta h}$), include variables such as allowed decrease in spacecraft orbit between re-boosts (Δh_c), altitude (h), spacecraft orbital period (T), and spacecraft mass (M).

Both total propellant mass required for re-boosting (M_{pTb}) and the mass of the corresponding propulsion system (M_{rbps}) are included in the calculations of spacecraft mass.

Appendix A: EQUATIONS, Section 2.1.2.

5.2.2. Sensor

The Earth Observation Sensor is being sized and configured for imaging in the visual (V: $0.3*10^{-6}$ to $0.75*10^{-6}$ m) and infrared (IR: $0.75*10^{-6}$ to $100*10^{-6}$ m) spectrum.[17] The sensor can operate in a scanning or a staring mode. A switch is used for selecting between the two. In scanning mode, mass of a scanning mirror (M_{mi}) and its power requirements (P_{mi}) are added to sensor mass (M_{SENS}) and sensor power (P_{SENS}). The setting of the switch also impacts the calculations in sections 5.2.2.1, Apert.Diam ;pnl;dnl.vi, and 5.2.2.4, Sens. DR.vi. These VI titles are spelled out in full in Appendix G: VI TITLES.

For imaging in the IR spectrum, mass (M_{cc}) and power (P_{cc}) of a cryogenic cooler is added to M_{SENS}, and P_{SENS}, respectively. Values for M_{cc} of 2.39 kg and for P_{cc} of 21 W have been suggested.[18] Mass of the sensor optics (M_o) is calculated as a function of sensor aperture diameter (D).[19,20]

The calculations of the sensor system are divided into four groups. In the first group, sensor aperture diameter (D) is calculated; in the second group, potential scanner power (P_{mi}) and mass (M_{mi}) are estimated; in the third group, sensor field of view (FOV)

and sensor swath width (S_i) are estimated; and in the fourth group, the sensor data rate is calculated (DR_s). These groups of calculations are discussed separately in sections 5.2.2.1, Aperture Diameter, 5.2.2.2, Scanner Power, 5.2.2.3, Sensor Horizon and Swath, and 5.2.2.4, Sensor Data Rate.

Sensor mass (M_{SENS}) and sensor power (P_{SENS}) are included in the calculations of spacecraft mass (M) and spacecraft power(P).

Appendix A: EQUATIONS, Section 2.2.

5.2.2.1. Aperture Diameter

Which variables are included in the equation for calculating sensor aperture diameter (D) depends on whether the dominating system noise is internally generated detector-noise or externally generated photon-noise. The two types are often called detector-noise limited (dnl) and photon-noise limited systems (pnl), respectively.[21] The sensor aperture diameter (D) forms the basis for calculating mass of the sensor optics (M_o). See section 5.2.2, Sensor, for further details.

The signal-noise ratio (S/N) and the electromagnetic flux from the observed object on the ground (F_{em}) and the detector instantaneous field of view ($\Delta\theta$) are included in both sets of variables. Additionally the dnl-set includes the variables sensor bandwidth (δf_n) and area per detector (A_d) and the detector figure-of-merit (D^*) unique for every detector type. The additional variables in the pnl-set are observed wavelength (λ) and detector integration time (T_i).

T_i is a function of dwell time, the time an object stays within a ground pixel (T_d), and the set overlap between ground pixels (γ). A value for γ of 100 indicates that there is a 50% overlap, and a value of 200 indicates that there is no overlap. Dwell time (T_d) is defined as a function of the size of the projected ground pixel (d_{pix}) and spacecraft velocity (V) and number of pixels scanned across-track (N_{pix}). d_{pix} is a function of detector size (d_d), sensor focal length (f), and spacecraft altitude (h). Size of Si (Silicon) detectors operating in the visual spectrum range from $10*10^{-6}$ to $30*10^{-6}$ m. For IR detectors, detector sizes range from $20*10^{-6}$ to $40*10^{-6}$ m for Platinium Silicon (PtSi), Germanium Silicon (GeSi), and Iridium Silicon (IrSi) detectors; and from $40*10^{-6}$ to $60*10^{-6}$ m, for Mercury Cadium Telluride (HgCdTe), and Indium Antimonide (InSb) detectors.[22]

For staring systems N_{pix}=1 and for a scanning system N_{pix} is defined as a function of sensor field of view (FOV) and detector instantaneous field of view ($\Delta\theta$).[23,24] *FOV* is a given input variable defined in section 5.2.2.3, Sensor Horizon and Swath. $\Delta\theta$ is a function of detector diameter (d_d) and sensor focal length (f). D^* the detector figure-of-merit ranges from about $10*10^9$ for PbS and HgCdTe detectors to $10*10^{10}$ for InAs detectors. Both are measured at 77 K.[25] Sensor bandwidth (δf_n) is calculated as a function of integration time (T_i).

The electromagnetic flux per unit area (F_{em}) is calculated using the Planck blackbody radiation law as a function of the observed wavelength (λ), the corresponding equivalent blackbody temperature (t), and sensor bandwidth ($\Delta\lambda$). For observations in the visual wavelengths t=5500 K, and for observations in the infrared wavelengths t=300 K. The calculation of F_{em} at t=5500 K takes into account the distance from the Earth to the Sun (h_{SE}) and its radius (R_s) to give the value of F_{em}, at the Earth rather than at the Sun. It is assumed that $\Delta\lambda$ is sufficiently small to justify a non-integral calculation of F_{em}.[26,27,28,29,30]

The variables that are included in the calculation of (S/N) differ between dnl-systems and pnl-systems. Both calculations include the variables, $\Delta\theta$, F_{em}, d_{pix}, V, T_i, and effective ground pixel size (d_{eff}). d_{eff} is set equal to the given sensor resolution, and can be defined as a function of the diameter of the diffraction limited ground pixel (d_λ), the diameter of the geometrically limited ground pixel (d_{pix}), V, and T_i. The S/N_{dnl} equation additionally includes the variables, δf_n, and A_d.

The S/N ratio and bits per sample (b) form the input variables in the calculation of the effective sensor signal-to-noise ratio (S/N_s^e). This signal-to-noise ratio includes the impact the analog to digital converter (DA-unit) has on the S/N ratio. The value of b typically ranges from 8 to 16.[31,32]

The f-value ($F^{\#}$) is calculated as a function of D and f.[33]

Appendix A: EQUATIONS. Section 2.2.1.

5.2.2.2. Scanner Power

In these calculations, it has been assumed that the scanning mirror oscillates rather than spins, and that each scan covers the full *FOV* angle. Time per scan (T_{mi}) can be calculated as a function of the number of pixels covered per scan (N_{pix}) and integration time (T_i). Both variables are defined in section 5.2.2.1, Aperture Diameter.

Mirror mass (M_{mi}) is defined as a function of the given variables, mirror radius (r_{mi}); mirror thickness (x_{mi}), mirror width (y_{mi}), and the density of the mirror material (υ_{mi}). For beryllium sheet typically used in these mirrors, υ_{mi}=1.85*10^3 kg/m^3.[34,35]

Scanner power (P_{mi}) can be calculated as a function of scanning mirror moment of inertia (I_{mi}); required acceleration and deceleration at beginning and end of scan (ξ_{mi}); percentage of time, T_{mi}, used for this acceleration and deceleration ($p_{\%}$); and T_{mi}.[36]

Required acceleration and deceleration ξ_{mi} can be calculated as a function of *FOV* and $p_{\%}$. The scanning mirror moment of inertia (I_{mi}) is defined as a function of its mass, M_{mi}, radius (r_{mi}), and thickness (x_{mi}).[37,38,39]

Appendix A: EQUATIONS. Section 2.2.2.

5.2.2.3. Sensor Horizon and Swath

The sensor field of view angle (*FOV*) is a given variable. Its value cannot exceed that of two times the spacecraft nadir horizon angle ($2*\beta_h$). And it, together with the radius of the observed position (R_P), define the function for calculating effective sensor swath width (S_i).[40] *FOV* is also used as input in equations in sections 5.2.2.1, Aperture Diameter, 5.2.2.2, Scanner Power, and 5.2.2.4, Sensor.

Appendix A: EQUATIONS, Section 2.2.3.

5.2.2.4. Sensor Data Rate

The data rate being generated by the sensor (DR_s) is defined as a function of the number of pixels observed simultaneously by spacecraft sensor per channel or band ($N^{sim}_{pix/ch}$); number of samples per pixel (s), bits per sample (b), sensor frame efficiency (e_{SF}), number of channels in which data is being acquired (N_{ch}), and integration time (T_i). N_{ch} is a given variable. Typical values for e_{SF} are between 90 and 95%, and for s, number of samples per pixel, typical values range from 1.4 and 1.8.[41,42] The variable b is defined in section 5.2.2.1, Aperture Diameter.

The values of the variables $N^{sim}_{pix/ch}$ and T_i depend on the setting of the scanner/staring switch. For a scanning system, $N^{sim}_{pix/ch}$ represents the set number of pixels scanned simultaneously or in parallel ($N^{s/sc}_{pix/ch}$). For a staring system $N^{sim}_{pix/ch}$ equals the total number of detector elements in the array. Number of detector elements is a function of sensor field of view, FOV, and detector instantaneous field of view, $\Delta\theta$.

There are two types of staring systems, the pushbroom and the full staring system. For pushbroom systems, the detector array consists of only one line of detectors placed perpendicular to the along-track direction. For full staring systems, it is assumed that the detector array has an equal number of detectors along-track as across-track.[43] Sizes for space certified detector arrays range, for visual systems, from 400x400 (160,000) to 2000x2000 (4,000,000). Arrays with $1*10^9$ detectors are anticipated in the near future.[44] Detector arrays of 12064x12064 (145,540,096) have already been made for the aeronautical sector.[45] For IR systems, detector array sizes range from 256x256 (65,536) to 512x512 (262,144).[46]

DR_s, sensor data rate, is used as input in the calculations in section 5.2.3, Data Processing and Data Storage.

Appendix A: EQUATIONS, Section 2.2.4.

5.2.3. Data Processing and Data Storage

The data processing and data storage (DP & DS) system is dimensioned to store and process the data flows being generated by the sensor and the spacecraft housekeeping (HK) systems. The DP & DS system consists of one data processing unit (CPU) and one data storage unit (DS). Calculations are grouped accordingly into two groups, a data processing group and a data storage group. These are discussed separately in sections 5.2.3.1, Data Processing, and 5.2.3.2, Data Storage.

In line with the assumptions made in the model, the following housekeeping systems (HK) are included in these calculations: Communications [Command and Processing (CP), Telemetry (T)]; Attitude Sensor Processing [Rate Gyro (RG), Earth Sensor (ES), Star Tracker (ST)]; Attitude Determination & Control [Kinematic Control (KI), Error Determination (ED), Thruster Control (ThrC), Reaction Wheel Control (RW), Ephermeric Propagation (EP), Complex Ephermeris (CE), Orbit Propagation (OP)]; Fault Detection [Monitors (M), Fault Correction (FC)]; and Other Functions [Power Management (PM), Thermal Control (TC), and Kalman Filter (KF)]. [47] For each of these systems, the number of samples generated per second (N_s^i), number of bits per sample ($N_{b/s}^i$), and number of instructions required to process each sample ($N_{i/s}^i$) are given as independent input variables.

Appendix A: EQUATIONS, Section 2.3.

5.2.3.1. Data Processing

The data processing unit (DP) always processes the housekeeping (HK) data flow. The sensor data flow is only processed if on-board processing has been selected. It is assumed that on-board processing compresses the sensor data flow to $K_{\%}$. The compression rate ($K_{\%}$) is a given input variable. Suggested values for $K_{\%}$ are between 50 and 90 %.[48] A low value of $K_{\%}$, i.e., high compression, should, in general, indicate that a higher number of instructions per sample ($N_{i/s}^s$) is required to process the sensor data flow.

Typically, $K_{\%}$=50% would require a $N_{i/s}^{s}$ in the 40-60 range.[49] Both input variables $K_{\%}$ and $N_{i/s}^{s}$ are dependent on the type of compression algorithm assumed used by designers.[50] The compression rate ($K_{\%}$) also impacts the data storage requirements as discussed in section 5.2.3.2, Data Storage. In the model, it is assumed that no compression is included in the processing of housekeeping data.

Processing power (N_{TIPS}) of the data processing (DS) unit is measured in thousand/million instructions per second (KIPS/MIPS). The general equation for calculating processing power takes into account both the number of instructions required to process each data bit, as well as the number of instructions required to process each sample. In this model, though, it is assumed that processing power (N_{TIPS}) can be calculated accurately by including in the calculations only the instructions required to process each housekeeping and data sample ($N_{i/s}^{i}$ and $N_{i/s}^{s}$).[51,52] The number of samples generated by a sensor per second (N_s^s) is calculated as a function of the sensor data rate (DR_s). The number of samples generated by each housekeeping system (N_s^i) is provided as given input.

Regression analysis indicates that processing power (N_{TIPS}) is proportional to the power (P_{CPU}) and mass (M_{CPU}) requirements of the data processing system. The factors ζ_{PCU}^{p} and ζ_{CPU}^{m} describing these power and mass relationships are determined by the processor technology selected.[53] The model includes processors with processing power levels ranging from 1.200 KIPS to 20.000 KIPS.[54] A switch in the program is used for selecting the appropriate technology.

Mass (M_{CPU}) and power (P_{CPU}) of the data processing system are included in the calculations of spacecraft mass (M) and power (P) requirements.

Appendix A: EQUATIONS, Section 2.3.1.

5.2.3.2. Data Storage

The data storage (DS) system is sized to store housekeeping and sensor data between the time this data is being generated and the time it can be downloaded to a ground station (GS). In the model, it is assumed that the housekeeping systems generate a constant data rate (DR_{HK}) during the complete spacecraft orbit, and that the spacecraft sensor generates a constant data rate (DR_s) only during the time it is turned on.

Between downloads, the data storage system stores housekeeping data generated during the time from the last data download ($T_{\Delta GS}$) plus the data generated during the time of that download (T_c). The data storage requirement by the housekeeping systems (DS_{HK}) can, therefore, be defined as a function of the variables DR_{HK}, $T_{\Delta GS}$, and T_c. DR_{HK} is a function of number of samples generated per second (N_s^i) by the housekeeping systems, and the corresponding number of bits per sample ($N_{b/s}^i$), both given input variables.

The sensor can be turned on during the time between ground stations ($T_{\Delta GS}$), but not during data downloads, defined as communication time (T_c). During that time all spacecraft systems are fine-tuned for communication rather than for sensor data generation. The time the sensor is turned on between ground stations (ϑ_T) is a given variable. This variable and the variables DR_s, $T_{\Delta GS}$, and $K_{\%}$ are included in the equation for calculating the sensor data storage requirement (DS_s). $K_{\%}$ is the compression data rate as defined in Section 5.2.3.1, Data Processing.

Two types of data storage devices are included in the model, the tape recorder and the solid state recorder. Their storage capacity limits are $2*10^9$ bits and $75*10^9$ bits, respectively.[55] A switch is used for selecting between the two. Mass (M_{DS}^T) and power (P_{DS}^T) estimates for the tape recorder are functions of the total data storage requirement (DS).[56]

The mass of the solid state recorder (M_{DS}^{SS}) can be calculated as a function of the variables, total required data storage (DS), base storage capacity (DS_F^{SS}), incremental data storage capacity (DS_{inc}^{SS}), mass of incremental storage capacity (M_{inc}^{SS}), and base mass (M_F^{SS}). Power of the solid state recorder can, likewise, be calculated as a function of DS, DS_{inc}^{SS}, power required per incremental storage capacity (P_{inc}^{SS}), and base power (P_F^{SS}).[57] Suggested values for these variables are: DS_F^{SS}=128*10^6 bits; DS_{inc}^{SS}=64*10^6 bits; M_{inc}^{SS}=0.9 kg; M_F^{SS}=6.17 kg; P_{inc}^{SS}=0.4376 W;[58] P_F^{SS}=3 W.[59]

The mass (M_{DS}) and power (P_{DS}) of the data storage system are included in the calculations of spacecraft mass (M) and power (P). Data storage requirements for the housekeeping systems (DS_{HK}) and the sensor (DS_s) serve as input for calculating the communication dumprate (DR_c) in section 5.2.4.1, Communication Dumprate.

Appendix A: EQUATIONS, Section 2.3.2.

5.2.4. Communication

The communication system is sized to dump, during available communication time (T_c) over any ground station, the sensor data (payload data) and housekeeping data (telemetry data) stored on the data storage (DS) system since the last communication downlink. The communication system consists of two transmitter/receiver units, filters/switches/diplexers, two hemispheric antennas (uplink), and one parabolic antenna (downlink).

The calculations in this subsystem are divided into three main groups. The required dumprate from spacecraft (SC) to ground station (DR_c) is calculated in the first group; in the second group, the communication system power requirements (P_{COM}) are calculated; and in the third group, mass of the communication system (M_{COM}) is calculated. These groups of calculations are discussed separately in sections 5.2.4.1, Communication Dumprate, 5.2.4.2, Communication Power, and 5.2.4.3, Communication Mass.

The communication (downlink) wavelength (λ_c), a given variable, is used in the calculations in sections 5.2.4.2, Communication Power, and 5.2.4.3, Communication Mass. Typically, remote sensing satellite downlinks are provided in the S-band (2.2-2.3

GHz), C-band (3.7-4.2 GHz), X-band (Military: 7.25-7.75 GHz), Ku-band (12.5-12.75 GHz), and the Ka-band (17.7-19.7 GHz). Most current systems use the S-band.[60,61]

Communication power (P_{COM}) and mass (M_{COM}) are included in the calculations of spacecraft mass (M) and power (P).

Appendix A: EQUATIONS, Section 2.4.

5.2.4.1. Communication Dumprate

Communication dumprate (DR_c) is calculated as a function of the volume of sensor data (DS_s) and housekeeping data (DS_{HK}) stored on the data storage system since the last data dump ($T_{\Delta GS}$), communication time (T_c), set up time (T_{ca}), and buffer time (T_{cb}). Set up time (T_{ca}) and buffer time (T_{cb}) are set at 30 seconds each and are estimates of the time it takes to set up and break down a communication link.[62,63]

The communication dumprate (DR_c) variable is included in the calculations in Section 5.2.4.2, Communication Power.

Appendix A: EQUATIONS, Section 2.4.1.

5.2.4.2. Communication Power

There are two types of units that require power in the communication systems, the transmitter and the receiver. Required power for the transmitter (P_{COM}^T) is calculated from the link equation as a function of the communication data rate (DR_c), signal-to-noise ratio (E_b/N_0), gain of spacecraft transmitting antenna (G_t), and gain of ground station receiving antenna (G_r), transmitter efficiency (e_T), and various path loss factors. Transmitter power is used as input in the equations in section 5.2.4.3, Communication Mass.

For the signal-to-noise ratio, ranges between 5 and 10 have been suggested. Gain of the transmitting (G_t) and receiving (G_r) antennas can, based on the link equation, be defined as functions of their respective antenna diameters (D_{ta} and D_{ra}), their respective efficiencies (ϖ_{ta} & ϖ_{ra}), and the carrier wavelength (λ_c).[64] Diameter of the spacecraft transmitting antenna (D_{ta}) is typically between 0.7 and 2.5 m.[65] Ground station receiving antenna diameters are larger and range from 4 to 18 m.[66] Antenna efficiency, a figure of merit between 0 and 1, range from 0.6 to 0.7 for well designed ground station receiving antennas. Lower values, around 0.55, are typical for spacecraft transmitting antennas.[67] High quality traveling wave tube (TWT) and solid state (SS) transmitter amplifiers have efficiencies between 35 and 45%.[68] For this model, it is assumed that the lighter solid state amplifiers are being used.

In the model, power required by the receiver (P_{COM}^R) is assumed fixed for each communication band. Suggested values of P_{COM} are 17.5 kg for the S-band and 10.4 kg for the X-band.[69] Data for the other communication bands were not available.

Appendix A: EQUATIONS, Section 2.4.2.

5.2.4.3. Communication Mass

In the model, it is assumed that the transmitter and receiver are bundled together as one mass unit. The mass of this unit is a function of required transmitter power (P_{COM}^T) and a given constant (ζ_{COM}'''). This constant defines for each communication band the relationship between required transmitter power (P_{COM}^T) and mass of the transmitter/receiver unit (M_{TR}). A value of 0.136 is suggested for the X-band and 0.153 for the S-band. Estimated mass of the filter/diplexer/switch (M_{FIL}) is set as a given input for each communication band. Suggested values are 1.5 kg for the X-band and 2 kg for the S-band.[70]

Mass of the hemispheric antenna (M_{HEM}) is defined as an independent input variable ranging from 0.25 to 0.4 kg.[71] For the parabolic antenna used for downlinks, mass (M_{ANT}''') is calculated as a function of antenna diameter (D_{ta}) and a constant (ζ_{ANT}''') defining the relationship between D_{ta} and M_{ANT}'''. In the model, ζ_{ANT}''' has been defined for some antenna types. Some suggested values are: parabola - fixed (S-band), 5.57, and parabola with feed array (C-band), 12.05.[72]

Appendix A: EQUATIONS. Section 2.4.3.

5.2.5. Guidance, Navigation and Control

The Guidance, Navigation, and Control (GN&C) System includes sensors for acquiring position data and control mechanisms for controlling spacecraft attitude. Altitude and velocity corrections are taken care of by the propulsion system.

The calculations in this subsystem are divided into three groups. In the first group, the main external torques working on the spacecraft are being calculated; in the second group, calculations related to the reaction wheel are being performed; and, in the last group, variables such as thrust force ($F_{Th/md}$) and propellant mass (M_{pTmd}) required for momentum dumping are being calculated. These groups of calculations are discussed separately in sections 5.2.5.1, Effective Atmospheric Drag, 5.2.5.2, Sizing the Reaction Wheel, and 5.2.5.3, Momentum Dumping.

A spacecraft's attitude changes through the working of external and internal torques. Internal torques can be generated from thruster misalignment, mismatch in thruster output, pumps, tape recorders, and scanning mirrors. The impact of these torques are not included in the calculations in the model as it is assumed that they either cancel themselves or each other out.[73,74]

External torques can be caused by the Earth's changing gravity field (the gravity-gradient) solar radiation, the Earth's magnetic field, and atmospheric friction. [75,76,77] For orbits lower than 500 km, atmospheric friction torque dominates.[78,79] In this model, therefore, only the impact of this type of torque (τ_{AD}) has been included in the calculations.

To facilitate high resolution imaging (1-5 m) and achieve the highest level of attitude control accuracy, between 10^{-3}° and 1°, the GN&C system is configured to make the spacecraft zero-momentum three-axis stabilized. Such a system includes one reaction wheel per spacecraft axis (x,y,z) and torquers for periodic dumping of the angular momentum (H_{rw}^{max}) built up in the reaction wheels.[80,81] In this model, electrical or chemical thrusters are used for periodic dumping of angular momentum.

The attitude determination system was to attain the same high level of accuracy configured to include a star tracker and an Earth sensor and a gyroscope. The star tracker provides a high 10^{-4} ° (arc sec) accuracy. Other sensors such as Sun sensors and magnetometers might be lighter, require less electrical and processing power, but their accuracy levels are lower, starting at 10^{-2} °(arc min) and 0.5°, respectively.[82,83,84] In Low Earth Orbits (LEO), Earth is the second brightest celestial object and covers up to 40% of the spacecraft's sky.[85,86,87] This makes the Earth sensor, despite its lower accuracy, 0.1° to 1°,[88] well suited for measuring the spacecraft's position relative to Earth. The gyroscope, with accuracy 0.003°[89] per hour, was added to give the GN&C system a high accuracy attitude reference in the periods between star observations.[90] Often gyroscopes are referred to as inertial sensors, and star trackers, Earth sensors, Sun sensors, and magnetometers are referred to as reference sensors.

The mass and power of the star tracker (M_{st}, P_{st}), the Earth sensor (M_{es}, P_{es}), and the gyroscope (M_{gs}, P_{gs}) are given as independent input variables. Some suggested values for these variables are: M_{st}=7.7 kg; P_{st}= 18 W; M_{es}= 2.5 kg; P_{es}= 8 W; M_{gs}= 0.8-3.5 kg; P_{gs}= 5-20 W.[91,92,93]

Appendix A: EQUATIONS, Section 2.5.

5.2.5.1. Effective Atmospheric Drag

Atmospheric friction in addition to slowing down and reducing spacecraft altitude (h) also creates a torque on the spacecraft (τ_{AD}). This torque is a function of the force being generated on the spacecraft (F_{AD}) by the atmospheric friction and the distance (r_{cp}) between the spacecraft center of mass and center of pressure for each axis.[94,95,96] The force, F_{AD}, is a function primarily of altitude (h), velocity (V), and atmospheric density (ρ).[97,98] Torque (τ_{AD}) together with spacecraft orbital period (T) and spacecraft moment of inertia (I_{SC}) are used for calculating generated angular momentum ($H_{AD/O}$) on the spacecraft per orbit and change in spacecraft pointing ($\Delta\varphi_{AD/O}$) per orbit.[99,100,101] The calculations of the variables $H_{AD/O}$ and $\Delta\varphi_{AD/O}$ are included to show the impact on the spacecraft from atmospheric friction torque (τ_{AD}) if no control mechanism had been in place. The atmospheric friction torque (τ_{AD}) is also used as input for calculating the reaction wheel parameters.

Appendix A: EQUATIONS, Section 2.5.1.

5.2.5.2. Sizing the Reaction Wheel

The reaction wheel system includes three reaction wheels, one for each axis, and one wheel drive electronics' unit. Mass (M_{rwe}) of the wheel drive electronics' unit is given, and it ranges from 1.9 and 3.9 kg.[102] To maintain spacecraft attitude, the reaction wheels have been dimensioned to generate internal torques that equal that of the atmospheric friction torque (τ_{AD}). These torques are being generated through accelerating the reaction wheels. The reaction wheel mass (M_{rw}) required to generate such a torque is a function of atmospheric friction torque (τ_{AD}), a given angular acceleration of the reaction wheels (ξ_{rw}), and a given reaction wheel radius (r_{rw}). Common reaction wheel radiuses range from 0.1 to 0.25 m.[103] Together, reaction wheel mass (M_{rw}) and reaction wheel radius (r_{rw}) form the input into the calculations of the reaction wheel moment of inertia (I_{rw}).[104,105]

Reaction wheels have a set maximum angular velocity (ω_{rw}^{max}) of about 6000 rpm ($6.28*10^2$ rad/sec).[106] When this velocity has been reached, further acceleration becomes impossible and the reaction wheel looses its torqueing effect. To despin the reaction wheels, their built up angular momentums have to be dumped. This issue is discussed in section 5.2.5.3, Momentum Dumping.

Assuming constant angular acceleration (ξ_{rw}) of the reaction wheel, time between momentum dumps ($T_{\Delta md}$) can be calculated as a function of angular velocity (ω_{rw}^{max}) and angular acceleration (ξ_{rw}). The power (P_{rw}) required to accelerate each reaction wheel to the given (ξ_{rw}) can be calculated as a function of reaction wheel moment of inertia (I_{rw}), acceleration (ξ_{rw}), and time between momentum dumps ($T_{\Delta md}$). Atmospheric friction torque (τ_{AD}) is included in this calculation through the moment of inertia (I_{rw}) variable.

Moment of inertia (I_{rw}) is also used as input into the calculations described in 5.2.5.3 Momentum Dumping. Reaction wheel system power (P_{rw}) and mass (M_{rws}) are included in the calculations of spacecraft power (P) and mass (M), respectively.

Appendix A: EQUATIONS, Section 2.5.2.

5.2.5.3. Momentum Dumping

The thruster propulsion system consists of three thrusters (one for each axis) and propellant tank(s) and a propellant management system. For a typical thruster system, the propellant tank mass (M_{mdpt}) is 5-15% of total propellant mass consumed, and the propellant management system mass (M_{mdpm}) represents 20-30% of the propellant tank mass (M_{mdpt}).[107]

The thruster propulsion system is sized to create sufficient torque (τ_{Th}) to dump the angular momentum (H_{rw}^{max}) built up in the reaction wheel during the time ($T_{\Delta md}$) between momentum dumps. Sometimes the term desaturation or momentum unloading is used instead of momentum dumping.

Angular momentum (H_{rw}^{max}) is calculated as a function of moment of inertia (I_{rw}) and maximum angular velocity (ω_{rw}^{max}) of each reaction wheel.[108,109,110] The torque (τ_{Th}) that each thruster has to generate is a function of this angular momentum and the specified burn time ($T_{b/p}$) per thruster pulse.[111] Typical values for $T_{b/p}$ are between 0.02 and 0.1 sec.[112] Thrust force ($F_{Th/md}$) or thrust can be calculated as a function of required torque (τ_{Th}) and the distance ($l_{Th\text{-}a}$) from the spacecraft principal axes to each thruster.[113] The calculated thrust level is used as an indicator for setting thruster mass (M_{mdT}). Typical thruster mass range from 0.1 to 2.3 kg.[114]

Two types of thruster propulsion systems are used, chemical and electrical. The model and the program has been set up so that the user can switch between the two types.

If a chemical propulsion system has been selected, only propellant mass ($M_{p/md}$) is calculated. For the electrical propulsion system, the model calculates both propellant mass ($M_{p/md}$) and power ($P_{Th/md}$). In general, if the thrust requirements are low, from 10^{-6} N to a maximum of 5 N, and a specific impulse (Isp_{Th}) higher than 225 s is requested, an electrical propulsion system would be preferred. Otherwise, a chemical propulsion system should be selected.[115,116]

Thruster propellant mass for both types of propulsion system can be calculated as a function of reaction wheel angular momentum (H_{rw}^{max}), specific impulse (Isp_{Th}), and distance ($l_{Th\text{-}a}$) from the spacecraft principal axes to each thruster. The function for calculating electrical power ($P_{Th/md}$) for the electrical propulsion system includes the same variables, as well as burn time ($T_{b/P}$) per thruster pulse and electric propulsion system efficiency (e_{elTh}). Typical values for propulsion system efficiency (e_{elTh}) are 0.9 for a resistojet thruster, 0.3 for an arcjet thruster, and 0.75 for an ion thruster.[117,118]

In the calculation of total propellant consumed (M_{pTmd}) by the momentum dumping thrusters over the spacecraft lifetime (T_{lt}), it is assumed that propellant consumed ($M_{p/md}$) per momentum dump remains constant over the spacecraft lifetime (T_{lt}). Given the small amounts of propellant consumed by the thrusters relative to spacecraft mass (M), this is a fair approximation.

The mass of the thruster propellant system (M_{mdps}), which includes the mass of propellant consumed (M_{pTmd}), is included in the calculations of total spacecraft mass (M). The electrical power ($P_{th/md}$) consumed by possible electrical thrusters is included in the calculations of spacecraft power (P).

Appendix A: EQUATIONS, Section 2.5.3.

5.2.6. Power

The power system is being configured and sized to meet the power requirements of all spacecraft subsystems, both during eclipse (P_{ec}) and during the time the spacecraft is in sunlight (P_d). It includes solar panels and a battery unit and a power control unit (PCU) and a regulator/converter unit and power wiring. In the model, it is assumed that $P_d = P_{ec}$.

Specific mass for the power control unit (ζ_{PCU}''') is set to 0.02 kg/W of required power (P_d). The corresponding variable (ζ_{CR}''') for the regulator/converter unit has been set to 0.025 kg/W.[119,120] The constant (ζ_{PW}''') that describes the relationship between power wiring mass (M_{PW}) and spacecraft mass (M) ranges from 0.01 to 0.04.[121]

The calculations in the power system are divided into two groups. The first group includes the solar array related calculations, and the second group includes the battery related ones. These groups of calculations are discussed separately in sections 5.2.6.1, Solar Array, and 5.2.6.2, Battery.

The mass of the power system (M_{POW}) is included in the calculations of spacecraft mass (M).

Appendix A: EQUATIONS, Section 2.6.

5.2.6.1. Solar Array

The power that the solar array has to produce (P_{sa}) during the full orbit can be calculated as a function of required power during eclipse (P_{ec}) and sunlight (P_d); duration of the eclipse (T_{ec}^{max}); and efficiency in the power path from the solar array directly to the spacecraft subsystems (e_{SS}), and in the power path from the solar array via the battery to the spacecraft subsystem (e_{SBS}). The values of e_{SS} and e_{SBS} depend on the type of power regulation system being utilized. Two types are in use, peak-power tracking (PPT) and direct energy transfer (DET). A switch is used for selecting between the two. Suggested values of the power path efficiencies are e_{SS}=0.8 and e_{SBS}=0.6 for a PPT, and e_{SS}=0.85 and e_{SBS}=0.65 for a DET system.[122]

The solar array area (A_{sa}) required for providing the power P_{sa} can be calculated as a function of P_{sa}, the power that can be produced by solar array per unit area at spacecraft end of life (P_{EOL}), and solar cell packing density (q_{cs}). For well designed solar arrays, values of 90% for q_{cs} are possible.[123]

P_{EOL} is a function of the power that can be produced by the solar array per unit area at spacecraft beginning of life (P_{BOL}), and the remaining efficiency of solar cells at spacecraft end of life(e_{EOL}). e_{EOL} can be calculated as a function of spacecraft lifetime (T_{lt}) and annual degradation ($\Delta e_{s/y}$) of the selected type of solar cell. For Silicon (Si) cells employed in LEO orbits, the worst-case value for $\Delta e_{s/y}$ equals 3.75%, and for Gallium-Arsenide (GaAs) cells the worst case value equals 2.75%.[124]

P_{BOL} can be calculated as a function of optimal power output by solar array per unit area (P_o), inherent solar cell degradation (I_d), and the worst-case solar incidence angle (η). The solar incidence angle is measured between the Sun line and a vector normal to the solar array surface. Inherent solar cell degradation (I_d) can be seen as a function of design and assembly, degree of shadowing of cells, and estimated solar array temperature. Values between 49% and 88% are suggested. Optimal power output, P_o, can be calculated as a function of solar incidence radiation (P_s) and solar cell efficiency (e_s).[125]

Values for both solar cell efficiency (e_s) and annual solar cell degradation ($\Delta e_{s/y}$) depend on the type of solar cell selected. In the model, a switch is used for selecting between the defined solar cell types: Si and GaAs.

The number of solar cells required (N_{CS}) to produce the end-of-life power (P_{EOL}) can be calculated from required solar array area (A_{sa}), area of each solar cell (A_{cs}), and solar cell packing density (q_{cs}). Solar cell area (A_{cs}) is typically 0.02*0.04 m, but larger 0.05*0.05 m cells are under development.[126,127,128,129]

The mass of the solar array (M_{SA}) can be calculated as a function of solar array area (A_{sa}) and a constant χ_A. The value of χ_A varies with type of solar panel and type of solar cell chosen. For a Si solar cell rigid-array, χ_A=3.08, and for a GaAs solar cell rigid-array, χ_A=4.09.[130] A switch is used for selecting solar array type.

Appendix A: EQUATIONS, Section 2.6.1.

5.2.6.2. *Battery*

Required battery capacity (C_B) to power all spacecraft systems during the eclipse time can be calculated as a function of required spacecraft power during the eclipse (P_{ec}), eclipse time (T_{ec}^{max}), battery depth of discharge (DoD_l), the average discharge level of the batteries (EP_{av}^{B}), and transmission efficiency between battery and spacecraft systems (e_{BS}).

Depth of discharge (DoD_l) represents the percentage of total battery capacity removed during the eclipse period. Values for DoD_l range between 40 and 60 % for Nickel Hydrogen (NiH2) batteries, and between 10 and 20 % for Nickel Cadmium (NiCd) batteries. There are two types of NiH2 battery designs, the independent pressure vessel (ipv) design and the common pressure vessel (cpv) design.

For EP_{av}^{B}, suggested values are in the 26-28 V range, and for the transmission efficiency (e_{BS}) a value of 90 % has been suggested.

Battery mass (M_{BAT}) is calculated as a function of battery energy (E_{BAT}) and the inverse specific energy (χ_E). E_{BAT} is a function of the battery unit's capacity (C_B) and its average discharge level (EP_{av}^{B}). The inverse specific energy density (χ_E) is dependent on battery type. For NiH2ipv batteries, the χ_E values range from 1/25 to 1/40; for NiH2cpv batteries, the χ_E values range from 1/45 and 1/60; and for NiCd batteries, the χ_E values range from 1/25 to 1/30.[131]

The values of DoD_l and the inverse energy density (χ_E) are selected through the battery class switch.

Appendix A: EQUATIONS, Section 2.6.2.

---

[1] Brown, C.D., *Spacecraft Mission Design*, AIAA Education Series, American Institute of Aeronautics and Astronautics, Washington, D.C., 1992, 5-8.
[2] Stark, J. P. W., Celestial Mechanics, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 59-81.
[3] Boden, D.G., Introduction to Astrodynamics, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 114-117.
[4] Brown, C.D., 63.
[5] Ibid., 59-70.
[6] Stark, J. P. W., Mission Analysis, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 98-100.

[7] Brown, C.D., Fig. 4.12, 72 & 73.
[8] Ibid., 77.
[9] Wertz, J.R., Space Mission Geometry. in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London. Kluwer Academic Publishers, 1991, 100.
[10] Chetty, P. R. K., *Satellite Technology and its Applications*, 2nd ed., Blue Ridge Summit, PA, Tab Books, 1991, 157-158.
[11] Brown, C.D., 43-44.
[12] Boden, D.G., 129-130.
[13] Sackheim, R.L., Wolf, R.S., Space Propulsion Systems, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 584.
[14] Ibid., 602.
[15] Ibid., Table 17-3, 585.
[16] Ibid.
[17] Brodsky, R.F., Defining and Sizing Space Payloads, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Fig. 9-3, 219.
[18] Conte, D., Payload Overview, System Design Audit (SDA), LightSats, TRW, August 1994, 204-205.
[19] Telecon with F. Kingsley, Space Optics Research Lab, July 24, 1995.
[20] Appendix C: EMPIRICALLY DERIVED EQUATIONS, 151.
[21] Jamieson, J.A., McFee, R.H., Plass, G.N., Grube, R.H., Richards, R.G., *Infrared Physics and Engineering*, New York, McGraw-Hill Book Company, 1963, 601-611.
[22] Cantella, M.J., Staring-Sensor Systems, in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Copublished by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington, 1993, Table 3.1, 162.
[23] Acetta, J.S., Infrared Search and Track Systems, in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Copublished by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington, 1993, 296-297.
[24] Meeting with S. Katzberg, March 16, 1995.
[25] Brodsky, R.F., Fig. 9-6, 234.
[26] Ibid., 220.
[27] Hopper, G.S., Forward-Looking Infrared Systems, in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Copublished by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington, 1993, 107-108.
[28] Notes from meeting with S. Katzberg, February 1, 1995, 5.
[29] Reeves, R. G., Anson, A., Landen, D. (Editors), *Manual of Remote Sensing*, Vol.1, American Society of Photogrammetry, Falls Church, VA, 1975, 60-61.
[30] Jamieson, J.A., 13-14.
[31] Notes from meeting with S. Katzberg, July 19, 1995.
[32] Davies, R.S., Communications Architecture, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*. London, Kluwer Academic Publishers, 1991, Table 13.4 and Fig 13.4, 452-453.
[33] Brodsky, R.F., 250.
[34] Doukas, P.G., McCandless, J.R., Sarafin, T.P., Britton, W.R., Structures and Mechanisms, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 11.41, 394.
[35] Notes from meeting with S. Katzberg, March 16, 1995, 3.
[36] Notes from telecon with S. Katzberg, April 18, 1995, 2-3; Notes from meeting with E. L. Dahlstrom April 19, 1995, 1-2.
[37] Notes from meeting with S. Katzberg, March 16, 1995, 5.

[38] Chetty. P. R. K., Appendix A (Reprint: CRC Handbook of Chemistry and Physics, 71st Edition. Copyright CRC Press, Inc., Boca Raton, FL), 513.
[39] Doukas. P.G., 407.
[40] Brown. C.D., Fig. 4.14, 75.
[41] Brodsky. R.F., 250.
[42] Notes from meeting with S. Katzberg, March 23, 1995, 4.
[43] Kramer. H.J., *Observation of the Earth and Its Environment, Survey of Missions and Sensors*, 2nd ed.. Berlin, Springer Verlag, 1994, Figure 147, 479.
[44] Cantella. M.J., Table 3.1, 162.
[45] Dornheim. M.A., New Sensors Show Two Paths to Reconnaissance, *Aviation Week and Space Technology*, A McGraw-Hill Publication, July 10, 1995, 44-45.
[46] Cantella. M.J., Table 3.1, 162.
[47] Glaseman. S., Hansen, L.J., Pollock, C.H., Thimlar, M., Spacecraft Computer Systems, in Wertz, J.R.. Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 16-6, 560.
[48] Davies. R.S., 456.
[49] Telecon with G. Ganoe, July 27, 1995.
[50] Meeting with G. Ganoe, March 24, 1995.
[51] Ibid.
[52] Glaseman. S., 545.
[53] Meeting with G. Ganoe, March 24, 1995.
[54] Hansen, L.J., Pollock, C.H., Spacecraft Computer Systems, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, 2nd ed., London, Kluwer Academic Publishers, 1992, Table 16-10, 626.
[55] Boatwright, J.E., Mueller, R.L., Command and Data Handling, in Wertz, J.R., Larson, W.J. (Editors). *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991. Table 11-26, 348.
[56] Appendix C: EMPIRICALLY DERIVED EQUATIONS, 151.
[57] Meeting with George Ganoe, March 24, 1995.
[58] Assuming that the same relationship exists between the Mass increment and Fixed Mass as between the Power increment and Fixed Power, the power increment equals
3 * (0.9/6.17)=0.4376.
[59] Boatwright, J.E.. Table 11-26, 348.
[60]Davies. R.S., Table 13-11, 474.
[61] Ford. J.. Communications, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 11-21, 340.
[62] Whitworth, G. G., Ground System Design and Sizing, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 539.
[63] Meeting with George Ganoe, March 21. 1995.
[64] Davies. R.S., 458-461.
[65] Ibid.. Table 13-15, 482.
[66] Whitworth, G. G., Table 15-5, 536.
[67] Davies. R.S., 459.
[68] Smith. H.. Telecommunications, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 343-345.
[69] Ford. J.. Table 11-22, 341.
[70] Ibid.
[71] Ibid.
[72] Davies. R.S., Table 13-15, 482.
[73] Zermuehlen, R.O., Zimbelman, H.F., Guidance, Navigation, and Control. in Wertz, J.R., Larson. W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers. 317.

[74] Griffin. M. D.. French, J. R.. *Space Vehicle Design*, AIAA Education Series. American Institute of Aeronautics and Astronautics, Washington, D.C.. 1991, 296.
[75] Zermuehlen. R.O., 314-316.
[76] Griffin. M. D.. 292-297.
[77] Fortesque. P. W.. Attitude Control, in Fortesque, P. W., Stark. J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 239-243.
[78] Ibid.. Table 10-1, 240.
[79] Zermuehlen. R.O., Table 11-7a/b, 315-316.
[80] Agrawal, B. N., *Design of Geosynchronous Spacecraft*, Englewood Cliffs, NJ, Prentice-Hall, 1986, 130.
[81] Zermuehlen, Table 11-6, 311 & 313.
[82] Griffin, M. D., 316 & Table 7.2, 319.
[83] Fortesque, P. W., Table 10.2, 247 & 249.
[84] Zermuehlen, R.O., Table 11-12, 323.
[85] Chetty, P. R. K., 179.
[86] Fortesque, P. W., 249.
[87] Griffin. M. D., 313.
[88] Zermuehlen. Table 11-12, 323.
[89] Ibid.
[90] Griffin, M. D.. 317.
[91] Chetty, P. R. K.. Table 4-8, 192 & Table 4-5, 186.
[92] Reeves, E., Spacecraft Design and Sizing, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 10-16, 276.
[93] Zermuehlen. R.O., Table 11-12, 323.
[94] Griffin, M. D.. 292.
[95] Fortesque, P. W., Dynamics of Spacecraft, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 43.
[96] Zermuehlen, R.O., 320.
[97] Griffin. M. D., 292.
[98] Gorney, D.J., Blake, J.B., Koons, H.C., Schultz, M., Vampola, A., Walterscheid, R.L., The Space Environment, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London. Kluwer Academic Publishers, 1991, 195.
[99] Zermuehlen. R.O., 321.
[100]Fortesque. P. W.. Dynamics of Spacecraft, 42.
[101] Griffin. M. D., 293.
[102] Chetty, P. R. K., Table 4-10, Table 4-9, 212.
[103] Ibid., Table 4-10, 214.
[104] Ibid., Appendix A, 513.
[105] Doukas, P.G.. 407.
[106] Chetty, P. R. K., Table 4-10, 214.
[107] Sackheim. R.L., 602.
[108] Fortesque. P. W., Dynamics of Spacecraft, 44.
[109] Chetty, P. R. K.. 206.
[110] Agrawal. B. N., 112.
[111] Fortesque, P. W., Attitude Control, 230.
[112] Sackheim. R.L.. Table 17-1, 580.
[113] Zermuehlen. R.O.. 320.
[114] Sackheim. R.L.. Table 17-8, 597.
[115] Ibid.. Table 17-2, 584 & Table 17-8, 597.
[116] Chetty, P. R. K.. Table 6-5, 294.
[117] Sackheim. R.L., Table 17-2, 584 & 592.
[118] Agrawal, B. N., 176.
[119] Reeves. E.. Table 10-23. 283.

[120] Agrawal, B. N., Table 6.5, 373.
[121] Reeves, E., Table 10-23, 283.
[122] McDermott, J.K., Power, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 357, 367 & Table 11-17, 368.
[123] Griffin, M. D., 419.
[124] McDermott, J.K., 358.
[125] Ibid., 357-358.
[126] Griffin, M. D., 420.
[127] Chetty, P. R. K., Table 3-17, 160.
[128] Agrawal, B. N., 332.
[129] Stark, J. P. W., Electrical Power Systems, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 263.
[130] Telecon with representative for AEC-Able Engineering, Data for their PUMA rigid array, August 7, 1995.
[131] McDermott, J.K., Table 11-35, 362 & 364.

6. DEMONSTRATING THE MODEL

The LabVIEW model developed for this research was used for demonstrating how a total systems analysis approach can benefit the analyses required in the early conceptual design stages, particularly for system and subsystem sizing and configuring. The demonstration seeks to answer some of the central questions that a design team of a remote sensing system focuses on during those stages of the design process. Each example in this demonstration is backed by a discussion, a chart, a table, and a reference to the relevant sections in Appendix A: EQUATIONS. Central sections are underlined. The data and figures in this chapter were generated from the developed model and the spreadsheet files produced by it, and provide examples of the kind of output that this model can generate.

To perform the analyses required for these demonstrations, a mission scenario was selected, and initial values for the main input variables were set. These were set to describe a three-axis stabilized, 200 kg, Earth observation system in a 403 km orbit, performing high resolution (8m) imaging in the visual spectrum. The sensor was set to staring mode, sensor data was not processed on-board, data was stored on a tape storage device, communication wavelength was set at $1.35*10^{-1}$ m, and the power system utilized GaAs solar cells and NiH2ipv type of batteries. A complete list of these initial (default) variable values and settings is given in Appendix D: DEFAULT VALUES FOR MODEL VARIABLES.

In Section 6.1, Major Model Interactions, the interactions between the variables: orbital period (T), orbital velocity (V), altitude (h), possible communication time (T_c), eclipse time (T_{ec}^{max}), and sensor wavelength (λ), and between these orbital variables and some of the spacecraft subsystems are discussed. The sensitivities of these variables to changes in Q value and altitude are included in the discussion. The type of analyses shown in this section should help the designer better understand the dynamics between central variables in the model.

Selecting the right technologies for a particular set of mission requirements is another issue that should be dealt with in the early conceptual design stages. Analyses for selecting the right technologies might indicate the need for new technology developments. An example illustrating the trade-offs between GaAs and Si solar cells is shown in Section 6.2, Selecting the Right Technology.

The last section, Section 6.3, Inter-Subsystem Trade-Offs, illustrates how a total systems model makes it possible to do sophisticated **inter-subsystem trade-off analyses**. The trade-offs between on-board processing vs. no on-board processing of sensor data was used to illustrate this issue. This trade-off was selected mainly based on comments from George Ganoe at NASA LaRC suggesting that this type of complete analysis had not been performed there.[1] In current Earth observation systems, on-board processing is utilized only to a limited extent. However, with increasing sensor data rates, for example through higher resolution imaging, on-board processing is becoming more important. The need for the development of smaller and more efficient processor technologies has been emphasized by JPL in their NEW MILLENNIUM Program.[2] The decision about whether to use on-board processing is a **configuration** issue. Section 6.3 also illustrates how the model can be used for detecting technology limits and bottlenecks, and for **sizing** the various subsystems for given mission requirements and spacecraft configurations.

As the model includes 300 variables, of which more than 100 are independent input variables, these analyses represent only a small fraction of the analyses possible with the model.

In setting the initial values for this demonstration, some design trade-offs had to be made. For example, effective pixel size has a minimum value for any given altitude. For values lower than this minimum, the signal-to-noise ratio equation for photon noise limited systems returns a negative square root value, giving a N.A. (Not Available) answer. The signal-to-noise ratio is used in the calculations of aperture diameter, and consequently in the calculation of sensor mass. The model was run hundreds of times to create consistent initial values.

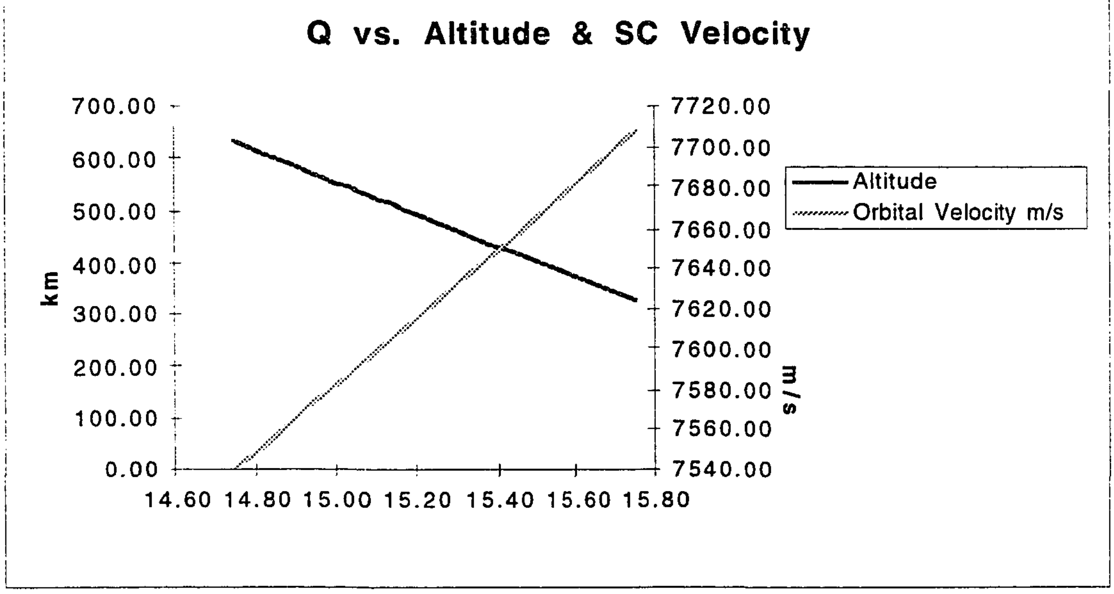
The initial values were used as default values in the model, in the sense that any variable not being tested or changed would be shown with its default value. A variable that had been tested would be returned to its default value after the test.

6.1. Major Model Interactions

The spacecraft defined in the model orbits around Earth in a circular orbit. A change in Q value, number of orbits per day, therefore, has an impact on all the orbital variables. The Q values in this demonstration range from 14.75 to 15.75 with the default value set to 15.5. For Q values higher than 16.25, corresponding to an altitude of 193 km, atmospheric friction would burn up the spacecraft.

An increase in Q over this range leads to an increase in orbital velocity (V) from 7541 m/s to 7708 m/s: a decrease in spacecraft altitude (h) from 631 km to 331 km (Figure 6-1), a decrease in the orbital period (T) from 97.3 min to 91.20 min (Figure 6-2), a decrease in possible communication time (T_c) from 7.9 min to 5.1 min (Figure 6-3), and an increase in eclipse time (T_{ec}^{max}) from 35.4 min to 36.4 min (Figure 6-4). These changes, though not linear, represent a 2.2 % increase in V, a **47.6 % decrease in h;** a 6.3 % decrease in T, a **35.4 % decrease in T_c**, and a 2.8 % increase in T_{ec}^{max}. From this preliminary analysis, it can be derived that a change in the Q value is likely to have a larger impact on the communication system[3] and the data storage system,[4] than on the power system.[5]

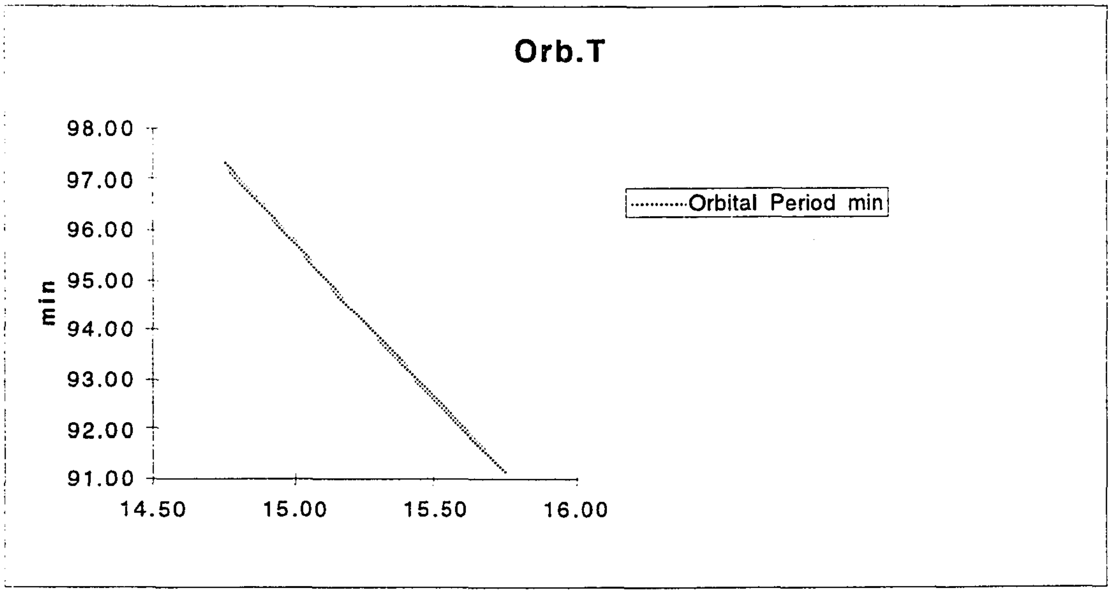
In the remainder of this section, altitude (h) derived from the Q value will be used instead of the Q value itself. The values of h range from 331 km to 631 km.



Q Value O/day	Altitude, km	Orbital Velocity m/s
14.75	631.38	7540.93
15.00	553.28	7583.29
15.10	522.65	7600.11
15.25	477.32	7625.19
15.50	403.41	7666.63
15.75	331.45	7707.63

The equations used are given in ***Appendix A: EQUATIONS****, Section 1.1.*

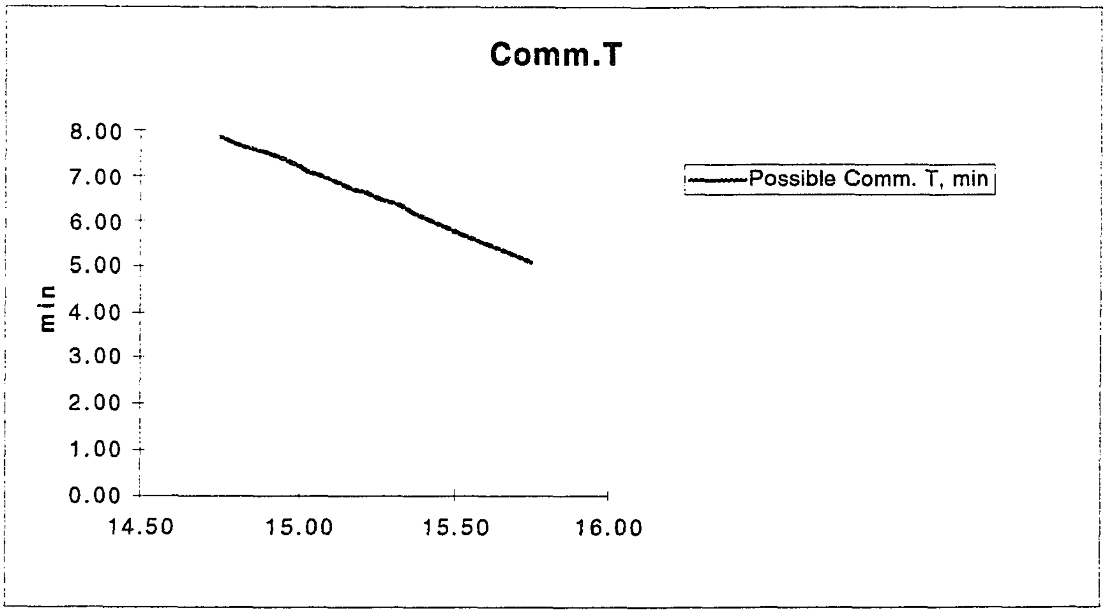
Figure 6-1 Q Value vs. Altitude and Spacecraft Velocity



Q Value O/day	Orbital Period min
14.75	97.30
15	95.70
15.1	95.10
15.25	94.10
15.5	92.60
15.75	91.20

The equations used are given in ***Appendix A: EQUATIONS****, Section 1.1.*

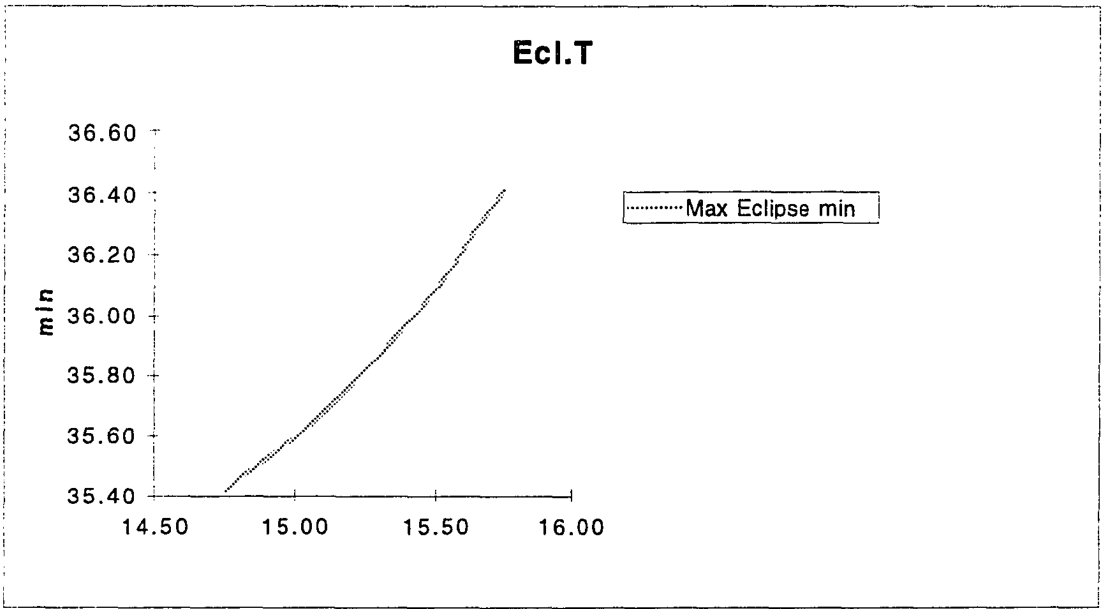
Figure 6-2 Q Value vs. Orbital Time



Q Value O/day	Possible Comm. T, min
14.75	7.90
15.00	7.21
15.10	6.93
15.25	6.52
15.50	5.81
15.75	5.09

The equations used are given in ***Appendix A: EQUATIONS****, Section 1.4.*

Figure 6-3 Q Value vs. Communication Time

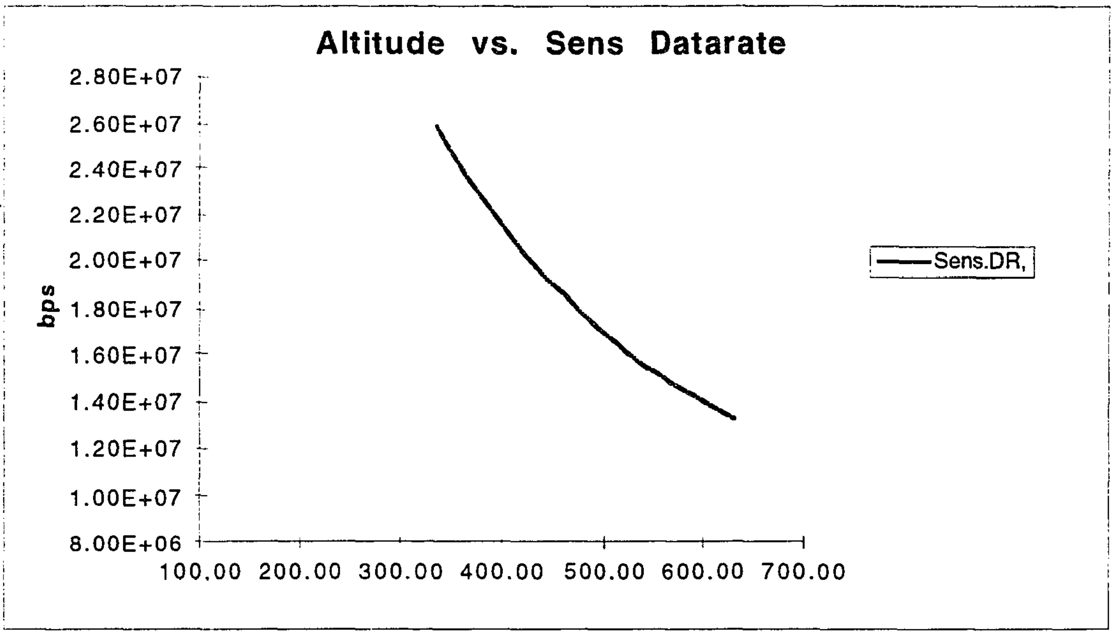


Q Value O/day	Max Eclipse min
14.75	35.40
15.00	35.60
15.10	35.70
15.25	35.80
15.50	36.10
15.75	36.40

The equations used are given in ***Appendix A: EQUATIONS****, Section 1.5.*

Figure 6-4 Q Value vs. Eclipse Time

For the sensor, the sensor data rate (DR_s) decreases with altitude because DR_s decreases with integration time (T_i) which increases with altitude. Over the altitude range, DR_s decreases from $2.59*10^7$ bps to $1.33*10^7$ bps (Figure 6-5) which represents a 48.6 % decrease. This percentage decrease is slightly higher than that of h, because T_i also increases with decreasing values of V, and V decreases with altitude.



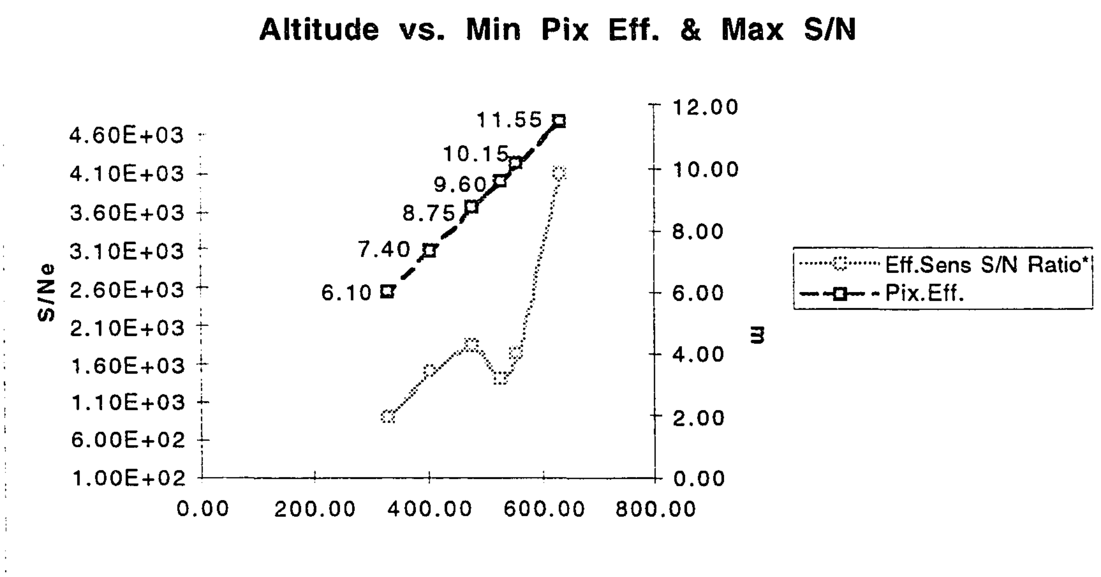
Altitude km	Sens.DR, bps
631.38	1.33E+07
553.28	1.53E+07
522.65	1.62E+07
477.32	1.78E+07
403.41	2.12E+07
331.45	2.59E+07

*The equations used are given in **Appendix A: EQUATIONS**, Sections 2.2.1. and 2.2.4.*

Figure 6-5 Altitude vs. Sensor Data Rate

Effective sensor signal-to-noise ratio (S/N_s^e) for a 12 m effective pixel size (d_{eff}) increases with altitude over the defined altitude range from 7.71 to $3.32*10^2$ (Figure 6-6, Figure 6-7, Figure 6-8). For d_{eff}=24 m, these values would be approximately halved. Figure 6-8 shows how sensor aperture diameter (D) increases from $1.95*10^{-2}$ to $1.08*10^{-1}$ m within the same altitude range. This indicates that sensor optics mass (M_o) increases with altitude.

The relationship between d_{eff} and S/N_s^e for a spacecraft sensor in a 403.3 km orbit is illustrated in Figure 6-7. The figure shows that S/N_s^e decreases from $1.51*10^3$ to 1.61 as d_{eff} increases from 7.4 m to 30 m. d_{eff}=7.4 m represents the smallest effective pixel size for

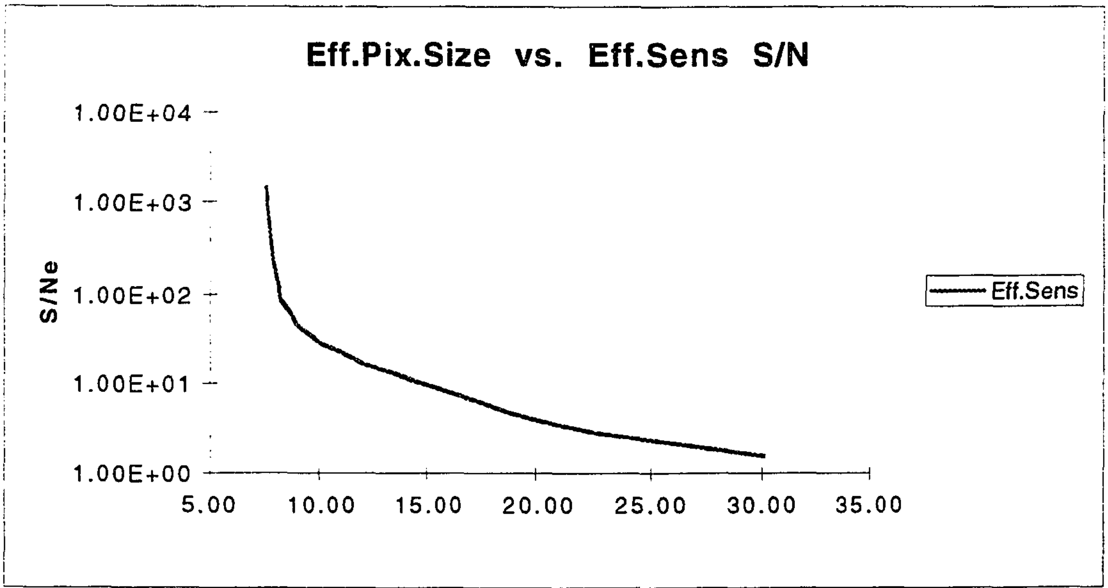


Altitude km	Pix.Eff.m	Eff.Sens S/N Ratio*
631.38	11.55	4.12E+03
553.28	10.15	1.75E+03
522.65	9.60	1.43E+03
477.32	8.75	1.85E+03
403.41	7.40	1.51E+03
331.45	6.10	9.12E+02

*The equations used are given in **Appendix A: EQUATIONS**, Section 2.2.1.*

Figure 6-6 Altitude vs. Minimum Effective Pixel Size and Maximum Effective Sensor S/N

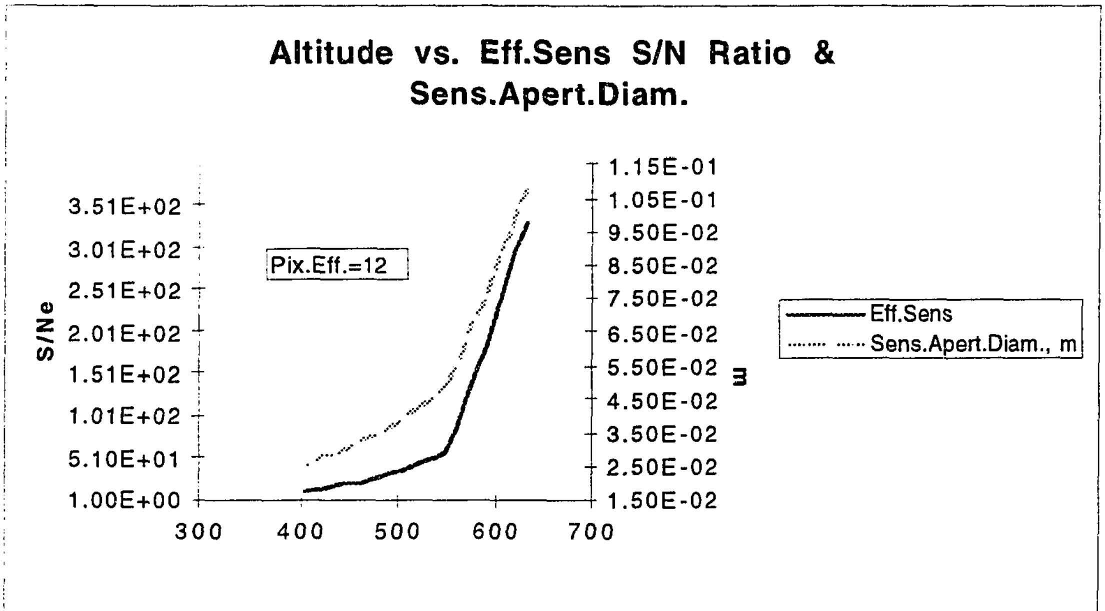
a sensor in the defined orbit. d_{eff} is defined as the RMS (root mean square) of the geometrically limited ground pixel (d_{pix}), the diffraction limited ground pixel (d_λ), and $V*T_i$. Solving this equation for d_λ gives negative and non-existent roots for d_{eff} values lower than these calculated minimum values. Minimum d_{eff} values for each given altitude are shown in Figure 6-6. The corresponding maximum values of S/N_s^e, as defined by the S/N_s^e-equation, are also shown in the figure. d_{eff} can only be reduced beyond its minimum for a given altitude by decreasing sensor detector diameter (d_d) or by increasing sensor focal length (f). The minimum d_{eff} values were used to determine the minimum d_{eff} that could be used for calculating the values in Figure 6-8. d_{eff} was set to 12 m, a value that gives positive square root values for all the selected altitudes.



Pix.Eff., m	Eff.Sens, S/Ne
7.40	1.51E+03
8.00	1.31E+02
10.00	2.93E+01
20.00	3.93E+00
30.00	1.61E+00

The equations used are given in ***Appendix A: EQUATIONS****, Section 2.2.1.*

Figure 6-7 Effective Pixel Size and Effective Sensor S/N

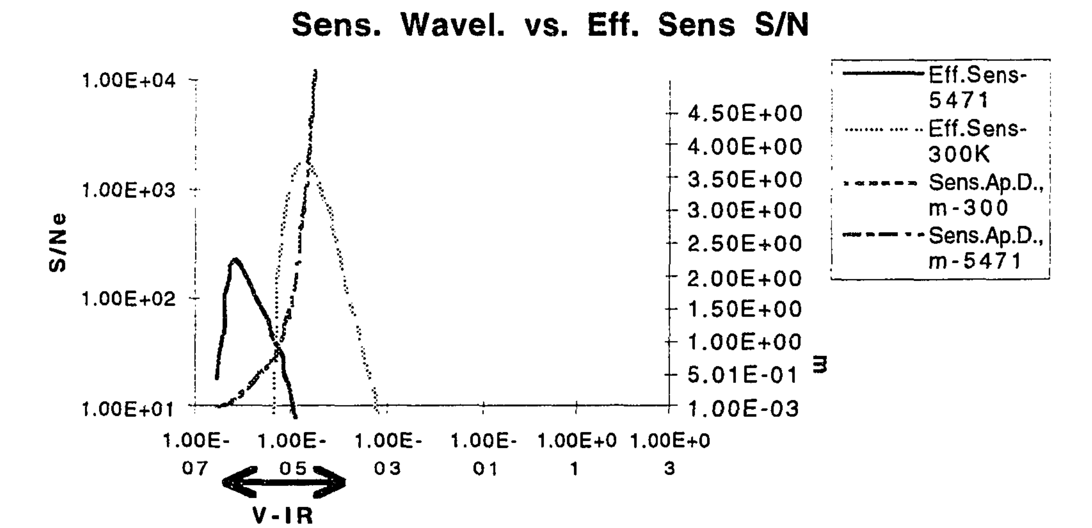


Altitude km	Eff.Sens.S/Ne	Sens.Apert.Diam., m
6.31E+02	3.32E+02	1.08E-01
5.53E+02	6.98E+01	5.13E-02
5.23E+02	4.83E+01	4.33E-02
4.77E+02	3.01E+01	3.50E-02
4.03E+02	1.51E+01	2.59E-02
3.31E+02	7.71E+00	1.95E-02

*The equations used are given in **Appendix A: EQUATIONS**, Section 2.2.1.*

Figure 6-8 Altitude vs. Effective Sensor S/N and Aperture Diameter

The sensor design parameters are also impacted by the observation wavelength (λ) and the black body temperature (t). Changes in these environmental parameters result in changes in S/N_s^e and the sensor aperture diameter (D). The changes are shown in Figure 6-9. In this figure, the signal is measured as electro-magnetic flux (F_{em}) as defined by Planck's black body equation. The figure shows that electro-magnetic radiation at t=5471 K produces a sufficient S/N_s^e for wavelengths (λ) in the visual spectrum, ranging from $3.00*10^{-7}$ m to $7.50*10^{-7}$ m. At these wavelengths, electromagnetic radiation at t=300 K does not produce a sufficient S/N_s^e. For wavelengths (λ) in the infrared spectrum from $7.50*10^{-7}$ m to $1.00*10^{-4}$ m , the S/N_s^e for t=300 K is sufficient and higher than that for t=5471K. For observations at wavelengths exceeding $1.00*10^{-4}$ m, other types of sensors



Sensor Wavel., m	Eff.Sens-5471	Eff.Sens-300K	Sens.Ap.D.,m-5471	Sens.Ap.D.,m-300
3.00E-07	1.92E+01	1.00E-50	4.71E-02	4.76E-02
5.00E-07	1.31E+02	2.82E-33	7.62E-02	7.93E-02
7.50E-07	2.26E+02	6.36E-20	1.14E-01	1.19E-01
5.00E-06	3.89E+01	8.49E+01	7.92E-01	7.90E-01
1.00E-05	1.12E+01	1.25E+03	1.58E+00	1.56E+00
2.00E-05	3.00E+00	1.89E+03	3.17E+00	3.12E+00
5.00E-05	5.00E-01	7.71E+02	7.93E+00	7.91E+00
9.00E-05	1.56E-01	3.03E+02	1.43E+01	1.43E+01
1.00E-04	1.27E-01	2.53E+02	1.59E+01	1.58E+01
5.00E-04	5.12E-03	1.24E+01	7.93E+01	7.93E+01
9.00E-04	1.58E-03	3.90E+00	1.43E+02	1.43E+02
5.00E-03	5.13E-05	1.29E-01	7.93E+02	7.93E+02
5.00E-02	5.13E-07	1.30E-03	7.93E+03	7.93E+03
5.00E-01	5.13E-09	1.30E-05	7.93E+04	7.93E+04
5.00E+00	5.13E-11	1.30E-07	7.93E+05	7.93E+05
5.00E+01	5.13E-13	1.30E-09	7.93E+06	7.93E+06
5.00E+02	5.13E-15	1.30E-11	7.93E+07	7.93E+07

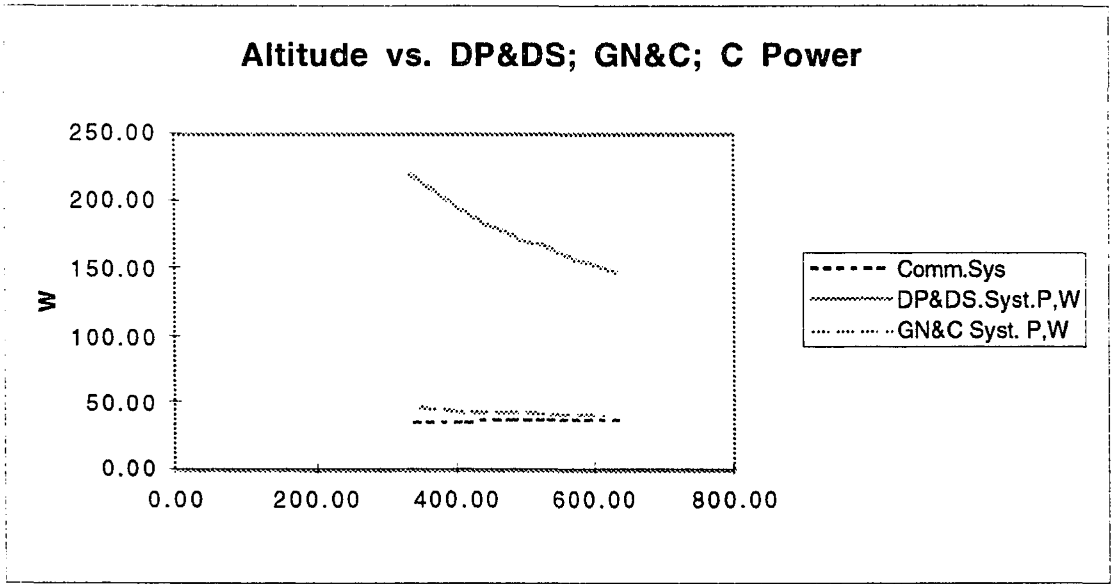
The equations used are given in ***Appendix A: EQUATIONS****, Section 2.2.1.*

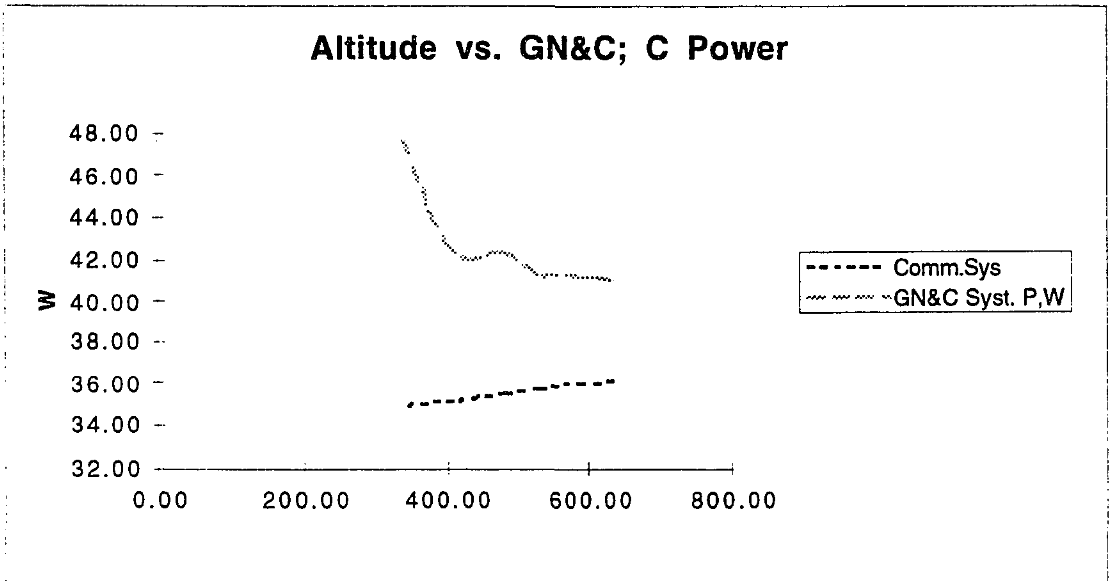
Figure 6-9 Sensor Wavelength vs. Effective Sensor S/N

such as radiometers need to be employed. A further discussion of these would fall outside the scope of this dissertation. The figure also shows that aperture diameter (D) and consequently sensor optics mass (M_o) increases with λ. In the visual spectrum, D increases from $4.71*10^{-2}$ m to $1.14*10^{-1}$ m, and in the infrared spectrum D increases from

$1.19*10^{-1}$ m to $1.58*10^{1}$ m. Already at λ=$2.00*10^{-5}$ m, D equals 3.12 m, which is prohibitively large for smaller Earth Observation Satellites in LEO, again indicating that other types of sensors should be preferred for these higher wavelengths (λ).

Figure 6-10 shows how the power required by the communication system (P_{COM}), the data processing and data storage system ($P_{P\&S}$), and the GN&C system (P_{GNC}) change with h..





Altitude km	Comm.Sys.P,W	DP&DS.Syst.P,W	GN&C Syst. P,W
631.38	36.26	147.98	41.09
553.28	35.97	160.63	41.31
522.65	35.84	166.40	41.31
477.32	35.65	176.01	42.44
403.41	35.32	195.27	42.45
331.45	34.99	220.32	47.85

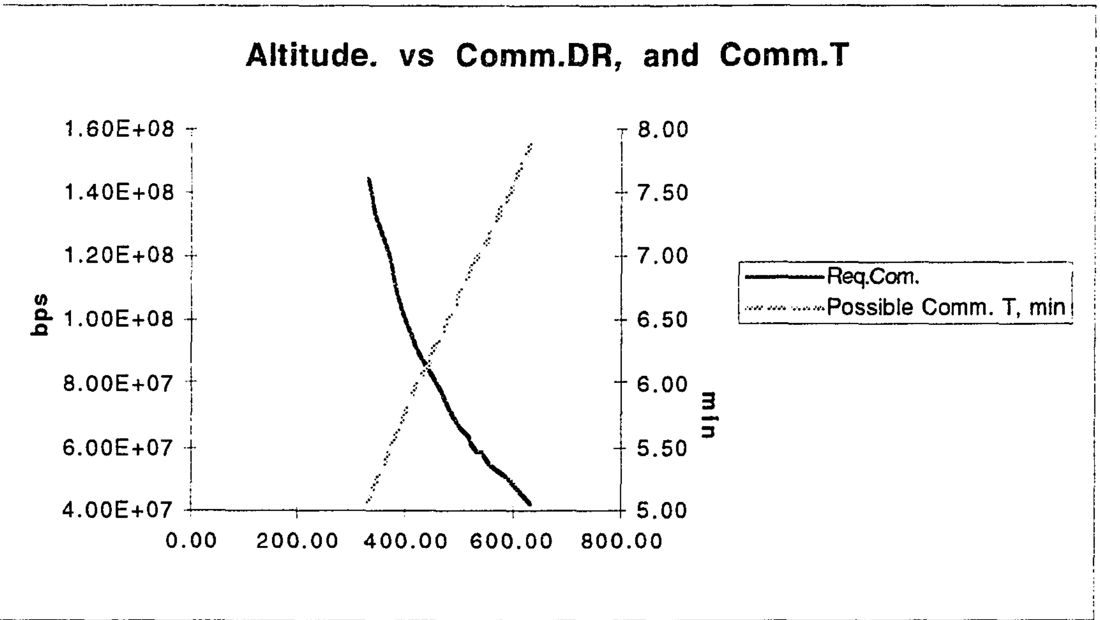
The equations used are given in ***Appendix A: EQUATIONS****, Sections 2.2.1., 2.2.3., 2.2.4., 2.3.1., 2.3.2., 2.4.1, 2.4.2., 2.5.2., and 2.5.3.* (Main sections are underlined.)

Figure 6-10 Altitude vs. Communication, Data Processing & Data Storage, and Guidance Navigation & Control System Power Requirements

In Figure 6-5, it was shown that DR_s decreases with h. The total number of instructions per second (N_{TIPS}) required for processing of housekeeping (HK) and sensor data, and consequently of CPU power (P_{CPU}) also decreases with h. So does the data storage requirement (DS) and the resulting power required by the data storage system (P_{DS}). The decrease in DS and P_{DS} come as a result of orbital time (T) and communication time (T_c) increasing with h. The combined effect is a decrease in power required by the data processing and data storage systems ($P_{P\&S}$) from 220.32 W to 147.98 W, a 32.8% decrease over the given altitude range. This significant decrease in $P_{P\&S}$ is a result, mainly, of the 35.4% increase in communication time (T_c).

Explaining the increase in power required by the communication system (P_{COM}) from 34.99 W to 36.26 W requires an expanded analysis of the variables in the P_{COM} equation.

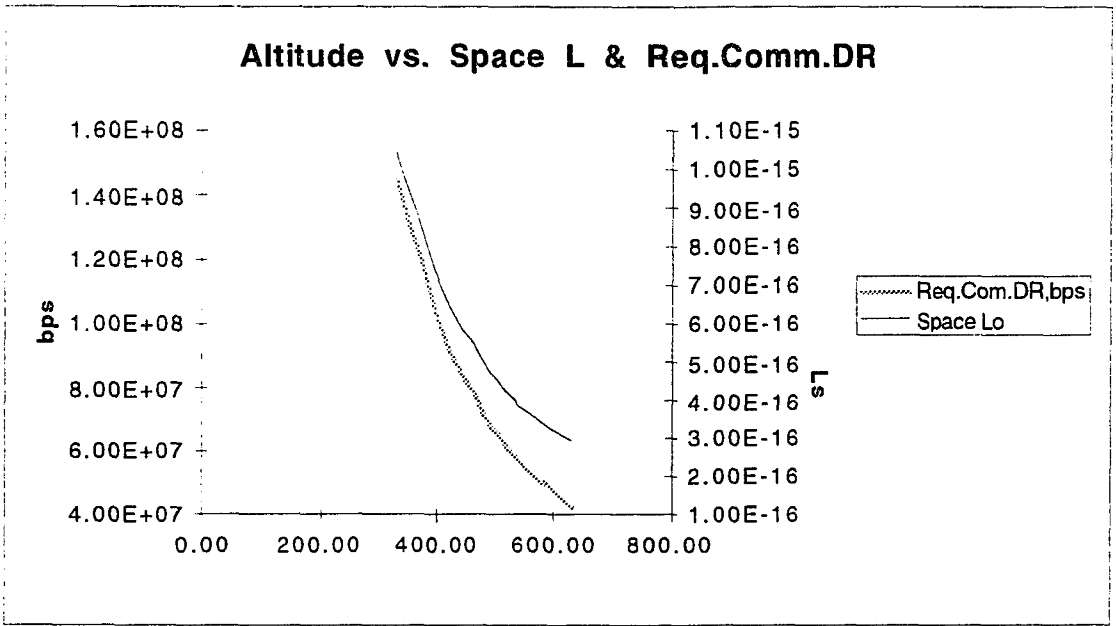
Figure 6-11 shows that the required communication dump-rate (DR_c) decreases by 70.4% with h as communication time (T_c), increases, and Figure 6-12 indicates that space loss (L_s), which is inversely proportional to P_{COM}, decreases by 72.4% with h. The combined effect is indicated by the increase in P_{COM}. Figure 6-13 gives a close-up of the relationship between T_c, h, and P_{COM}.



Altitude km	Req.Com.bps	Possible Comm. T, min
631.00	4.26E+07	7.90
553.00	5.47E+07	7.21
523.00	6.09E+07	6.93
477.00	7.22E+07	6.52
403.00	9.93E+07	5.81
331.00	1.44E+08	5.09

*The equations used are given in **Appendix A: EQUATIONS**, Sections 1.4., 2.2.1., 2.2.3., 2.2.4., 2.3.1., 2.3.2., and 2.4.1.* (Main sections are underlined.)

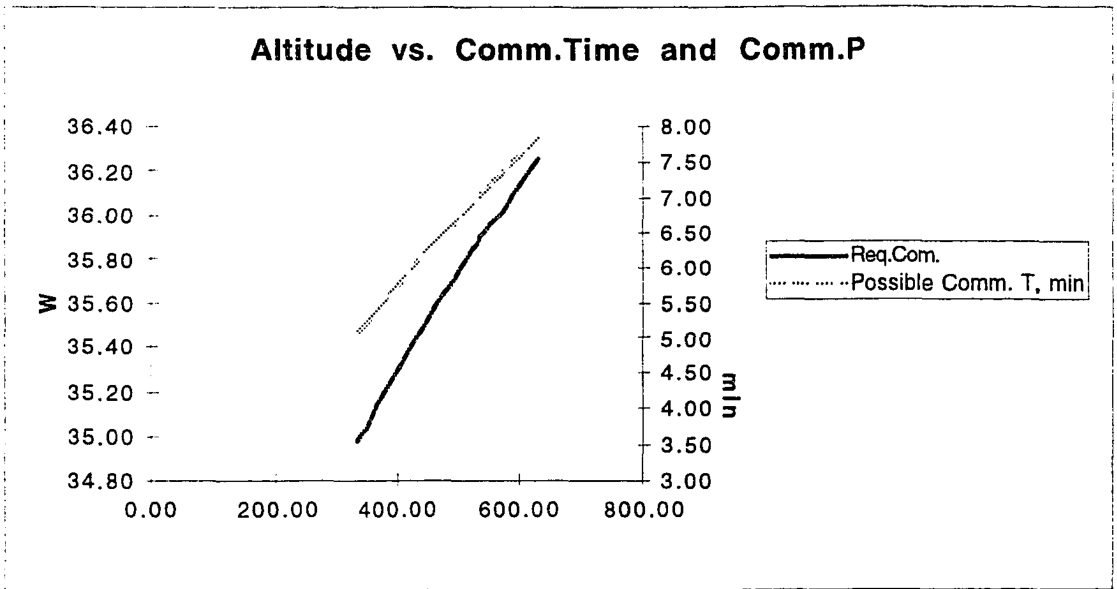
Figure 6-11 Altitude vs. Communication Data Rate and Communication Time



Altitude km	Space Lo, Ls	Req.Com.DR,bps
631.38	2.90E-16	4.26E+07
553.28	3.77E-16	5.47E+07
522.65	4.23E-16	6.09E+07
477.32	5.07E-16	7.22E+07
403.41	7.09E-16	9.93E+07
331.45	1.05E-15	1.44E+08

The equations used are given in ***Appendix A: EQUATIONS****, Sections 1.4., 2.2.1., 2.2.3., 2.2.4., 2.3.1., 2.3.2., 2.4.1., and 2.4.2.* (Main sections are underlined.)

Figure 6-12 Altitude vs. Space Loss and Communication Data Rate

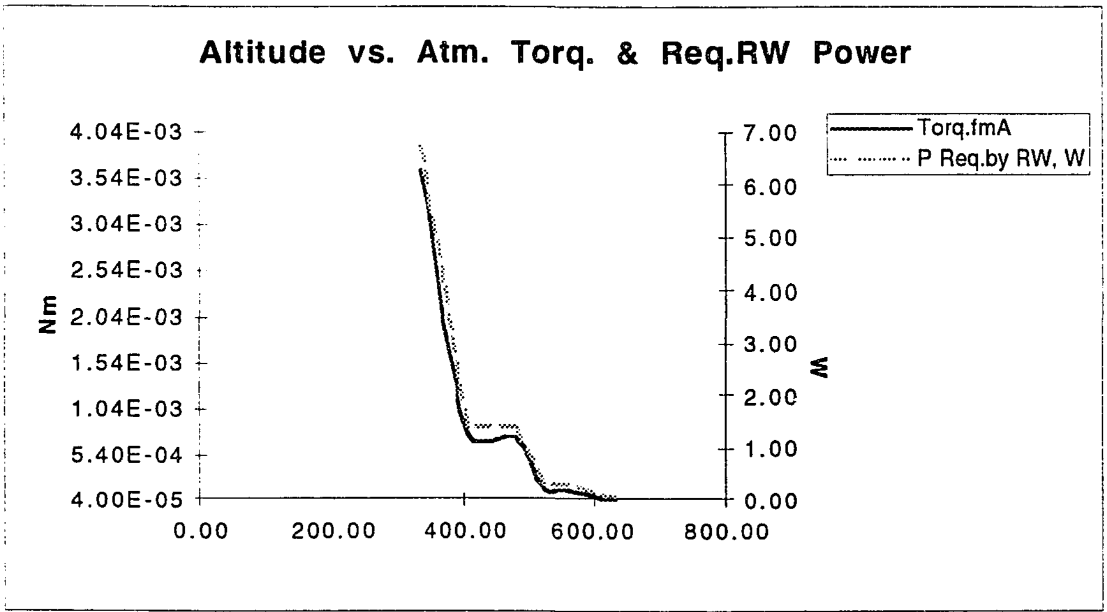


Altitude km	Req.Com.,W	Possible Comm. T, min
631.38	36.26	7.90
553.28	35.97	7.21
522.65	35.84	6.93
477.32	35.65	6.52
403.41	35.32	5.81
331.45	34.99	5.09

The equations used are given in ***Appendix A: EQUATIONS****, Sections* *1.4., 2.2.1., 2.2.3., 2.2.4., 2.3.1., 2.3.2., 2.4.1., and 2.4.2. (Main sections are underlined.)*

Figure 6-13 Altitude vs. Communication Time and Communication Power

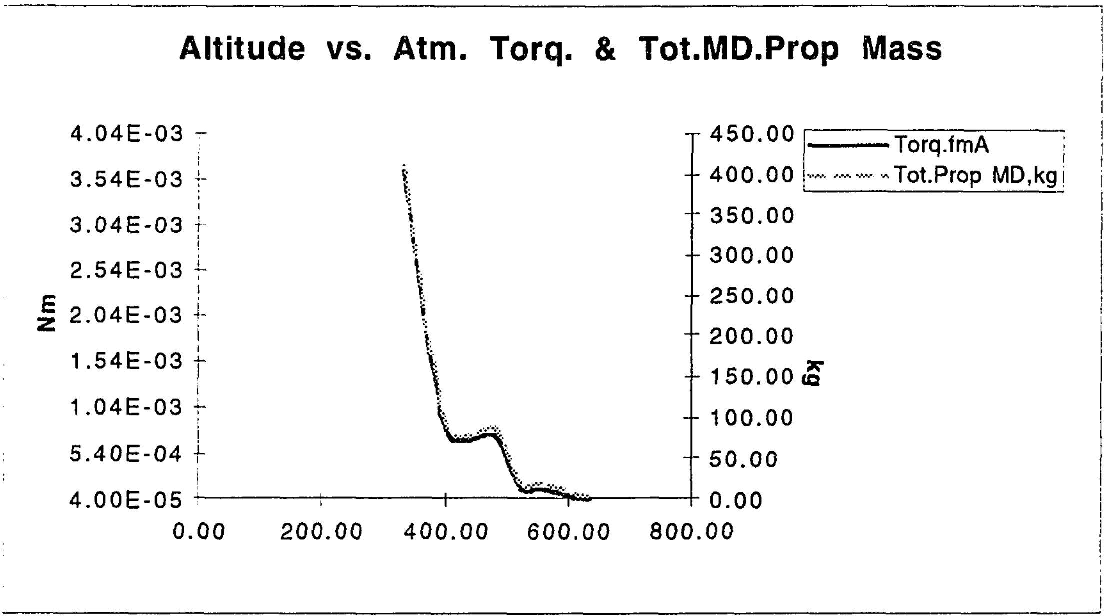
Power for the GN&C system (P_{GNC}) decreases with h from 47.85 W to 41.09 W as atmospheric density (ρ) and atmospheric torque (τ_{AD}) and consequently power required by the reaction wheel (P_{rw}) decreases with h. Assuming that chemical propulsion is used for the momentum dumping thrusters, the change in P_{GNC} equals the change in the power required by the reaction wheel (P_{rw}). This is confirmed in Figure 6-14 which shows a 6.76 W decrease in P_{rw} over the given altitude interval. Also, Figure 6-14 and Figure 6-15 show the direct and proportional relationship between τ_{AD}, and P_{rw} and total propellant required for momentum dumping (M_{pTmd}). The decrease in τ_{AD} equals $3.58*10^{-3}$ Nm and the decrease in M_{pTmd} equals 409.47 kg, giving the same 98.6% decrease for τ_{AD}, P_{rw}, and M_{pTmd}.



Altitude km	Torq.fmA,Nm	P Req.by RW, W
631.38	4.88E-05	9.20E-02
553.28	1.66E-04	3.12E-01
522.65	1.66E-04	3.14E-01
477.32	7.63E-04	1.44E+00
403.41	7.71E-04	1.45E+00
331.45	3.63E-03	6.85E+00

The equations used are given in ***Appendix** A:* ***EQUATIONS**, Sections 2.5.1. and 2.5.2.*

Figure 6-14 Altitude vs. Torque from Atmospheric Friction and Reaction Wheel Power

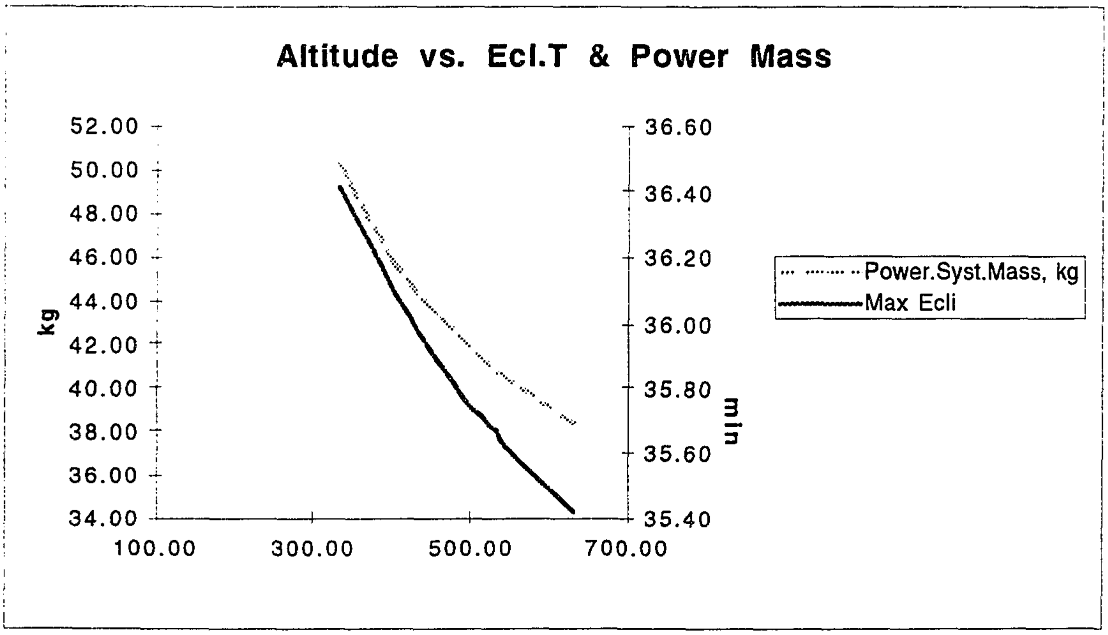


Altitude km	Torq.fmA,Nm	Tot.Prop MD,kg
631.38	4.88E-05	5.58
553.28	1.66E-04	18.92
522.65	1.66E-04	19.00
477.32	7.63E-04	87.09
403.41	7.71E-04	88.04
331.45	3.63E-03	415.05

The equations used are given in ***Appendix A: EQUATIONS****, Sections 2.5.1., 2.5.2, and* *2.5.3.* (Main sections are underlined.)

Figure 6-15 Altitude vs. Torque from Atmospheric Friction and Propellant for Momentum Dumping

The net decrease in power system mass (M_{POW}) as a result of a net decrease in the total satellite power requirements is shown in Figure 6-16. The figure shows a 23.7% decrease in M_{POW} from 50.29 kg to 38.39 kg over the given altitude interval. This decrease seems to be driven by the total satellite power requirements rather than the eclipse time (T_{ec}^{max}). Figure 6-4 shows that T_{ec}^{max} only increases by 2.8%, as Figure 6-10 shows a net decrease of 25.7% in the power requirements by the data processing and data storage system, the communication system, and the GN&C systems.



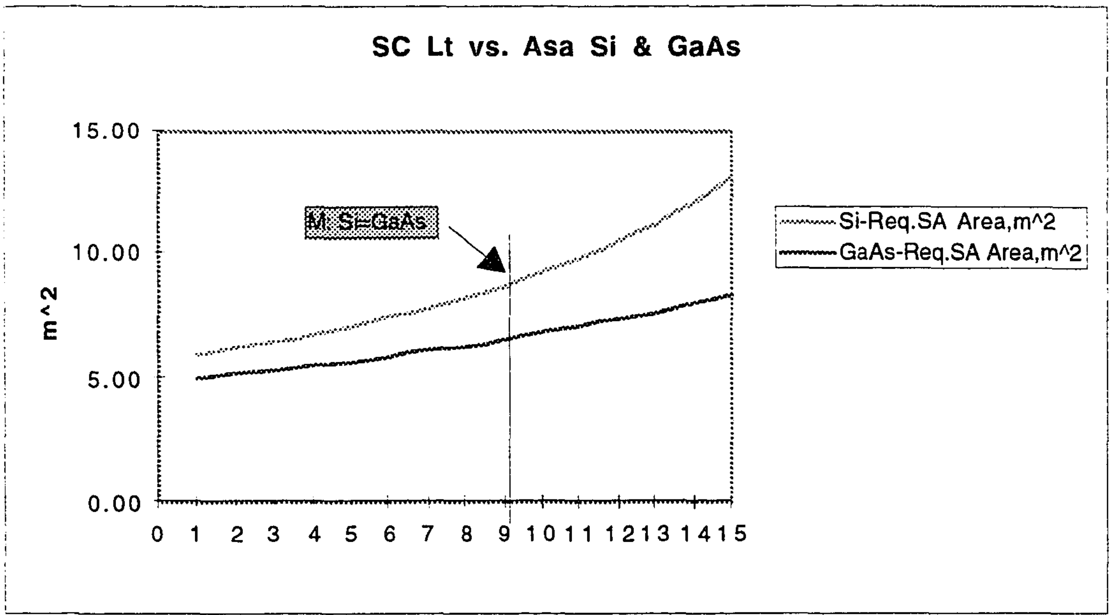
Altitude km	Max Ecli,min	Power.Syst.Mass, kg
631.38	35.42	38.39
553.28	35.60	40.32
522.65	35.69	41.19
477.32	35.83	42.79
403.41	36.09	45.72
331.45	36.42	50.29

The equations used are given in ***Appendix A: EQUATIONS****, Sections 1.5., 2.6.1, and 2.6.2.*

Figure 6-16 Altitude vs. Eclipse Time and Power Systems Mass

6.2. Selecting the Right Technology

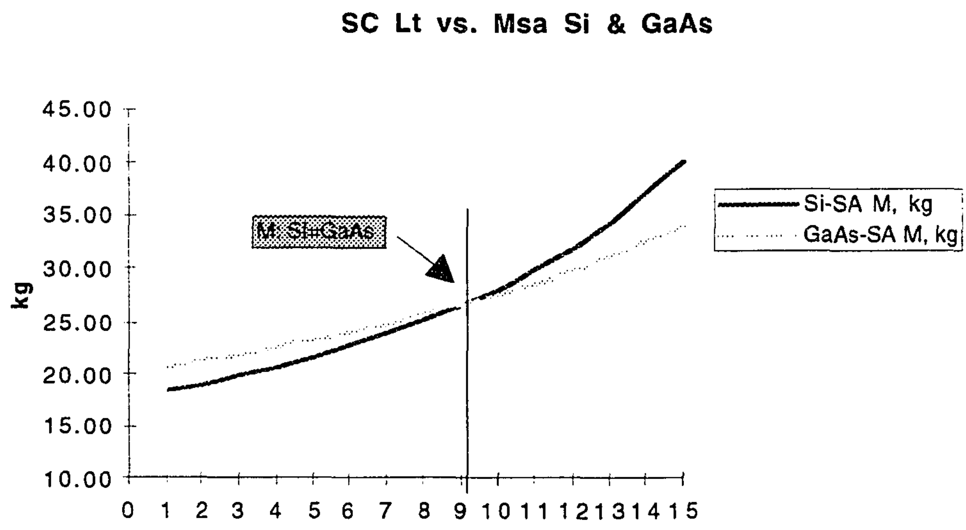
A preliminary analysis of GaAs cells versus Si cells based on solar array area indicates, as shown in Figure 6-17, that GaAs cells should be preferred over Si cells for any mission length. GaAs cells have a higher efficiency (e_s) and lower annual degradation ($\Delta e_{s/y}$) than Si cells resulting in a lower required solar array area (A_{sa}). However, as mass per unit area (χ_A) for GaAs cells is higher than that of Si cells, solar array mass (M_{SA}) as a function of solar array area (A_{sa}) should be used for drawing the final conclusions. In the context of this demonstration, Figure 6-18 indicates that Si cells should be preferred for missions with lifetimes (T_{lt}) lower than about 9.1 years. For longer missions, the higher annual solar cell degradation ($\Delta e_{s/y}$) for Si cells leads to M_{SA} for GaAs being lower than M_{SA} for Si. To make GaAs cells attractive for shorter missions, their mass per unit area (χ_A) has to be reduced.



SC Lifetime, y	Si-Req.SA Area,m^2	GaAs-Req.SA Area,m^2
1.00	5.97	5.07
2.00	6.21	5.21
3.00	6.48	5.37
4.00	6.76	5.54
5.00	7.07	5.71
6.00	7.42	5.90
7.00	7.79	6.10
8.00	8.21	6.32
9.00	8.68	6.55
10.00	9.20	6.80
11.00	9.78	7.06
12.00	10.50	7.35
13.00	11.20	7.67
14.00	12.10	8.01
15.00	13.10	8.39

The equations used are given in ***Appendix A: EQUATIONS****. Section 2.6.1.*

Figure 6-17 Spacecraft Lifetime vs. Solar Array Area using GaAs or Si Cells



SC Lifetime, y	Si-SA M, kg	GaAs-SA M, kg
1.00	18.40	20.70
2.00	19.10	21.30
3.00	19.90	22.00
4.00	20.80	22.70
5.00	21.80	23.40
6.00	22.80	24.10
7.00	24.00	25.00
8.00	25.30	25.90
9.00	26.70	26.80
10.00	28.30	27.80
11.00	30.10	28.90
12.00	32.20	30.10
13.00	34.50	31.40
14.00	37.30	32.80
15.00	40.50	34.30

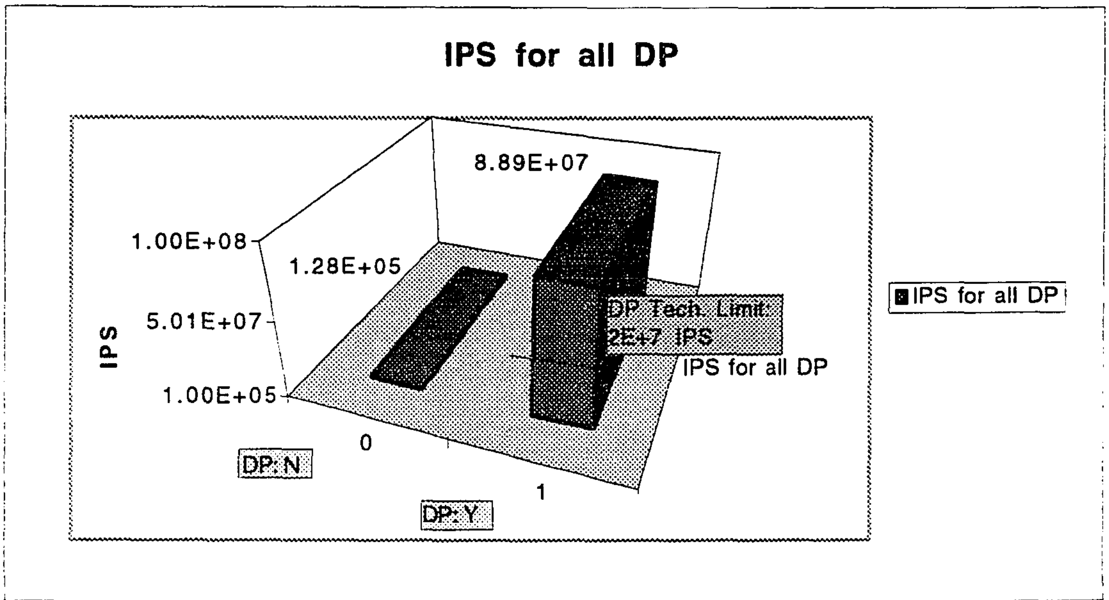
*The equations used are given in **Appendix A: EQUATIONS**. Section 2.6.1.*

Figure 6-18 Spacecraft Lifetime vs. Solar Array Mass using GaAs or Si Cells

6.3. Inter-Subsystem Trade-Offs

The model can be used for doing inter-subsystem analyses of the power and mass impacts (Figure 5-2) of on-board processing of the Earth observation sensor data. Such an analysis is discussed in this section. The analysis encompasses the data processing and data storage systems, the communication system, and the power system.

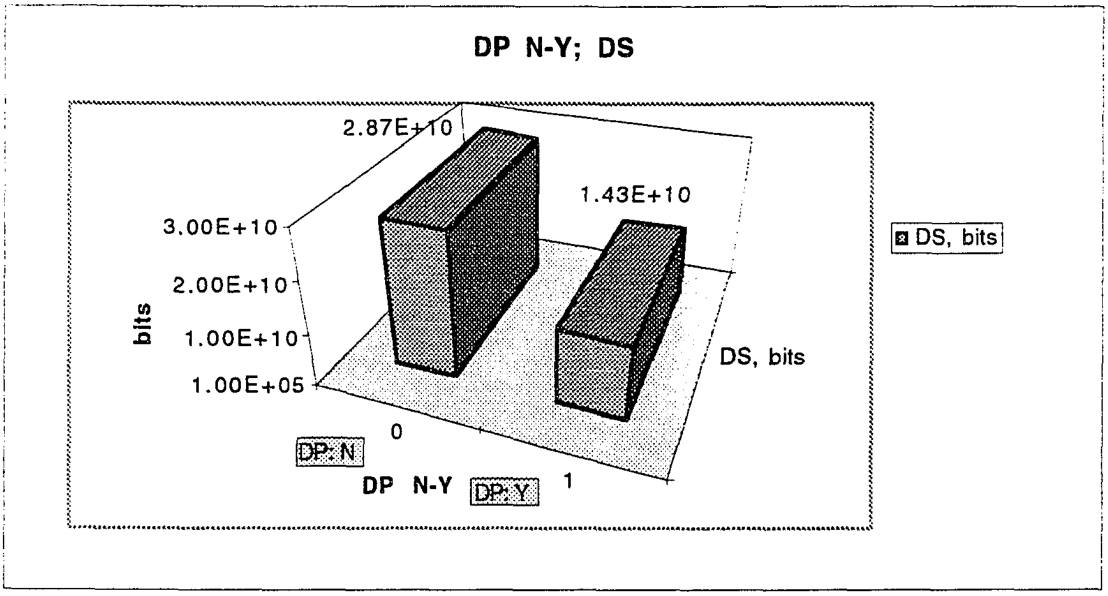
As shown in Figures 6.19, 6.20, and 6.21, going from the no-processing option to the processing option increases the required processing power (N_{TIPS}), from $1.28*10^5$ IPS to $8.89*10^7$ IPS; and decreases the required data storage capacity (DS), from $2.87*10^{+10}$ bits to $1.43*10^{10}$ bits, and the communication dump-rate (DR_c), from $9.93*10^7$ bps to $4.96*10^7$ bps. Processing power increases (N_{TIPS}) because the processor in the processing option has to process both housekeeping and Earth observation sensor data. The processing routine compresses the sensor data rate (DR_s) by 50% resulting in the halved values for DS and DR_c.



Sens.Proc.Y-N	IPS for all DP
0.00	1.28E+05
1.00	8.89E+07

The equations used are given in ***Appendix A: EQUATIONS****, Sections 2.2.1., 2.2.3., 2.2.4., and* *2.3.1.* (Main sections are underlined.)

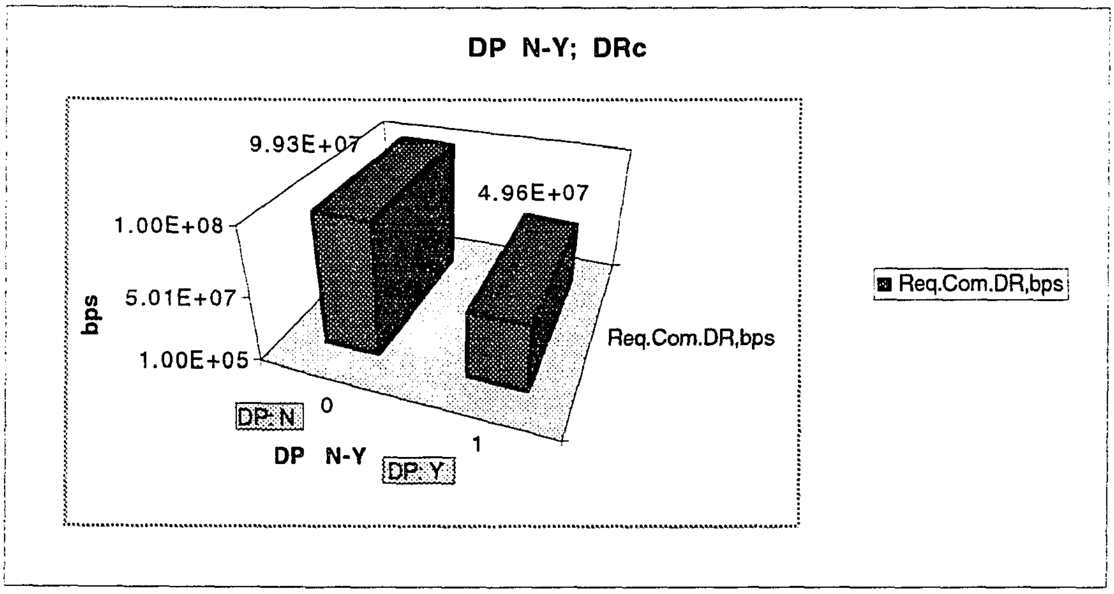
Figure 6-19 Impact of On-Board Processing on the Required Data Processing Power (IPS)



Sens.Proc.Y-N	DS, bits
0.00	2.87E+10
1.00	1.43E+10

The equations used are given in ***Appendix A: EQUATIONS****, Sections 2.2.1., 2.2.3, 2.2.4., 2.3.1., and* 2.3.2. (Main sections are underlined.)

Figure 6-20 Impact of On-Board Processing on the Required Data Storage Capacity (bits)

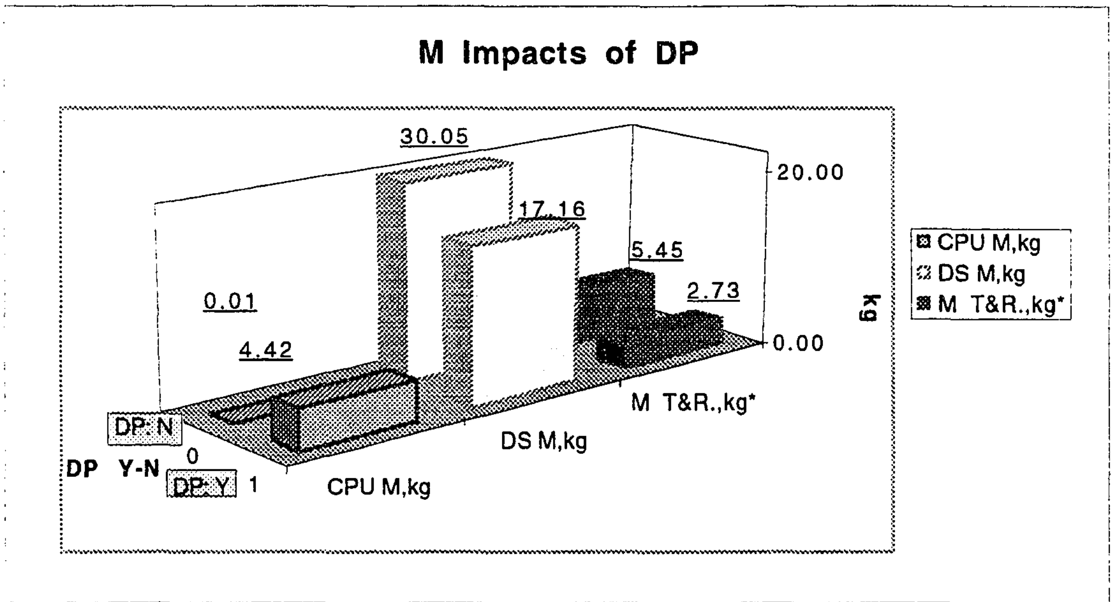


Sens.Proc.Y-N	Req.Com.DR,bps
0.00	9.93E+07
1.00	4.96E+07

The equations used are given in ***Appendix A: EQUATIONS****, Sections 1.4., 2.2.1., 2.2.3., 2.2.4., 2.3.1., 2.3.2., and* *2.4.1.* (Main sections are underlined.)

Figure 6-21 Impact of On-Board Processing on the Required Communication Data Rate (bps)

Figure 6-22 shows the mass impact of the processing option. The mass of the data processing system (M_{PCU}) increases by 4.41 kg, the mass of the data storage system (M_{DS}) decreases by 12.9 kg, and the mass of the communication receiver and transmitter unit decreases by 2.72 kg. The net decrease is 11.21 kg.

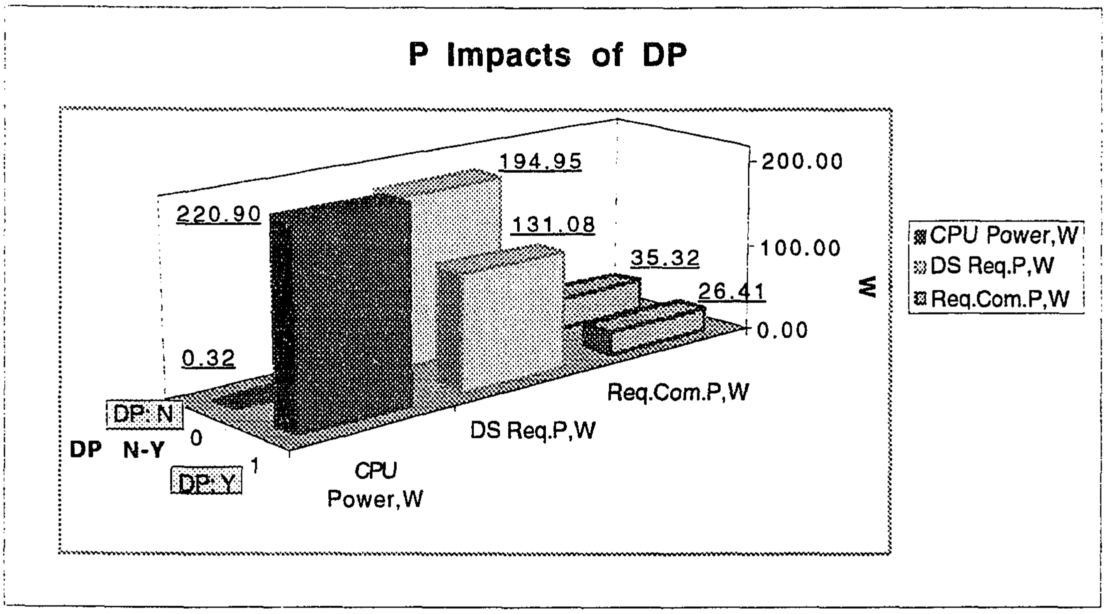


Sens.Proc.Y-N	CPU M,kg	DS M,kg	M T&R.,kg*
0.00	0.01	30.10	5.45
1.00	4.42	17.20	2.73
	4.41	-12.9	-2.72

The equations used are given in ***Appendix A: EQUATIONS****, Sections 1.4., 2.2.1., 2.2.4., 2.3.1., 2.3.2., 2.4.1., 2.4.2., and 2.4.3.* (Main sections are underlined.)

Figure 6-22 Impact of On-Board Processing on the Data Processing, the Data Storage, and the Communication System Masses (kg)

The change in power requirements shown in Figure 6-23 gives a different picture. Required power for the data processing system (P_{CPU}) increases by 220.58 W; required power for the data storage system (P_{DS}) decreases by 63.87 W, and required power for the communication system decreases by 8.91 W. The net result is an increase of 147.8 W.

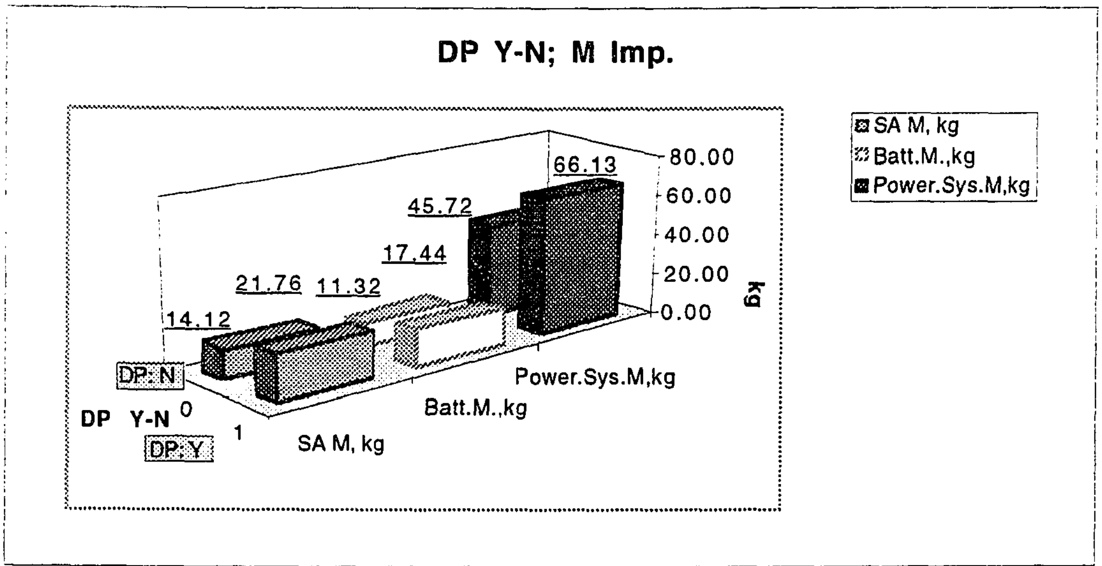


Sens.Proc.Y-N	CPU Power,W	DS Req.P,W	Req.Com.P,W
0.00	0.32	194.95	35.32
1.00	220.90	131.08	26.41
	220.58	-63.87	-8.91

The equations used are given in ***Appendix A: EQUATIONS****, Sections 1.4., 2.2.1., 2.2.4., 2.3.1., 2.3.2., 2.4.1., and 2.4.2. (Main sections are underlined.)*

Figure 6-23 Impact of On-Board Processing on the Data Processing, the Data Storage, and the Communication System Power Requirements (W)

Figure 6-24 shows the mass impacts on the power system of this increase in the power requirements. Solar array mass (M_{SA}) increases from 14.12 kg to 21.76 kg, battery mass (M_{BAT}) increases from 11.32 kg to 17.44 kg, and the total power system mass (M_{POW}), including the solar array and the battery, increases by 20.41 kg from 45.72 kg to 66.13 kg.



Sens.Proc.Y-N	SA M, kg	Batt.M.,kg	Power.Sy.kg
0.00	14.12	11.32	45.72
1.00	21.76	17.44	66.13
			20.41

The equations used are given in ***Appendix A: EQUATIONS****, Sections 1.5., 2.6.1., and 2.6.2.* (Main sections are underlined.)

Figure 6-24 Impact of On-Board Processing on the Power System Mass (kg)

Comparing this 20.41 kg increase in the total power system mass, with the 11.21 kg net savings in the data processing, data storage, and communication system masses, indicates that the no-processing option should be preferred. That option would mean a 9.2 kg reduction in the total spacecraft mass.

The on-board processing option becomes more attractive by employing compression algorithms with fewer number of instructions per sensor sample ($N_{i/s}^{s}$). The break-even point is at 35 IP/Sample. For values of $N_{i/s}^{s}$ lower than that, the subsystem mass decreases are larger than the increase in the total power system mass (M_{POW}) making the processing option more attractive. The break-even point of 35 IP/Sample, is below what is possible with current compression algorithms, indicating that research into making these algorithms more efficient would be beneficial.

Processing technology represents another potential area of research. Figure 6-19 shows that, for the on-board processing option, the required processing power (N_{TIPS}) exceeds the technology limit of current available processors. This holds true for all values of $N_{i/s}^{s}$ higher than 11.2 IP/Sample. Research on space qualified processors with higher capacities should therefore be considered. In the previous discussion, it was assumed that processor technology with sufficient processor power was available or could be made available.

From these demonstrations, some highlights can be extracted. In section 6.1, it was shown that an increase in Q from 14.75 to 15.75 led to a 47.6 % decrease in h; a 35.4 % decrease in T_c; a 48.6 % increase in DR_s; a 32.8% increase in $P_{P\&S}$; a 98.6% increase in τ_{AD}, P_{rw}, and M_{pTmd}; a 72.4% increase in L_s; and a 23.7% increase in M_{POW}, making these variables the most sensitive to changes in Q. Results from section 6.2 indicated that research for reducing the mass of GaAs solar cells should be considered. The results derived in section 6.3 show that research should be considered for making compression algorithms more efficient, and for developing space qualified processors with higher capacities.

---

[1] Conversations with George Ganoe, NASA Langley Research Center, Hampton, Virginia.
[2] Marino, K., The New Millennium Program, *JPL Universe*, February 10, 1995.
[3] 5.2.4. Communication, 51.
[4] 5.2.3.2. Data Storage, 50.
[5] 5.2.6. Power, 58.

7. FEEDBACK FROM POTENTIAL USERS

To get an initial idea about how the method and modeling approach suggested in this research was perceived, some general questions were directed to five of the people who were involved at different levels during the modeling process. One might categorize this as a nonrandom, purposeful sample. Four of the five work at the NASA Langley Research Center. The fifth has been working as a NASA contractor for a number of years. Three of the four NASA experts work in the same section and branch (Branch A). Their comments were, therefore, grouped together. Only one of them has seen an early version of the model run. The two others were providing subsystem information, and were exposed primarily to that part of the model dealing with their subsystem. The fourth NASA expert works in a separate branch (Branch B). He has not seen the model run, but he was given a detailed presentation of the method, the developed model, and its analysis capabilities. The external expert was involved in the overall model development and has seen the model run a number of times. All five read and had available a draft of this dissertation as reference for their comments. To get a true picture of how a design method such as this one will be perceived, it needs to be applied to a real project. Such an application is suggested for future research in Chapter 8, CONCLUSIONS AND RECOMMENDATIONS. Questions and responses are provided in full in Appendix F: COMMENTS ON METHOD.

In response to what he saw as being "*new about the method and the suggested modeling approach*," the external expert emphasized that "this modeling approach represents a new effort to develop a tool of the appropriate scale, complexity and flexibility for the conceptual design process." He has seen that "inflexible computer models that require large efforts to develop have resulted in tools that were unable to adapt to new problems." He emphasizes that this "modeling approach seeks to capture important subsystem interactions and accommodate changes in parameters," and further that a graphical programming language allows "each module to be easily understood" and "modules to be modified for different spacecraft missions in the future." He sees that the "appropriate ease of use and flexibility" in LabVIEW makes it a tool that "would be useful for conceptual design." Other "smaller and simpler tools would miss important subsystem interactions" and "larger and more complex tools would be less flexible to support the conceptual design process."

The external expert says about how he sees "*the suggested method and modeling approach fitting into current design processes*": "This modeling approach is appropriate for a small design team working on conceptual design." However, he stresses that the "introduction of a new common tool, requiring each team member to adapt, will be extremely difficult in an existing design team." He sees the adoption of this kind of tool with "a new team, or a new project " as being more promising. He sees that over time "the total systems model would capture more of the expertise of the engineers," giving the engineers time to "concentrate on tracking new technology changes, and let the model handle routine analysis." Over time, "the depth of the model could increase gradually," but he sees it as important "that the scale of the model remains manageable so that the assumptions within the model are understood."

He says about the advantages "*of applying this method and modeling approach to the design process*": "The total systems model can provide each team member with the approximate response from other subsystems as they explore alternatives within their own subsystem," "the common programming environment" allows "analysis procedures to be updated and communicated in a functional form," and "the ease of programming" allows "the model to respond to changes with a flexibility appropriate to the conceptual design process." These characteristics make it possible "for each team member to explore many design alternatives rapidly". Being able to explore many alternatives early "will improve the selection of a point design." He also sees that the total systems model approach will allow "the team to preserve the reasons for the point design, in case a change in requirements forces a redesign and a return to the conceptual design process. The "ease of programming" has the additional benefit that the model can easily be adapted to "compute different parameters (mass, power, cost) and allow the design to be optimized for different objectives."

As for the "disadvantages of applying this method and modeling approach to the design process" he seems to concentrate on issues related to implementation. He emphasizes that "the benefits of the tool will only be fully demonstrated after the tool is used by a design team on a real problem." Such a demonstration would require real commitment from both the engineers and their managers, and it would require that engineers learn the "graphical programming environment," and that they make "a commitment to make the tool work." This would include a willingness to "expose each calculation they make in conducting an analysis," and a willingness to "remove the ambiguity from where they apply engineering judgment." The external expert emphasizes that this can be difficult as people who have built their reputations on "the value of their expertise will not be motivated to reduce their expertise to a handful of equations." He, therefore, anticipates that the introduction of a method and modeling approach such as that suggested in this research can "become a problem of psychology and politics."

NASA expert 3 says about what he saw as "*being new about the method and the suggested modeling approach*": "space systems design taking into account subsystem interactions is common practice today." It is true that there are some approaches that take into account subsystem interactions, but they are few and they are in most cases geared towards the later stages of the design process. The method suggested in this research focuses on the conceptual design stage. That there is a need for such tools is mirrored in similar efforts by engineers at JPL, who talk about reengineering the design process, and some initial attempts by NASA centers such as Goddard, Johnson, and Langley. He also says that the method and modeling approach does not take into "consideration the entire life cycle." The method can be made to include any phases of the life cycle a given design problem requires. The model made for this demonstration focused on the operational phase, as the sizing and configuring required in the conceptual design stages will be determined primarily by operational phase considerations. If time had been available, manufacturing and testing considerations could also have been included in the model. The same expert claims that the method does not include a concurrent engineering capability. The basis of this statement is not clear, as the focus on total systems analysis, and the mathematical modeling of interactions, makes it possible for designers applying the suggested method, "from the very beginning of a design activity, to consider all elements of the product cycle, from product concept through design, manufacture, service, and even disposal."[1] The method at this stage does not include a cost module, but rather a proxy cost in the form of spacecraft mass.

NASA expert 1 says about how he sees the "*suggested method and modeling approach fitting into current design processes*": "the model would need to be extended to a more generic one before more than a narrow set of missions could be accommodated in even this small part of the total design effort." The idea of the model was to demonstrate the method. A design team using the method would be developing their own total systems model to fit their needs. The "ease of programming," and the modularity of LabVIEW makes modifications and adaptations to different problems simple and quick. NASA expert 3 says, "the method used here alone does not provide for a system design, for example, it was not intended to provide a hardware configuration which is required for thermal design, lifetime estimates, propellant trades, orbit selection, and launch mass estimates." The method provides for a system design if the developed model includes these capabilities. The model developed here does. Thermal design was not included in the model at the recommendation of NASA experts. The model does, however, include mass estimates and hardware configurations sufficient, as shown in Appendix E: THE LABVIEW MODEL, for estimating propellant mass, launch mass, and for selecting orbit.

About "*advantages and disadvantages of applying this method and modeling approach to the design process.*" NASA expert 1 emphasizes that "the advantage of performing design using the proposed method is that an integration of the requirements of the various subsystems can be automatically tracked and kept in synchronization." He also says "allocations of resources to the various subsystems can be easily done [using the model] and the consequences of those allocations can be easily shown." On the other hand, he anticipates that "a great deal of information must be given to the model before any of the advantages can be realized." This might not quite be the case. As part of the modeling process, input, such as data points and mathematical relationships provided by the subsystem engineers, will be embedded in the model and made available to the user. For example, relevant data points may be provided either directly through mouse driven front panel menus or switches, or through "HELP" windows. Examples of both are shown in Appendix E: THE LABVIEW MODEL. Additionally, preferred variable values may be defined as default values and saved with the model. Together, these features make it possible to run quite different, involved, and complex scenarios in minutes, by just changing the value of the variable being investigated. The high number of cases run on the model attest to this. Over time, the input or knowledge embedded in these models will accumulate, steadily increasing their utility. NASA expert 2 says "the advantage is that better detail in the subsystems provides better accuracy in the overall system prediction results." However, he adds that "the disadvantage is in the point design nature for each realization of the method." This statement seems to be contradicted by the flexibility built into LabVIEW used for realizing the method. For a given set of mission requirements, a number of designs can be evaluated quickly and effectively by a user through switches and pull-down menus. The ease of programming and modularity of LabVIEW also makes more involved model modifications a relatively simple task.

About what is "*new about the method and the suggested modeling approach*", NASA expert 4 in Branch B says: "...the best description of the newness of the approach is that JPL and the author of this research have been in a technology race, neither aware of the research of the other. Using a metaphor, a group at JPL has developed the equivalent of Taguchi Methods with the support of substantial funding. The author has independently developed the equivalent of response surface methodology on meager funding. Both lead the rest of NASA in concept and elegance of implementation. Both are neck and neck in terms of the current state of the art." He goes on to say: "The two distinct implementations require similar modeling and do about the same thing today, but the dynamic LabVIEW approach of the author should lead further into the future, given the same amount of support." About the equations developed for this research, he says: "...there is no known NASA application of multidisciplinary design optimization of a satellite at the system level. The show stopper has been the nonexistence of a simple but adequately descriptive system of equations at the system level. Based on the reported simulation results, it appears that the equation system integrated by the author is adequate to be used as the first such example."

Regarding how NASA expert 4 sees "*the suggested method and modeling approach fitting into the current design processes.*" he emphasizes that, "the current, but as yet largely unimplemented, NASA design processes require" that design is performed at "the functional level, which is best described by abstract mathematical models which are largely independent of implementation." He sees that the "the tool developed by the author" is at that level, but that it also "permits the necessary excursions to lower levels to determine size, complexity, and reliability estimates necessary for cost and schedule estimates." He also sees that the developed tool "permits a simple extension to the process and dynamic models which will be needed in the future."

NASA expert 4 says about "*the advantages and disadvantages of applying this method and modeling approach to the design process*": "...the ability of the mathematical models within this method and approach to model, integrate, and determine the behavior of the system functional requirements is a necessity which does not currently exist within NASA, outside of the design-to-cost facility at JPL." He adds that "the approach to building a system of system level equations is valuable for application within multidisciplinary design optimization." He further implies that, when "models must be built for each system," this should be seen as an "advantage, rather than a disadvantage", and that the process of building these models "will facilitate the communication necessary within the group to ensure success, and permit "the simple inclusion of new knowledge as it is acquired." About "potential disadvantages," he says that "engineers do not like to make visible mistakes in their own discipline", and that the suggested approach "tends to reduce 'computer' mistakes so that most of the mistakes will be within the engineering disciplines." According to him, "experience indicates that this may turn off many engineers and discourage them from using this [the suggested] approach."

These comments indicate a favorable impression of the suggested method and modeling approach. The external expert emphasizes the method's focus on subsystems interactions, and the flexibility and ease of programming built into the method through the graphical programming language LabVIEW. These characteristics, he says, makes the model useful for the conceptual design stages, and makes it possible for the various subsystem specialists to explore many design alternatives rapidly and effectively. NASA expert 1 emphasizes that this method and modeling approach makes it possible to integrate, automatically track, and keep in synchronization subsystem requirements. NASA expert 4 emphasizes that the method developed in this research is at the abstract mathematical level required in the functional level design called for by NASA, but that it also can be used for lower level analysis of size, complexity, and reliability required for cost and schedule estimates. He further stresses that the process of building models for each system will facilitate communication between the members of a design team. However, some of the not so favorable comments from NASA experts 1-3 seem to indicate that there might be problems with implementing the suggested method in existing organizations. Some of the comments from NASA expert 4 and the external expert point in the same direction. The external expert stresses, based on his own experience, that the introduction of a new common tool, requiring all team members to adapt, will be extremely difficult in existing design teams. He also sees the unwillingness of engineers to expose their calculations and to reduce their engineering judgment to a handful of equations, as a problem for implementing the suggested method. Along the same lines, NASA expert 4 suggests that the potential of the model to reduce computer mistakes and making engineering mistakes more visible, may discourage many engineers from wanting to use the suggested approach. These and other organizational issues, such as management commitment should be investigated as part of a second step towards implementing the suggested method.

In summary, as a group, the experts seem to agree that the capabilities of the method and modeling approach facilitate the design of complex systems, especially in the early conceptual design stages. There also seems to be a common understanding among these experts that there are a number of management and organizational issues that need to be addressed before this method and modeling approach can be effectively implemented into a design situation.

---

[1]Turino, J. L., *Managing Concurrent Engineering: Buying Time to Market*, New York, Van Nostrand Reinhold, 1992, 8.

8. CONCLUSIONS AND RECOMMENDATIONS

Through this research a total method was developed and demonstrated for sizing and configuring space systems in the early conceptual stages of the design process. As part of the demonstration a complex and realistic model of an Earth remote sensing imager system was developed. The model was developed based on solid physics, engineering, and math theory, as well as on expert opinions. LABVIEW, a state of the art graphical programming language, was used for modeling and modifying the model, and for performing, effectively and rapidly, numerous analysis runs. The model was successfully used to analyze inter-subsystem size and configuration issues central to the conceptual stages of the design process. Issues, as suggested in the literature and by experts, such as technology selection, bottlenecks and technology limits, and on-board data processing were dealt with in these analyses.

There are five contributions coming from this research. First is the development of a method for sizing and configuring space systems focusing on the early conceptual design stages (Hypothesis 1a). Second is the demonstration of how this method can be applied in the design process for Earth remote sensing V/IR imager systems (Hypothesis 1b). Third is the demonstration of how a total systems analysis model can be used, especially in the conceptual design stages, for analyzing the total system impacts of complex and involved inter-subsystem issues, such as on-board processing (Hypothesis 1c). Fourth is the demonstration of how large, complex, and real life total systems models can be built, modified, and analyzed using a common programming tool, in this case LabVIEW (Hypothesis 2). Fifth is the derivation and collection of a full list of system level equations for remote sensing systems (Hypothesis 3).

The method developed in this research emphasizes a total systems analysis approach and the mathematical modeling of interactions. These two concepts are emphasized in the systems engineering and concurrent engineering literature, and by a select number of experts in the space industry.

The method utilizes a descriptive modeling approach to "reveal the structure of complex systems" to show "how elements [subsystems] interact with other elements [subsystems]." The modeling approach is quantitative, making it possible to integrate interrelated elements [subsystems] "as a system rather than having them treated on an individual [subsystem] basis," to consider "major variables of a problem on a simultaneous basis," and to enable "comparisons of many possible solutions [which] can aid in selecting the best of them rapidly and effectively." "...provisions for ease of modifications..." are incorporated into the method through the use of LabVIEW.[1] These concepts form the theoretical foundation of the method. They are captured in its two cornerstones, the total systems analysis approach and the mathematical modeling of subsystem interactions, as well as in the method's emphasis on rapid model development, ease of model modification, and quick and effective evaluation of alternative design options. In the demonstrations of the developed model, spacecraft mass was used as the evaluation and optimization criterion.

The total systems analysis approach is emphasized by Dr. Eileen Stansbery at JSC. In his evaluation of this particular method, Eric L. Dahlstrom, who has provided technical support to NASA for more than 10 years on projects such as the Space Station and the Space Shuttle, especially emphasizes the need for using total systems models in the conceptual design stages.[2] The importance of mathematically modeling the interactions between subsystems is stressed in the "First Lunar Outpost Effectiveness Report." It states that "defining the interactions between major elements of the [total] system" makes it possible to "understand how changes in the performance of one major system affect the performance of another major system or the overall accomplishment of mission objectives."

By incorporating LabVIEW as part of the method and using it for developing the required total systems models, it became possible to create these models quickly and easily, at a 75 % timesaving as compared to line code, to make them easy to modify, and to enable quick and effective analysis of design options. These issues are especially important in the early conceptual design stages. LabVIEW's ease of programming and modification is emphasized by R.M. Wise.[3] E.L. Dahlstrom says about the effectiveness of programming in LabVIEW, "the graphical programming environment of LabView allows a direct conversion of equations into functioning code."[4]

In the process of defining and developing the model for this research, it was shown how easy LabVIEW could be modified to accommodate new information as it became available. The sophisticated analysis tools developed in LabVIEW, including the automatic generation of spreadsheet files for post-analysis purposes, made it possible to run the model and generate results in minutes.

Through focusing on a total systems analysis approach and the mathematical modeling of interactions between subsystems, and by using LabVIEW, it came possible to quickly and efficiently develop a realistic model for testing the method. The model developed for this research describes the interactions between subsystems on a V/IR Earth remote sensing system. The model was developed in close cooperation with subsystem specialists, primarily at NASA Langley Research Center, and contains 300 variables and 130 equations, and uses 1.7 MB of code.

The model was used for demonstrating the benefits of applying the method and modeling approach for making design decisions in the early conceptual design stages. Focus was placed on the major issues of these design stages, size and configuration, and on demonstrating central issues emphasized in the literature and by experts in the space industry.

The demonstration was divided into three parts. In the first part, the major model interactions were analyzed and discussed, highlighting the ability of a descriptive model to show the interactions and dynamics between subsystems and system variables. The demonstrations show the impacts of changes in the Q value on the sensor system, the data processing and storage system, the communications system, and the power system. Based on the results of these demonstrations it was shown that the variables h, T_c, DR_s, $P_{P\&S}$, τ_{AD}, P_{rw}, M_{pTmd}, L_s, and M_{POW} are the variables most sensitive to changes in Q. This information provides insight into the model dynamics and emphasizes that a descriptive model can teach us "more about the system."[5]

In the second part, the model was used for evaluating GaAs and Si types of solar cell for different mission lengths. This type of technology evaluation analysis might reveal a need for development of new technologies to fit mission requirements. Such developments should start early in the design process. In this analysis, it was shown that Si cells should be preferred for missions under 9.1 years long. The analysis performed in this part of the demonstration included only one subsystem.

The third part of the demonstration focused on showing the ability of a total systems model to deal with interactions between subsystems. The demonstration evaluates the impact of on-board processing of sensor data; it shows "how changes in the performance of one major system affect the performance of another major system,"[6] and how a quantitative total systems model makes it possible to consider "major variables of a problem on a simultaneous basis."[7] For example, a change in the sensor data rate impacts the power and mass requirements of the data processing and data storage system, the communication system, and the power system. Isolated subsystem analysis could not have captured these simultaneous impacts. The analyses show that for the given input values and the assumptions made, on-board processing would not be preferred. From the analyses, it was also revealed that the processor system would hit its technology limit if on-board processing was attempted, indicating that developments of new data processor systems are required.

The author hopes that this research can encourage engineers and project managers in the space sector to apply the developed method to other types of space projects. Focus should, initially, be on other satellite project categories, such as communication satellites, as they would entail only minor modifications to the mathematical models developed for this research. Future research should also include the expansion of the method and model to include a cost and value module. The current model was developed with this in mind, and an expansion should not be very time consuming; however, getting the relevant cost and value data might be. A further expansion might also include research into efforts of integrating into the method and model probability analysis and optimization schedules.

The author also suggests research into developing a better mathematical understanding of the relationships between a subsystem's variables and its power and mass. Current focus on point designs derived from physical components seems to neglect these relationships. However, for the development of the sophisticated, mathematical, total systems models required for designing the space systems of the next century, an understanding of these relationships is required.

As suggested in this research and indicated by the feedback from potential users, research also needs to be undertaken into how to implement the suggested method in a real design project. This research might include involving the subsystem experts in the actual model building, and looking into issues of management, psychology, and politics of implementing this type of design method into an organization.

---

[1] Blanchard, B.S., Fabrycky, W.J., *Systems Engineering and Analysis*, Englewood Cliffs, NJ, Prentice-Hall, 1981, 270.
[2] Private communication.
[3] Wise, R.M., Department of Surgery, Thoracic and Cardiovascular Division, School of Medicine, University of Maryland, LabVIEW our Choice for Cardiothoracic Research, Instrumentation Newsletter, *Technical News from National Instruments*, Autumn 1995, Vol. 7, Special Academic Edition, A-4.
[4] Private communication.
[5] Blanchard, B.S., 270.
[6] First Lunar Outpost System Effectiveness Report, EXPO Internal Document No. XE-92-005, Exploration Programs Office, NASA-JSC, May 1992.
[7] Blanchard, B.S., 270.

Appendix A: EQUATIONS

TABLE OF CONTENTS

LIST OF FIGURES

1. Orbital Analysis

1.1. Two Body Motion in Circular Orbits

$T = T_E / Q$ 1

$h = \sqrt[3]{T^2 \mu / 4\pi^2} - R_E$ 2

$\omega = \sqrt{\mu / r^3}$ 3

$T = 2\pi / \omega$ 4

$V = \sqrt{\mu / r}$ 5

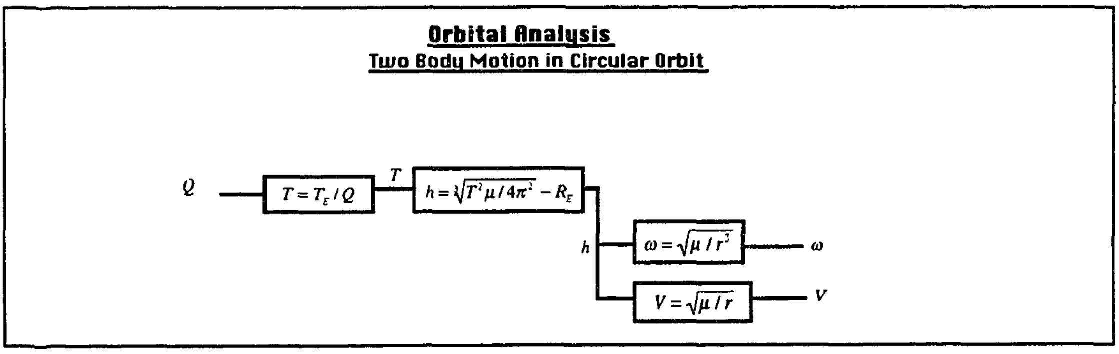


Figure 1-1 Two Body Motion in Circular Orbit

1.2. Angular Displacement, in °, and in km

$\Delta\phi = 2\pi T(1/T_E - 1/T_{ES})$ 6

$AD_e = \Delta\phi R_E$ 7

[1] Stark, J. P. W., Mission Analysis, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 100.

[2] Ibid., Fig 5.17, 101.

[3] Brown, C.D., *Spacecraft Mission Design*, AIAA Education Series, American Institute of Aeronautics and Astronautics, Washington, D.C., 1992, Fig 2.1., 6 & 7.

[4] Ibid.

[5] Ibid.

[6] Stark, J. P. W.. 100.

[7] Ibid.

$$AD_{La} = \Delta\phi R_E \cos La$$

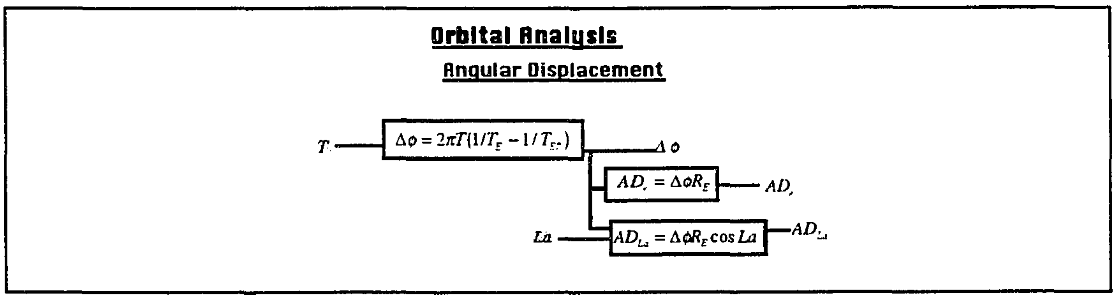


Figure 1-2 Angular Displacement

1.3. Spacecraft Horizon and Swath Width

$\alpha_h = a\cos(R_p / r)$ [8]

$S_w = 2\alpha_h R_p$ [9]

$\beta_h = a\sin\left(R_p / r\right)$ [10]

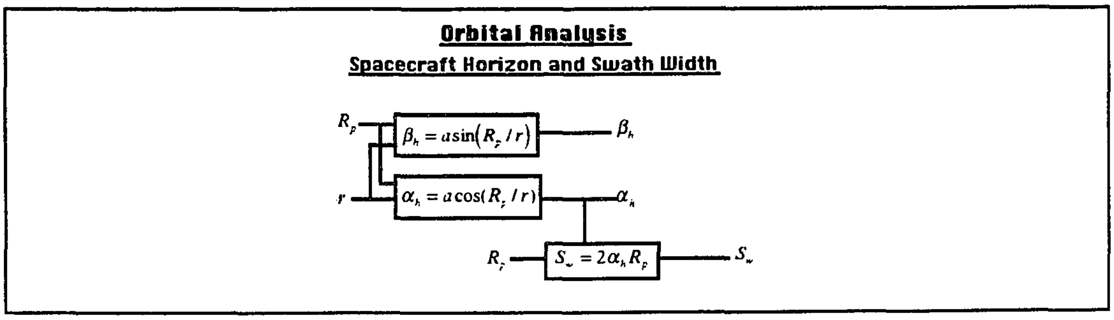


Figure 1-3 Spacecraft Horizon and Swath Width

1.4. Communication Time

$\beta_c = \beta_h - \varepsilon_c$ [11,12]

$\Gamma_c' = a\sin\left[(r / R_p)\sin\beta_c\right]$ [13]

---

[8] Brown, C.D., Fig 4.12, 72.
[9] Ibid., Fig 4.13, 73.
[10] Ibid., Fig 4.12, 72.
[11] Ibid., 77.
[12] Wertz, J.R., Space Mission Geometry, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 100.

$$\alpha_c = \Gamma'_c - \beta_c \qquad 14$$

$$T_c = 2\alpha_c\sqrt{r^3 / \mu} \qquad 15$$

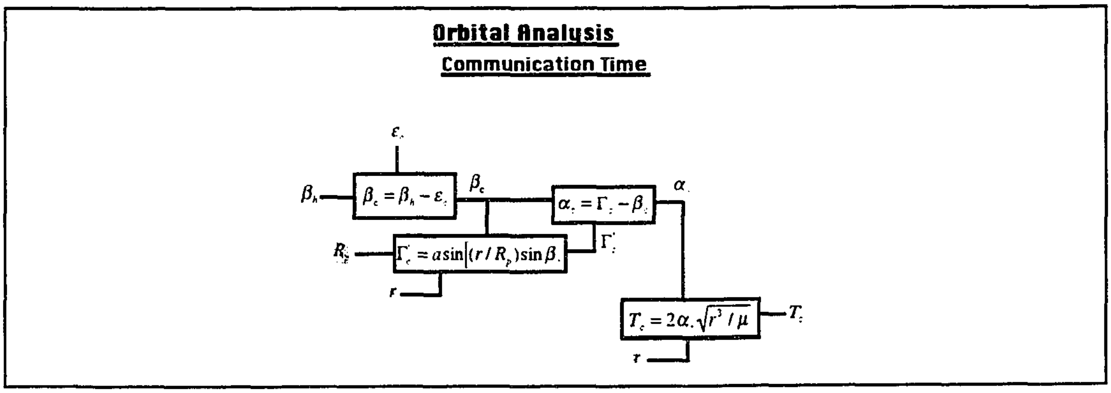


Figure 1-4 Communication Time

1.5. Eclipse Time

$$T_{ec}^{\max} = (T / \pi)\left(a\cos\sin a\cos\left(R_E / r\right)\right) \qquad 16$$

$$T_{ec}^{\min} = T_{ec}^{\max}\left\{1 - (r / R_E)\sin^2\left(i + (23.5\pi / 180)\right)\right\} \qquad 17$$

$$T_{ec}^{eff} = \sqrt{\left[\left(T_{ec}^{\max 2} + T_{ec}^{\min 2}\right) / 2\right]} \qquad 18$$

[13] Brown. C.D.. Fig 4.14. 75, 77.
[14] Ibid.
[15] Ibid.. 77.
[16] Chetty. P. R. K., *Satellite Technology and its Applications*, 2nd ed., Blue Ridge Summit, PA, Tab Books. 1991, 157.
[17] Ibid.. 158.
[18] Ibid.

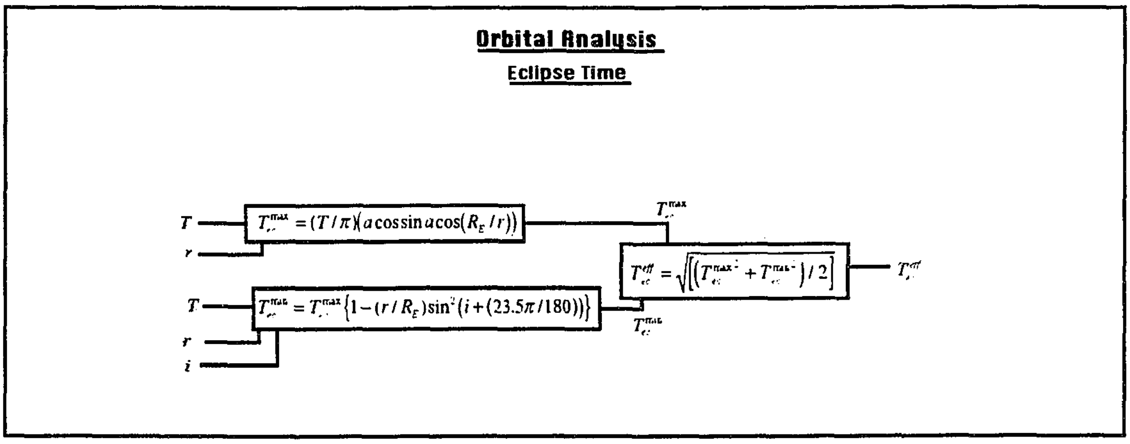


Figure 1-5 Eclipse Time

2. Satellite System Analysis

2.1. Propulsion System

2.1.1. Propellant Required for Launch, Using a Hohman Transfer Approximation

$Az = a\sin(\cos i / \cos La)$ [19,20]

$V_E = (2\pi R_E \cos La / T_E)\sin Az$

$a = (r_{ta} + r_{tp})/2$ [21,22]

$V_{tp} = \sqrt{2\mu / r_{tp} - \mu / a}$ [23]

$V_{ta} = \sqrt{2\mu / r_{ta} - \mu / a}$ [24]

$\Delta V_{Ee} = V_{tp} - V_E$ [25]

---

[19] Boden, D.G., Introduction to Astrodynamics, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 136.
[20] Griffin, M. D., French, J. R., *Space Vehicle Design*, AIAA Education Series, American Institute of Aeronautics and Astronautics, Washington, D.C., 1991, 108.
[21] Brown, C.D., Fig 3.4, 43-44.
[22] Boden, D.G., 127-128 & 138.
[23] Ibid.
[24] Ibid.
[25] Ibid.

$$\Delta V_{eo} = V - V_{ta} \qquad 26$$

$$\Delta V = \Delta V_{Ee} + \Delta V_{eo} \qquad 27$$

$$M_p = M(e^{\Delta V/gIsp} - 1)$$

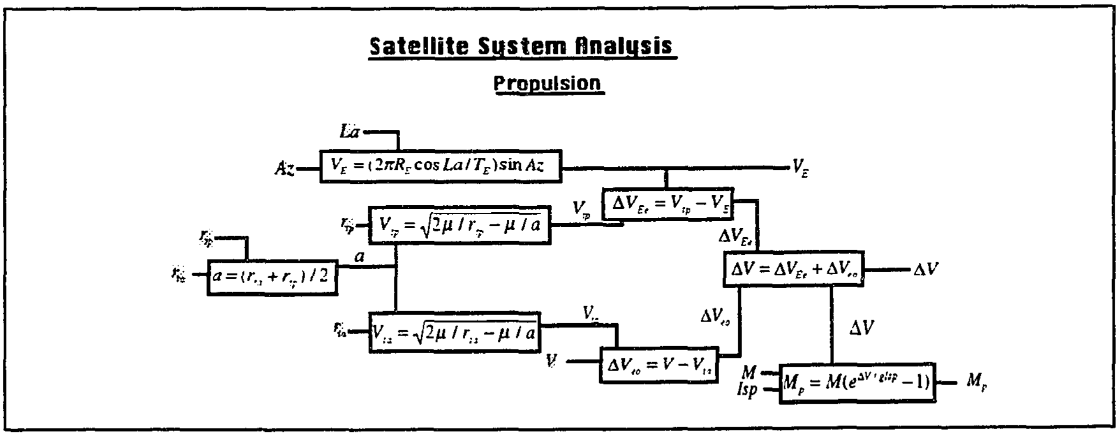


Figure 2-1 ΔV and Propellant Mass Required to Launch Spacecraft

2.1.2. Propellant Required for Orbit Reboosting.

$$r_l = r - \Delta h_c \qquad 28$$

$$V_l = \sqrt{\mu / r_l} \qquad 29$$

$$a_c = (r + r_l)/2 \qquad 30$$

$$V_{cp} = \sqrt{2\mu / r_l - \mu / a_c} \qquad 31$$

$$V_{ca} = \sqrt{2\mu / r - \mu / a_c} \qquad 32$$

[26] Ibid.
[27] Ibid.
[28] Author's notes 26 March 1995, 196-197.
[29] Brown, C.D., Fig. 2.1., 6; Author's notes 26 March 1995, 196-197.
[30] Brown, C.D., Fig 3.4, 43 & 44; Boden, D.G., 128 & 131; Author's notes 26 March 1995, 196-197.
[31] Ibid.
[32] Ibid.

$$\Delta V_{lc} = V_{cp} - V_l \qquad \mathbf{33}$$

$$\Delta V_{co} = V - V_{ca} \qquad \mathbf{34}$$

$$\Delta V_c = \Delta V_{lc} + \Delta V_{co} \qquad \mathbf{35}$$

$$M_{pi} = M(e^{\Delta V_c / g Isp_b} - 1)$$

$$T_{co} = 2\pi\sqrt{a_c^3/\mu} \qquad \mathbf{36}$$

$$T_{Hc} = 1/2T_{co} \qquad \mathbf{37}$$

$$T_{\Delta h} = \Delta h_c T/2\pi(C_D A_{SC}/M)\rho r^2 \qquad \mathbf{38}$$

$$N_{rb} = T_{lt}/(T_{\Delta h} + T_{Hc}) \qquad \mathbf{39}$$

$$F_E^{Th} = M_{pi}^{max} Isp_b g/T_E^{bt} \qquad \mathbf{40,41,42}$$

[33] Ibid.
[34] Ibid.
[35] Ibid.
[36] Brown, C.D., 6-7.
[37] Brown, C.D., Fig 3.4, 43 & 44; Boden, D.G., 128 & 131; Author's notes 26 March 1995, 197-199 and 27 April 1995, 238-242.
[38] Boden, D.G., 127-128 & 138; Author's notes 26 March 1995, 197-199 and 27 April 1995, 238-242.
[39] Author's notes 26 March 1995, 197-199.
[40] Zermuehlen, R.O., Zimbelman, H.F., Guidance, Navigation, and Control, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 320.
[41] Author's notes 11 March 1995, 186 & 188, 27 March 1995, 200-202, and 24 June 1995, 262-263.
[42] Sackheim, R.L., Wolf, R.S., Space Propulsion Systems, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 581.

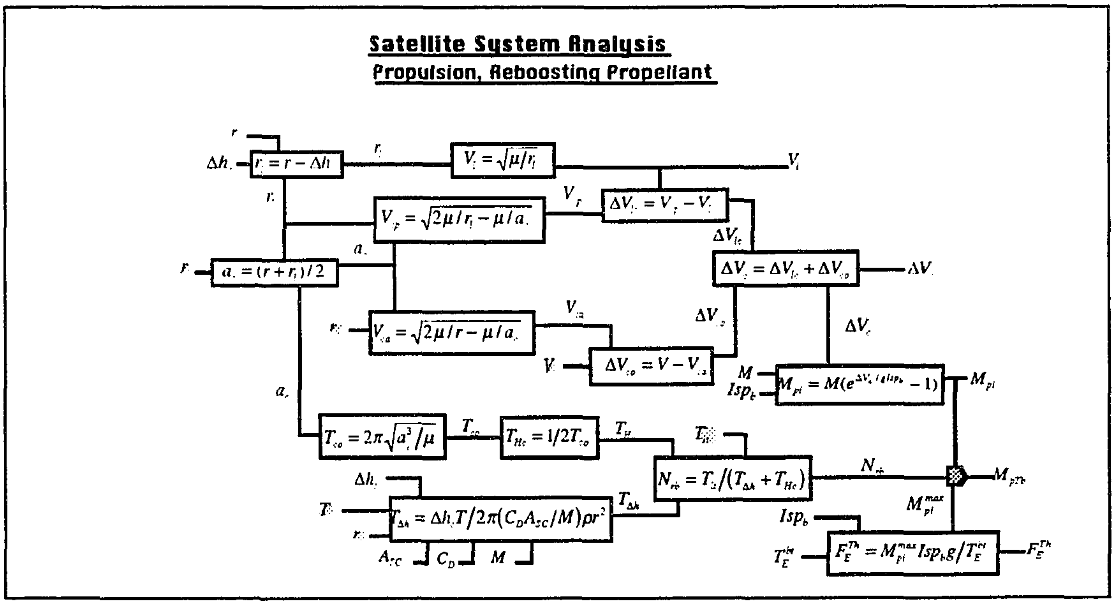


Figure 2-2 Propellant for Reboosting

2.2. Sensor System

$M_o = 3.8598e^{6.532D}$ **43**

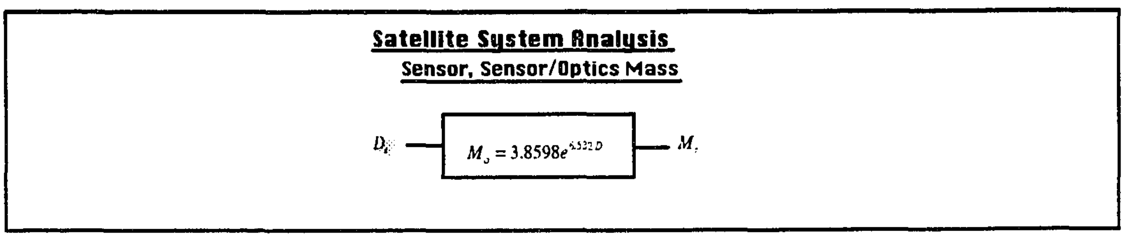


Figure 2-3 Sensor/Optics Mass

2.2.1. S/N Ratio and Sensor Aperture Diameter, and Focal Length for Detector Noise Limited (dnl) and Photon Noise Limited (pnl) Systems

$B = \left[(R_S / h_{ES})^2 \right]; B = 1$ **44,45,46**

---

[43] Author's Excel file: Sat.Syst.An.;D. vs. Mo.

[44] Reeves. R. G., Anson, A., Landen, D. (Editors), *Manual of Remote Sensing*, Vol.1, American Society of Photogrammetry, Falls Church, VA, 1975, 60-61.

[45] Jamieson. J.A., McFee, R.H., Plass, G.N., Grube, R.H., Richards, R.G., *Infrared Physics and Engineering*, New York, McGraw-Hill Book Company. 1963, 13-14.

$$F_{em} = B\left[\frac{2\pi c^P c^2}{\lambda^5}\left(\left(e^{(c^P c/\lambda c^B t)} - 1\right)\right)^{-1}\right]\Delta\lambda$$ [47,48,49]

$$\Delta\theta = a\tan(d_d / f)$$ [50]

$$P_D = \left(\frac{(D\Delta\theta)^2}{4}\right)F_{em}$$ [51]

$$\text{Sc.}: N_{pix} = FOV / \Delta\theta;\ \text{St.}: N_{pix} = 1$$ [52,53,54]

$$T_d = \frac{h\left(d_d/f\right)}{VN_{pix}}$$ [55,56]

$$T_i = \gamma T_d$$ [57]

$$\delta f_n = \psi\frac{1}{T_i}$$ [58,59]

$$A_d = d_d^2, \text{ or } \pi(d_d/2)^2$$ [60]

---

[46] Author's notes 25 August 1995, 293; Meetings with S. Katzberg, 23 May 1995 and 25 May 1995.

[47] Hopper, G.S., Forward-Looking Infrared Systems, in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Co-published by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington, 1993, 107-108; Author's notes 05 January 1995, 88.

[48] Brodsky, R.F., Defining and Sizing Space Payloads, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 220.

[49] Silva, L. R., Radiation and Instrumentation in Remote Sensing, in Swain, P. H., Davis, S. M. (Editors), *Remote Sensing, The Quantitative Approach*, New York, McGraw-Hill, Inc., 1978, 27-28.

[50] Hopper, G.S., 116; Author's notes 08 January 1995, 92-93, 94.

[51] Hopper, G.S., 116-117; Author's notes 08 January 1995, 92-93.

[52] Acetta, J.S., Infrared Search and Track Systems, in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Co-published by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington, 1993, 296-297.

[53] Notes and author's comments from meeting with S. Katzberg, 16 March 1995, 3.

[54] Author's notes 14 July 1995, 265.

[55] Hopper, G.S., 132.

[56] Notes and author's comments from meeting with S. Katzberg, 16 March 1995, 3.

[57] Ibid., 2.

[58] Ibid., 63; Author's notes 14 December 1994, 84.

[59] Ibid., 132.

[60] Ibid., 62; Author's notes 14 December 1994, 84.

$$NEP = \frac{(A_d)^{\frac{1}{2}}(\delta f_n)^{\frac{1}{2}}}{D^*} \qquad \mathbf{61}$$

$$d_\lambda = \frac{1.22\lambda h}{D} \qquad \mathbf{62}$$

$$d_{pix} = h\frac{d_d}{f} \qquad \mathbf{63,64}$$

$$d_{eff} = \sqrt{(d_\lambda)^2 + (d_{pix})^2 + (VT_i)^2} \qquad \mathbf{65}$$

$$\frac{S}{N}_{dnl} = \frac{P_D}{NEP}$$

$$\frac{S}{N}_{dnl} = \frac{D^* F_{em}\left(a\tan\left(\frac{d_d}{f}\right)\right)^2 (1.22\lambda h)^2}{4\left((\delta f_n)^{\frac{1}{2}}(A_d)^{\frac{1}{2}}\right)\left\{(d_{eff})^2 - (d_{pix})^2 - (VT_i)^2\right\}} \qquad \mathbf{66}$$

$$N_p = P_D\left(\frac{\lambda}{c^P c}\right) \qquad \mathbf{67}$$

$$\frac{S}{N}_{pnl} = \sqrt{N_p T_i} \qquad \mathbf{68}$$

$$\frac{S}{N}_{pnl} = \sqrt{\frac{(1.22\lambda h)^2 F_{em}(a\tan(d_d/f))^2 \lambda T_i}{4\left[(d_{eff})^2 - (d_{pix})^2 - (VT_i)^2\right]c^P c}} \qquad \mathbf{69}$$

---

[61] Ibid., 59-64; Author's notes 14 December 1994, 84.
[62] Brodsky, R.F., Fig 9.4b, 230.
[63] Ibid., 228.
[64] Hopper, G.S., 116; Author's notes 08 January 1995, 92-93.
[65] Meetings with S. Katzberg, 01 February 1995 and 12 February 1995.
[66] Author's notes from meeting with S. Katzberg, 12 February 1995, 5-6; Author's derivations 15 & 17 February 1995, 146-155, and 01 March 1995, 164-165.
[67] Hopper, G.S., 107-108; Author's notes 05 January 1995, 88; Author's derivations 01 March 1995, 159; Author's notes from meeting with S. Katzberg, 01 February 1995, 5 and 23 February 1995.
[68] Author's derivations 01 March 1995, 159; Author's notes from meeting with S. Katzberg, 01 February 1995, 5 and 23 February 1995.
[69] Author's notes from meeting with S. Katzberg, 12 February 1995, 5-6; Author's derivations 01 March 1995, 159-165.

$$\frac{S}{N_s} = \left[\left(\frac{S}{N_{dnl}}\right)^{-2} + \left(\frac{S}{N_{pnl}}\right)^{-2}\right]^{-1/2} \qquad \mathbf{70}$$

$$\frac{S^e}{N_s} = \left[\left(\frac{S}{N_s}\right)^{-2} + \left(\frac{1}{12 * 2^b}\right)^{2}\right]^{-1/2} \qquad \mathbf{71,72}$$

$$D_{dnl} = \sqrt{\frac{4\left(S/N_s^e (\delta f_n)^{\frac{1}{2}} (A_d)^{\frac{1}{2}}\right)}{\left(a\tan\left(\frac{d_d}{f}\right)\right)^2 F_{em} D^*}} \qquad \mathbf{73}$$

$$D_{pnl} = \sqrt{\frac{4c^P c\left(S/N_s^e\right)^2}{\left(a\tan(d_d/f)\right)^2 F_{em} \lambda T_i}} \qquad \mathbf{74}$$

$$F^\# = \frac{f}{D} \qquad \mathbf{75}$$

---

70 Author's notes from meeting with S. Katzberg, 26 July 1995.

71 Ibid.

72 Davies, R.S., Communications Architecture, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 13.4, 452 & Fig 13.4, 453.

73 Ibid.

74 Ibid.

75 Brodsky, R. F., 228.

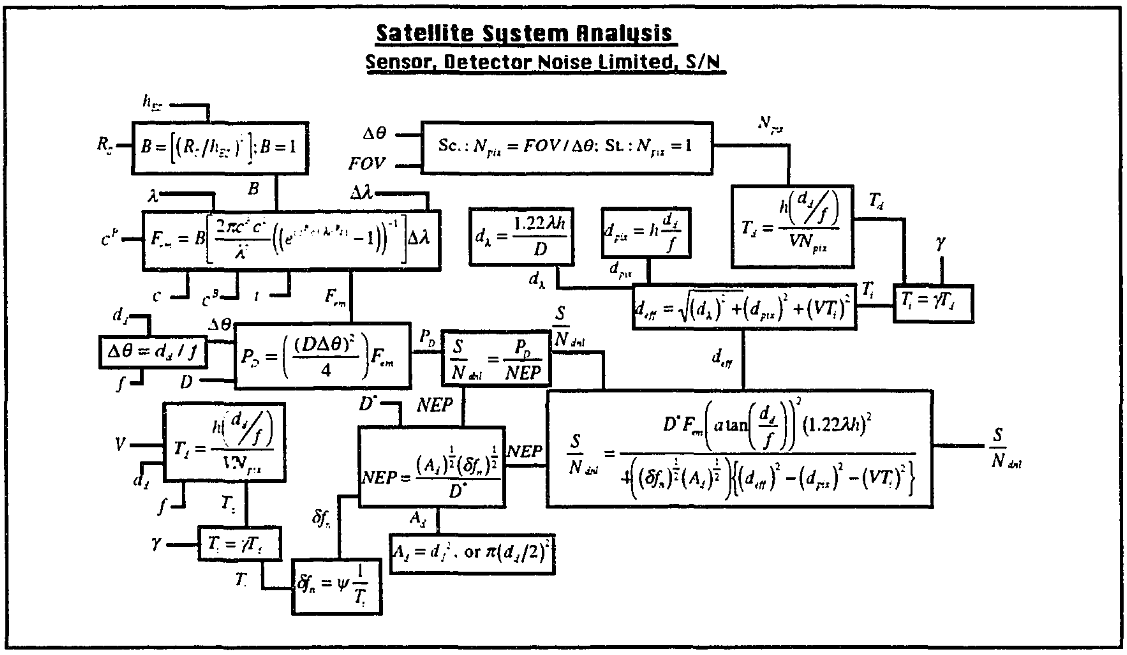


Figure 2-4 S/N ; Detector Noise Limited System

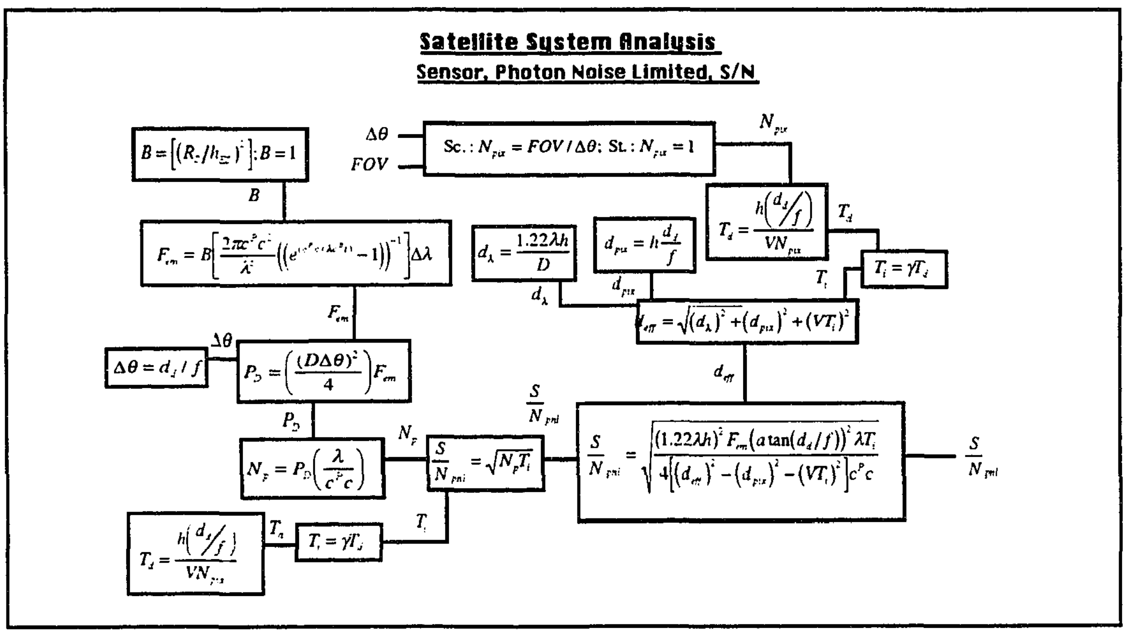


Figure 2-5 S/N : Photon Noise Limited System

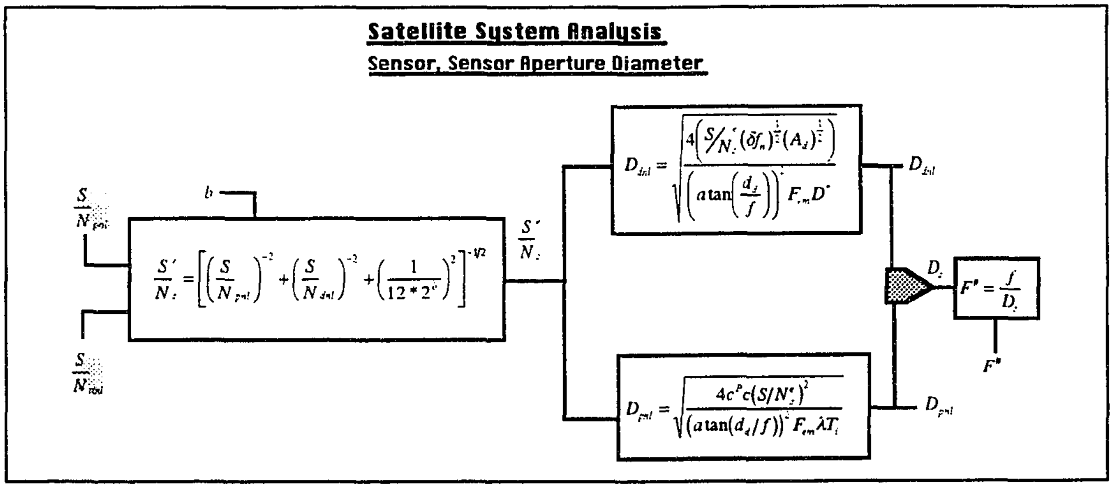


Figure 2-6 Sensor Aperture Diameter

2.2.2. Scanner Power Consumption

$T_{mi} = N_{pix} T_i$ **76**

$\xi_{mi} = FOV \big/ \left[2(p_{\%} T_{mi})^2 + \left((p_{\%} T_{mi})((1 - 2p_{\%}) T_{mi}) \right) \right]$ **77,78**

$M_{mi} = (2 r_{mi} y_{mi} x_{mi}) \upsilon_{mi}$ **79**

$I_{mi} = M_{mi} \left(\left((r_{mi})^2 + (x_{mi})^2 \right) \big/ 12 \right)$ **80,81,82,83**

$P_{mi} = I_{mi} (\xi_{mi})^2 p_{\%} T_{mi}$ **84,85**

---

[76] Author's notes 14 April 1995, 221-222, and 16 July 1995, 267.
[77] Ibid.
[78] Author's notes 19 April 1995, 227.
[79] Ibid.
[80] Ibid.
[81] Author's notes from meeting with S. Katzberg, 16 March 1995, 5; Author's notes 28 January 1995, 137-141 and 01 March 1995, 166-168.
[82] Chetty, P. R. K., Appendix A (Reprint: CRC Handbook of Chemistry and Physics, 71st Edition, Copyright CRC Press, Inc., Boca Raton, FL), 513.
[83] Doukas, P.G., McCandless, J.R., Sarafin, T.P., Britton, W.R., Structures and Mechanisms, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 407.
[84] Author's notes 12 March 1995, 194, and 14 April 1995, 221-222.
[85] Author's notes 19 April 1995, 228-229; Telecon with S. Katzberg, 18 April 1995, 2-3; Telecon with E. L. Dahlstrom, 19 April 1995, 1-2.

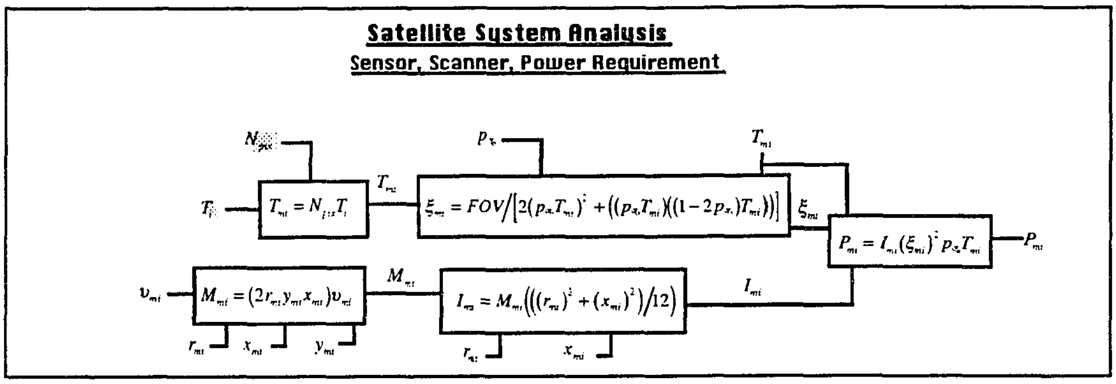


Figure 2-7 Scanner Power Requirement

2.2.3. Sensor Horizon and Swath Width

$\beta_i = FOV/2$ **86**

$\Gamma' = a\sin\left((r/R_p)\sin\beta_i\right)$ **87**

$\alpha_i = \left(\Gamma' - \beta_i\right)$ **88**

$S_i = 2\alpha_i R_p$ **89**

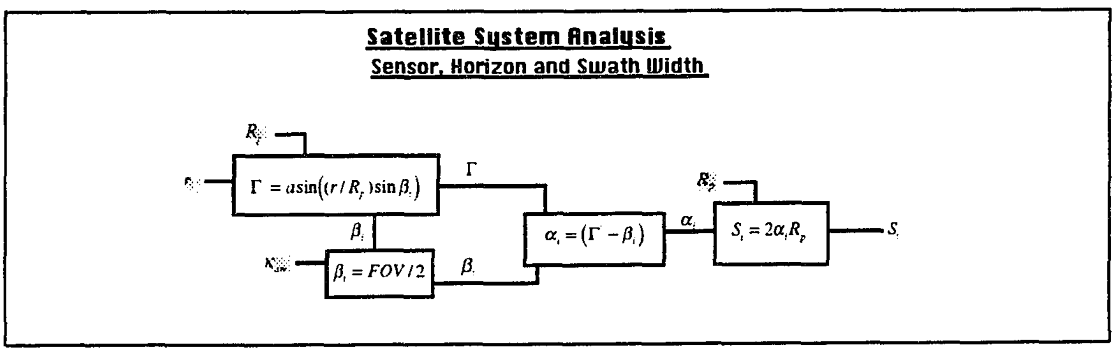


Figure 2-8 Sensor Horizon and Swath Width

[86] Brown, C.D., Fig 4.14, 75.
[87] Ibid.
[88] Ibid.
[89] Ibid.

2.2.4. Sensor Data Rate

$$\text{Sc.}: N^{sim}_{pix/ch} = N^{s/sc}_{pix/ch};\ \text{St.}: N^{sim}_{pix/ch} = (FOV/\Delta\theta)^2$$ **90,91**

$$DR_s = {N^{sim}_{pix/ch} sbe_{SF} N_{ch}} \Big/ {T_i}$$ **92,93**

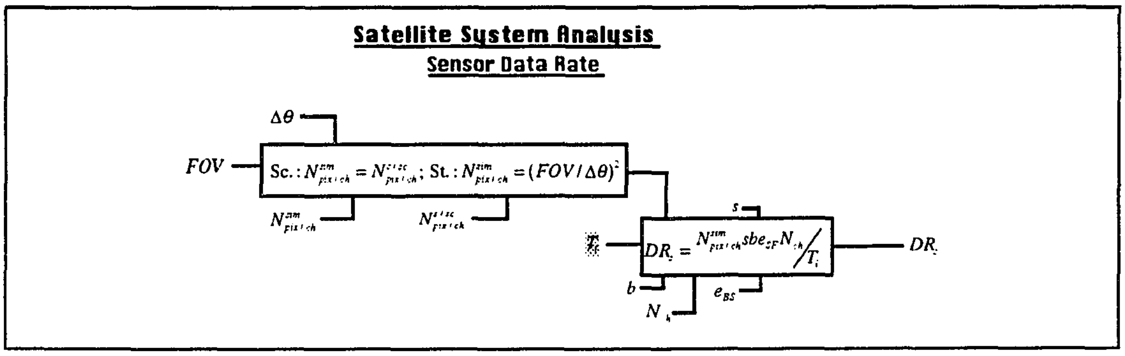


Figure 2-9 Sensor Data Rate

2.3. Data Storage and Processing System

2.3.1. Data Processing

$$N^s_s = DR_s / b$$ **94**

$$N^i_{IPS} = N^i_s N^i_{i/s}$$ **95,96**

$$N^s_{IPS} = N^s_s N^s_{i/s}$$ **97,98**

---

[90] Author's notes 14 July 1995, 265.

[91] Cantella, M.J., Staring-Sensor Systems. in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Copublished by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington. 1993, 159-163.

[92] Brodsky, R. F., 250.

[93] Author's notes from meeting with S. Katzberg, 23 March 1995, 4; Author's notes 26 March 1995, 196.

[94] Author's notes 04 April 1995, 212, and 06 April 1995, 218.

[95] Telecon with G. Ganoe, 31 March 1995; Author's notes 31 March 1995, 206-207, 04 April 1995, 212-213, and 06 April 1995, 218.

[96] Glaseman, S., Hansen, L.J., Pollock, C.H., Thimlar, M., Spacecraft Computer Systems, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 545.

[97] Ibid.

[98] Telecon with G. Ganoe, 31 March 1995; Author's notes 31 March 1995, 206, 04 April 1995, 212-213, and 06 April 1995, 218.

$$N_{TIPS}^{HK} = \sum N_{IPS}^{i}$$ [99]

$$N_{TIPS} = N_{TIPS}^{HK} + N_{IPS}^{s}\delta_{P}$$ [100]

$$P_{CPU} = N_{TIPS}\zeta_{CPU}^{p}$$ [101,102,103,104]

$$M_{CPU} = N_{TIPS}\zeta_{CPU}^{m}$$ [105,106,107,108]

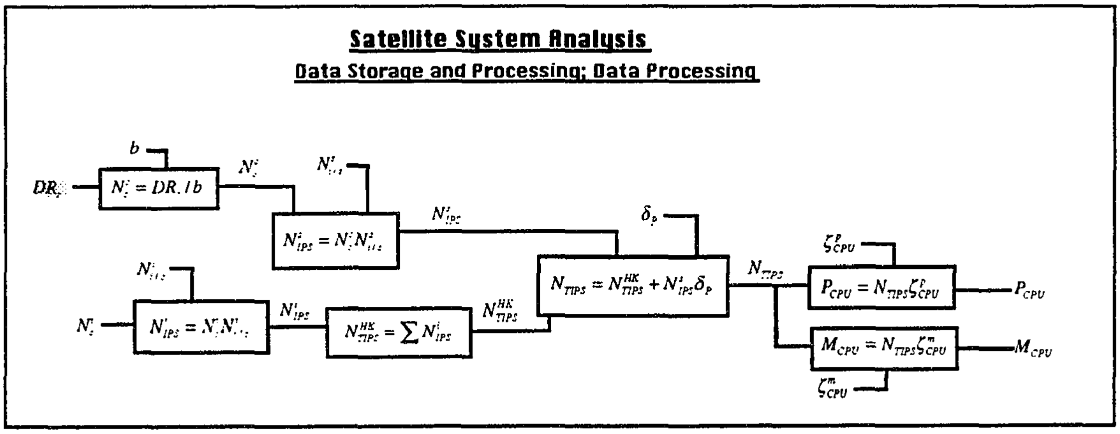


Figure 2-10 Calculating Processing Power; Power Requirements

2.3.2. Data Storage

$$T_{\Delta GS} = \left[(T - N_{GS}T_{c})/N_{GS}\right]$$ [109]

$$DR_{HK} = \sum N_{s}^{i}N_{b/s}^{i}$$ [110]

---

[99] Author's notes 31 March 1995, 208, 04 April 1995, 212-213, and 06 April 1995, 218.
[100] Author's notes 04 April 1995, 213, and 06 April 1995, 218.
[101] Author's notes 31 March 1995, 209, and 04 April 1995, 213.
[102] Glaseman. S., Table 16.9, 569.
[103] Hansen. L.J., Pollock, C.H., Spacecraft Computer Systems, in Wertz, J.R., Larson, W.J. (Editors). *Space Mission Analysis and Design*, 2nd ed., London, Kluwer Academic Publishers, 1992, Table 16.10, 626.
[104] Author's Excel files: Sat.Syst.An.;CPU Regr. M&P=1 & Sat.Syst.An.;CPUNewRegr. M&P=1
[105] Ibid.
[106] Hansen. L.J., Table 16.10, 626.
[107] Author's notes 31 March 1995, 209, and 04 April 1995, 213.
[108] Glaseman. S., Table 16.9, 569.
[109] Author's notes 30 March 1995, 203.
[110] Author's notes 06 April 1995, 218-219.

$$DS_{HK} = DR_{HK}(T_C + T_{\Delta GS})$$ [111]

$$DS_s = DR_s K_{\%} T_{\Delta GS} \vartheta_T$$ [112,113,114,115]

$$DS = DS_s + DS_{HK}$$ [116]

$$P^T_{DS} = 0.0002 DS^{0.5727}$$ [117,118]

$$P^{SS}_{DS} = P^{SS}_F + \left(\left(DS - DS^{SS}_F\right) / DS^{SS}_{inc}\right) P^{SS}_{inc}$$ [119,120]

$$M^T_{DS} = DS\left(9 * 10^{-10}\right) + 4.2596$$ [121,122]

$$M^{SS}_{DS} = M^{SS}_F + \left(\left(DS - DS^{SS}_F\right) / DS^{SS}_{inc}\right) M^{SS}_{inc}$$ [123,124]

---

[111] Ibid.
[112] Author's derivation; Author's notes 30 March 1995, 203.
[113] Author's notes 04 April 1995, 213.
[114] Author's notes 06 April 1995, 218-219.
[115] Davies, E. S., 456.
[116] Ibid.
[117] Boatwright, J.E., Mueller, R.L., Command and Data Handling, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 11.26, 348.
[118] Author's Excel file: Sat.Syst.An.;DS Regression, vP=0; Author's notes 04 April 1995, 213.
[119] Author's notes 30 March 1995, 204-205, and 18 April 1995, 224; Author's notes from meeting with G. Ganoe, 24 March 1995, 4.
[120] Boatwright, J.E., Table 11.26, 348.
[121] Author's Excel file: Sat.Syst.An.;DS Regression, vM=0; Author's notes 04 April 1995, 213.
[122] Boatwright, J.E., Table 11.26, 348.
[123] Ibid.
[124] Author's notes 30 March 1995, 204-205, and 18 April 1995, 224; Author's notes from meeting with G. Ganoe, 24 March 1995, 4.

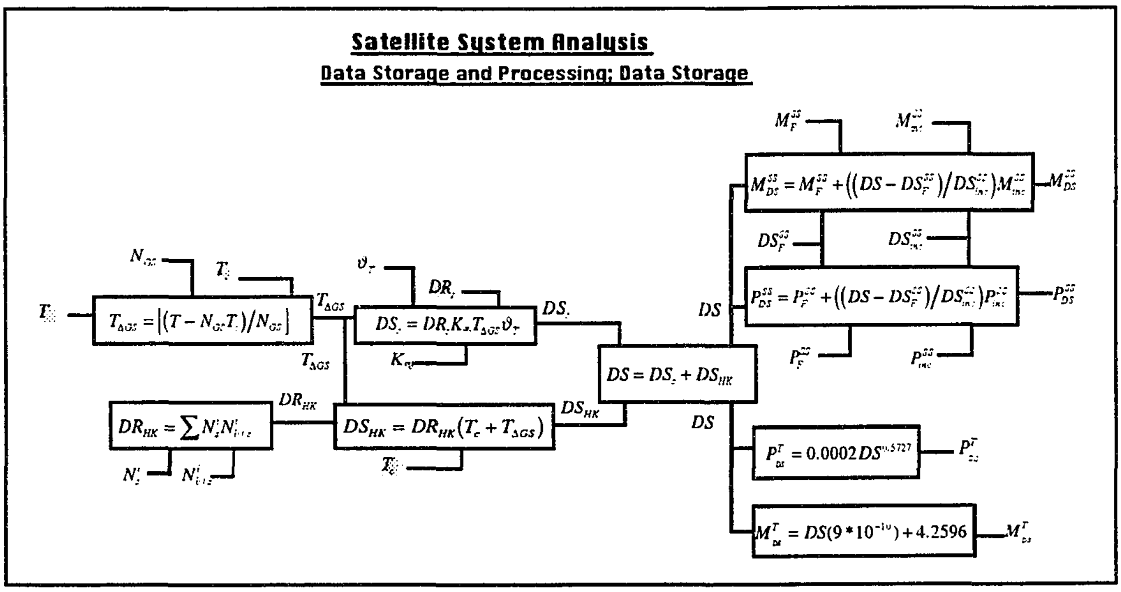


Figure 2-11 Calculating Storage Requirements; Power Requirements

2.4. Communication System

2.4.1. Communication Dumprate

$$DR_c^s = DS_s / (T_c - (T_{ca} + T_{cb}))$$ **125,126,127**

$$DR_c^{HK} = DS_{HK} / (T_c - (T_{ca} + T_{cb}))$$ **128**

$$DR_c = DR_c^{HK} + DR_c^s$$ **129**

---

[125] Whitworth, G. G., Ground System Design and Sizing, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 539.

[126] Author's notes from meeting with George Ganoe, 21 March 1995, 1; Author's notes 30 March 1995, 203.

[127] Author's notes 04 April 1995, 213, and 06 April 1995, 218-219.

[128] Ibid.

[129] Ibid.

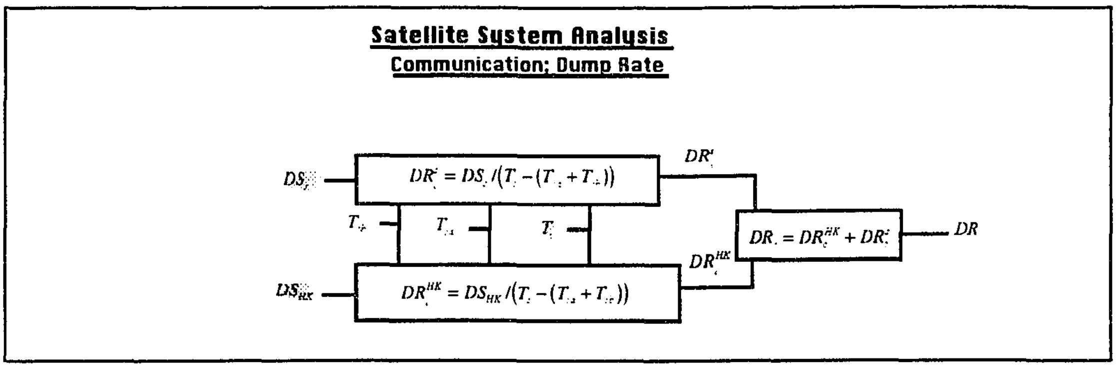


Figure 2-12 Communication Dumprate

2.4.2. Communication Power

$L_s = (\lambda_c / 4\pi h)^2$ **130**

$P^T_{COM} = E_b c^B t_s DR_c / L_l G_t L_s L_a G_r N_0 e_T$ **131,132**

$G_r = \pi^2 D^2_{ra} \varpi_{ra} / \lambda^2_c$ **133**

$G_t = \pi^2 D^2_{ta} \varpi_{ta} / \lambda^2_c$ **134**

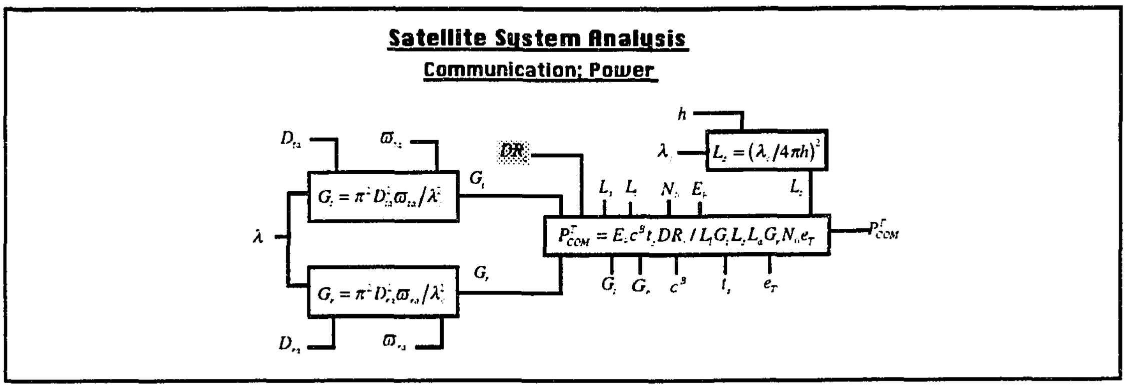


Figure 2-13 Power Required for the Communication System

---

[130] Davies, R. S., 463.

[131] Ibid.. 458-461.

[132] Smith, H., Telecommunications, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 343-345.

[133] Davies, R. S., 458-461.

[134] Ibid.

2.4.3. Communication Mass

$$M_{TR} = 2(\zeta^{m}_{COM} P^{T}_{COM}) \quad \textbf{135}$$

$$M^{m}_{ANT} = D_{ta} \zeta^{m}_{ANT} \quad \textbf{136}$$

$$M_{ANT} = M^{m}_{ANT} + 2M_{HEM} + M^{o}_{ANT} \quad \textbf{137}$$

2.5. Guidance Navigation & Control (GN&C)

2.5.1. Effects on Spacecraft from Atmospheric Drag

$$F_{AD} = (1/2) C_d A_{SC} \rho V^2 \quad \textbf{138,139,140}$$

$$\tau_{AD} = F_{AD} r_{cp} \quad \textbf{141,142,143}$$

$$H_{AD/O} = \tau_{AD} T \quad \textbf{144,145,146}$$

$$\Delta\varphi_{AD/O} = \left((\tau_{AD}/I_{SC}) T^2\right)(180/\pi) \quad \textbf{147}$$

---

[135] Ford, J., Communications, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 11-22, 341.

[136] Davies, R. S., Table 13-15, 482.

[137] Ibid.

[138] Griffin, M. D., 292.

[139] Author's notes 10 March 1995, 183, and 12 March 1995, 193.

[140] Gorney, D.J., Blake, J.B., Koons, H.C., Schultz, M., Vampola, A., Walterscheid, R.L., The Space Environment, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 195.

[141] Griffin, M. D., 292.

[142] Fortesque, P. W., Dynamics of Spacecraft, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 43.

[143] Zermuehlen, R.O., 320.

[144] Ibid., 321.

[145] Author's notes 12 March 1995, 193.

[146] Fortesque, P. W., 42.

[147] Griffin, M. D., 293.

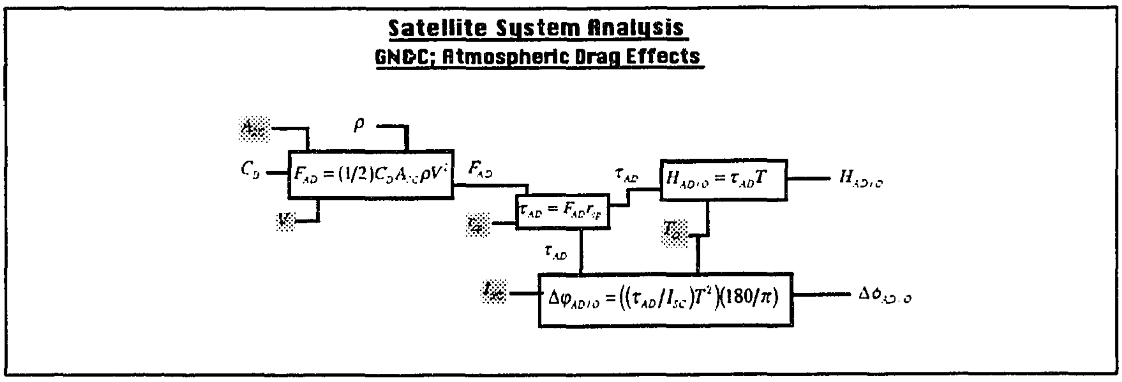


Figure 2-14 Atmospheric Drag Effects

2.5.2. Sizing the Reaction Wheel

$M_{rw} = 2\tau_{AD}/\xi_{rw} r_{rw}^2$ **148**

$I_{rw} = M_{rw}(r_{rw}^2/2)$ **149,150,151**

$T_{\Delta nd} = \omega_{rw}^{max}/\xi_{rw}$ **152**

$P_{rw} = I_{rw}(\xi_{rw})^2 T_{\Delta nd}$ **153,154**

[148] Author's notes 10 March 1995, 183.
[149] Ibid.
[150] Chetty, P. R. K., Appendix A, 513.
[151] Doukas, P.G, 407.
[152] Author's notes 27 March 1995, 200-202.
[153] Author's notes 12 March 1995, 194.
[154] Author's notes 19 April 1995, 228-229; Telecon with S. Katzberg, 18 April 1995, 2-3; Telecon with E.L. Dahlstrom, 19 April 1995, 1-2.

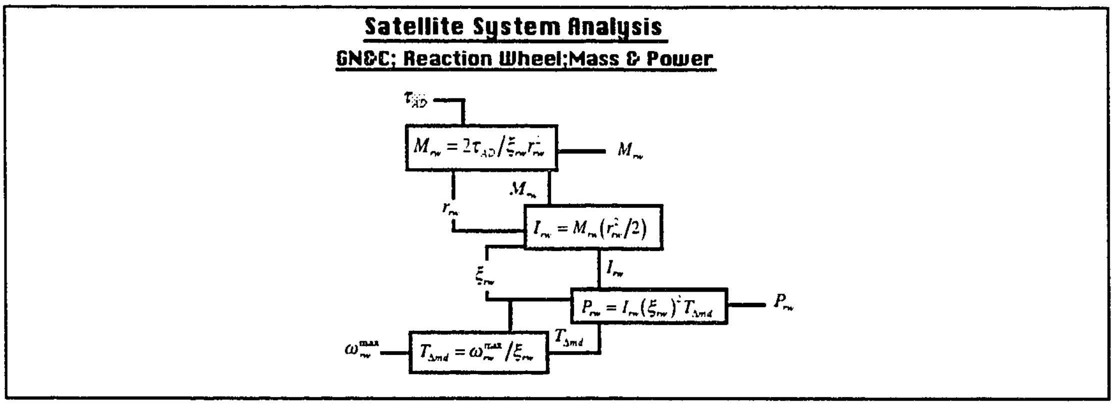


Figure 2-15 Reaction Wheel; Mass and Power

2.5.3. Momentum Dumping Using Thrusters; Force, Propellant Mass, and Electrical Power for Electrical Thrusters

$H_{rw}^{\max} = I_{rw}\omega_{rw}^{\max}$ **155,156,157**

$\tau_{Th} = H_{rw}^{\max}/T_{b/P}$ **158,159,160**

$F_{Th/md} = \tau_{Th}/l_{Th-a}$ **161,162,163**

$F_{Th/md} = H_{rw}^{\max}/T_{b/P}\,l_{Th-a}$ **164**

$F_{Th/md} = M_{p/md} Isp_{Th} g/T_{b/P}$ **165,166,167**

$M_{p/md} = H_{rw}^{\max}/g Isp_{Th}\,l_{Th-a}$

---

[155] Fortesque, P. W., 44.
[156] Chetty, P.R.K., 206.
[157] Agrawal, B. N., *Design of Geosynchronous Spacecraft*, Englewood Cliffs, NJ, Prentice-Hall, 1986, 112.
[158] Author's notes 10 March 1995, 183.
[159] Fortesque, P. W., Attitude Control, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 230.
[160] Author's notes 10 March 1995, 183, and 27 March 1995, 200-202.
[161] Sackheim, R.L., 581.
[162] Author's notes 11 March 1995, 188&186, 12 March 1995, 194, and 27 March 1995, 200-202.
[163] Zermuehlen, R.O., 320.
[164] Author's notes 27 March 1995, 200-202.
[165] Zermuehlen, R.O., 320.
[166] Author's notes 11 March 1995, 186 & 188, and 27 March 1995, 200-202.
[167] Sackheim, R.L., 581.

$$P_{Th/md} = M_{p/md}\left(Isp_{Th}g\right)^2 \big/ 2e_{elTh}T_{b/P}$$ **168,169,170**

$$P_{Th/md} = \left(H_{rw}^{\max} Isp_{Th}g\right)\big/ 2e_{elTh}T_{b/P}l_{Th-a}$$ **171**

$$M_{pTmd} = M_{p/md}\,T_{lt}/T_{\Delta md}$$ **172**

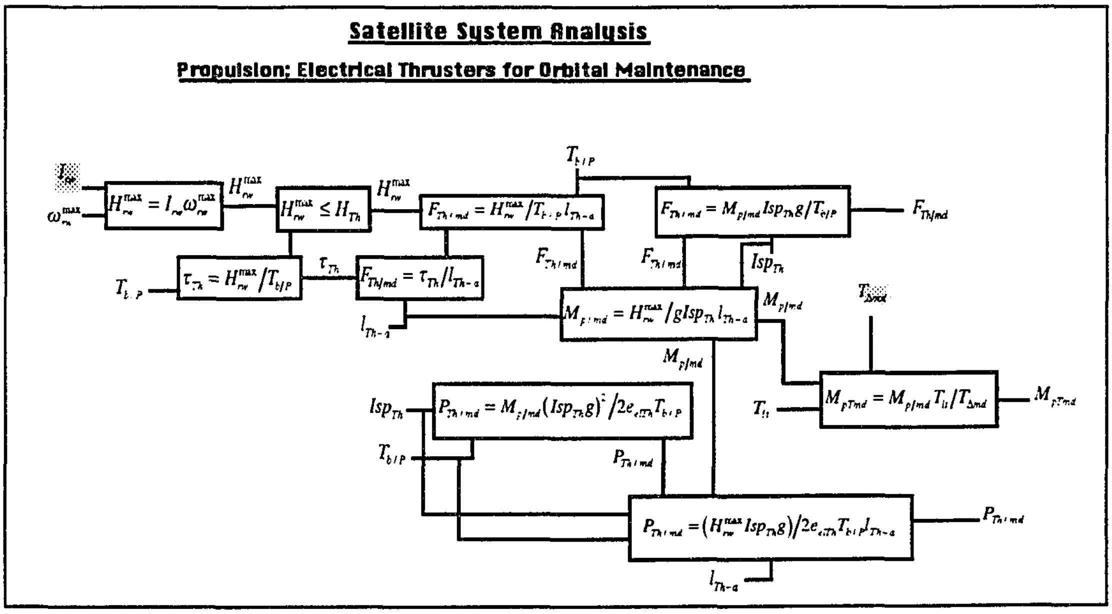


Figure 2-16 Momentum Dumping by Electrical Thrusters; Thrust, Propellant Mass, and Power

2.6. Power System

2.6.1. Solar Array Power and Size

$$P_{sa} = [(P_{ec}T_{ec}^{\max}/e_{SBS}) + (P_d(T - T_{ec}^{\max})/e_{SS})]/(T - T_{ec}^{\max})$$ **173**

$$P_o = P_s e_S$$ **174,175**

[168] Ibid., 592.
[169] Author's notes 11 March 1995, 189, and 27 March 1995, 200-202.
[170] Agrawal, B. N., 176.
[171] Author's notes 27 March 1995, 200-202.
[172] Ibid.
[173] McDermott, J.K., Power, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*. London, Kluwer Academic Publishers, 1991, 357, 367 & Table 11-17, 368.
[174] Ibid.
[175] Ibid., Table 11.31, 354.

$P_{BOL} = P_o I_d \cos\eta$ **176**

$e_{EOL} = 1 - \left(\Delta e_{s/y} T_{lt}\right)$ **177**

$P_{EOL} = P_{BOL} e_{EOL}$ **178**

$A_{sa} = P_{sa} / \left(P_{EOL} q_{cs}\right)$ **179,180**

$N_{CS} = \left(A_{sa} / A_{cs}\right) q_{cs}$ **181**

$M_{SA} = \chi_A * A_{sa}$ **182**

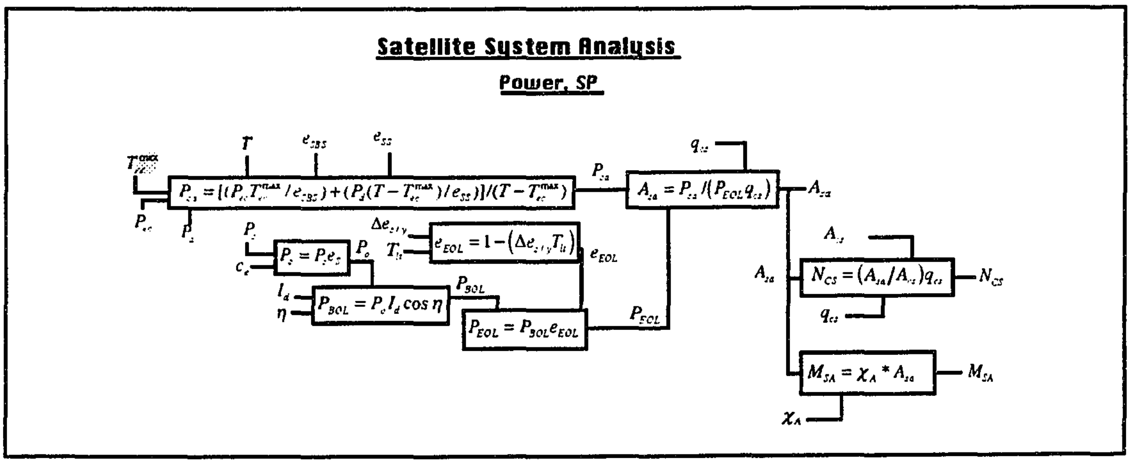


Figure 2-17 Sizing the Solar Array

2.6.2. Battery

$C_B = P_{ec} T_{ec}^{\max} / DoD_l EP_{avg}^B e_{BS}$ **183**

$E_{BAT} = C_B * EP_{avg}^B$ **184**

---

[176] McDermott, K., 358.
[177] Ibid.
[178] Ibid.
[179] Ibid.
[180] Griffin, M. D., 419-420.
[181] Ibid.
[182] Telecon with representative for AEC - Able Engineering, 07 August 1995; Author's notes 07 August 1995, 277-278.
[183] McDermott, K., 364.

$$M_{BAT} = E_{BAT} * \chi_E$$ [185,186]

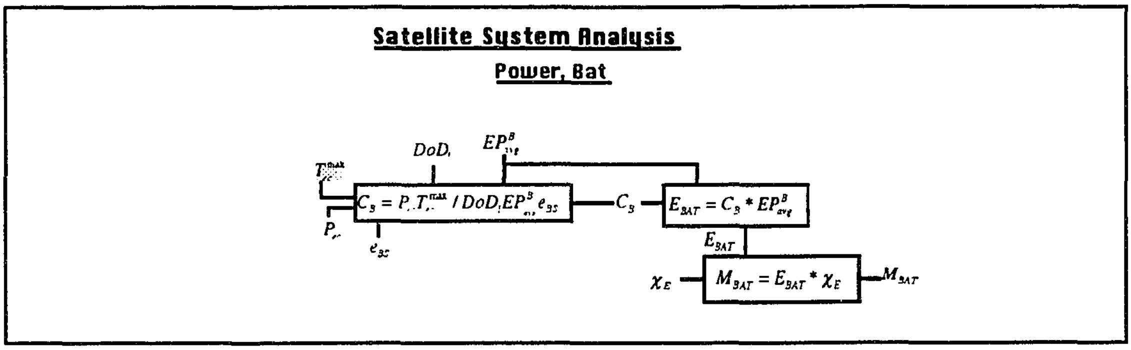


Figure 2-18 Battery Capacity

---

[184] Ibid., 362.
[185] Author's notes 24 January 1995, 141.
[186] Ibid.

Appendix B: NOTATION

LATIN

Lower Case

a: Semi-Major Axis of Transfer Ellipse, in *km*.

a_c: Semi-Major Axis of Transfer Ellipse for Reboost Correction, in *km*.

b; Number of bits per Sample.

c^P; Planck's Constant: $6.626*10^{-34} Js$.

c^B; Boltzman's Constant: $1.381*10^{-23} JK^{-1}$.

d_{eff}: Effective Diameter or Width of Image Pixel Projected on the Ground, or Sensor Resolution, in *m*.

d_d: Diameter or Width of Detector, in *m*.

d_λ: Diameter or Width of the Diffraction Limited Pixel Projected on the Ground, in *m*.

d_{pix}: Diameter or Width of the Pixel Projected on the Ground as Defined by the Sensor's Optics (Geometric Limited), in *m*.

e_{BS}: Transmission Efficiency between Battery and Consuming System, in %.

e_{EOL}: Remaining SA Efficiency at EOL, in W/m^2.

e_{elTh}: Efficiency for the Electric Propulsion System, defined as the ratio of Kinetic Energy generated to the Input Energy.

e_s: Solar Cell Efficiency: Silicon≈18%, Gallium Arsenide≈21%, in %.

e_{SF}: Sensor Frame Efficiency, Fraction of Time for Data Transmission, typically in the range of 0.9-0.95, in %.

e_{SBS}; Efficiency in Path from Solar Array via Battery, to Consuming System, in %.

e_{SS}; Efficiency in Path Directly from Solar Array to Consuming System, in %.

e_T; Transmitter Efficiency, in the Range of 35-45%, for both TWT (traveling wave tube) and SS (solid state) transmitters, in %.

$\Delta e_{s/y}$; Annual Degradation of Solar Cells, in %.

f; Sensor Focal Length, in m.

δf_n; Electrical Bandwidth of Sensor, in Hz.

g; Gravitational Acceleration: 9.8 m/s^2.

h; SC Altitude, in m.

h_{sea}; Earth Position's Altitude above Sea Level, in m.

h_{ES}; Distance from Earth to the Sun, in km.

$\Delta h_{AD/min}$; Reduction in Satellite Altitude, as a Function of Atmospheric Density, in m/min.

Δh_c; Allowed decrease in Satellite Orbit between Reboosts, in km.

i; Orbital Inclination.

j; Time Units after Launch Corrected for Launch Position's Latitudinal Distance from Equator.

k; Time Units after Launch.

l_{Th-a}; Distance from SC Principal Axes to Thrusters, along all axes, x,y,z, in m.

m_{cb}; Correction Mass added to SC end Mass, M, for Calculating the Average Time Between Reboosts. m_{cb} should be set as for example $0.5M_{bTp}$, based on the first Iteration for Calculating Reboosting Propellant, in kg.

m_{mb}; Propellant Margin to Remain in Propellant Tanks after All Reboosts are Completed, in kg.

n_{bg}; Number of SC Orbital Descent and Ascent Cycles, shown on Orbital Altitude Graph.

$p_{\%}$; Percentage of T_{mi} at each End of a Swath used for Acceleration and Deceleration, in %.

q_{cs};	Solar Cell Packing Density, for well Designed Arrays density may be 90%, in %.

r;	R_O+h.

r_l;	r-Δh_c.

$r_{tp}=R_0$; Radius of Transfer Ellipse, at Perigee, in *km*.

$r_{ta}=r$;	Radius of Transfer Ellipse, at Apogee, in *km*.

r_{cp};	Distance from Center of Pressure and Center of Mass, along all axes, x,y,z,, in *m*.

r_{rw};	Radius of Reaction Wheel, in *m*.

r_{mi};	Radius of Scanning Mirror, in *m*.

s;	Number of Samples per Pixel, typically values between 1.4 and 1.8, which are > 1 time Constant, are being Used.

t;	The Absolute Temperature of a Black Body Emitting Electromagnetic Radiation at Given Wavelength as defined in Planck's Equation. For IR Systems, t=300*K*, and for Visual Systems t=6000*K*.

t_s;	Communication System Noise Temperature, in *K*.

x_{mi};	Thickness of Scanning Mirror, in *m*.

y_{mi};	Width of Scanning Mirror, Perpendicular to the Scan Vector, in *m*.

y_{CS};	Width of Solar Cell, in *m*.

y_{SA};	Width of Solar Array(s), in *m*.

z_{CS};	Length of Solar Cell, in *m*.

z_{SA};	Length of Solar Array(s), in *m*.

Upper Case

Az;	The Orbital Azimuth Angle is measured from true North, to the SC Launch Vector, in *rad.*

A_d;	Area of Detector , in m^2.

A_{SC};	Cross Sectional Area of SC, in m^2.

A_{sa}; Required Area for SA, in m^2.

A_{cs}; Area per Solar Cell, typically $0.02x0.04m^2$, in m^2.

AD_e; Angular Displacement of SC per Orbital Period at Equator, in *km*.

AD_{La}; Angular Displacement of SC per Orbital Period, at a given La, in *km*.

C_B; Battery Capacity, in *Ahr*.

C_D; Drag Coefficient for SC: Approximately 2.2.

D; Real Sensor Aperture Diameter, in *m*.

D_{dnl}; Sensor Aperture Diameter (Detector Noise Limited System), in *m*.

D_{pnl}; Sensor Aperture Diameter (Photon Noise Limited System), in *m*.

D^*; A Detectivity Figure of Merit, Unique and given for Every Detector Type, in $cmHz^{(1/2)}W^{-1}$.

D_{ta}; Aperture Diameter of Transmitting Antenna, in *m*.

D_{ra}; Aperture Diameter of Receiving Antenna, in *m*.

DR_s; Data Rate Generated by Sensor, in bits per *sec*.

DR_{HK}; Data Rate Generated by House Keeping Systems, in bits per *sec*.

DR_c^s; Required Dump Rate from SC to GS, to Unload Stored Sensor Data when Passing over the GS, *Mb/s*.

DR_c^{HK}; Required Dump Rate from SC to GS, to Unload Stored House Keeping Data when Passing over the GS, *Mb/s*.

DR_c; Total Required Dump Rate from SC to GS ($DR_c^s + DR_c^{HK}$), *Mb/s*.

DS_s; On-Board Data Storage Capacity Required for Sensor Data, in *Mb*.

DS_{HK}; On-Board Data Storage Capacity Required for House Keeping Data, in *Mb*.

DS_F^{SS}; The Fixed (Base) Storage Capacity for a Solid State Recorder, in *Mb*.

DS_{inc}^{SS}; Incremental Storage Capacity for a Solid State Recorder, in *Mb*.

DS; Total Required On-Board Data Storage Capacity (DS_s+DS_s), in *Mb*.

DoD_l; Limit on Battery Depth of Charge, in %.

E_b; Received Communication Energy per Bit, E_b/N_0, should be between 5 and 10.

E_{BAT}: Energy Generated by SC Battery, in *Whr.*

EP_{avg}^{B}: Battery's Average Voltage, in *V.*

$F^{\#}$: F-number; F-Stop, in Integer Values.

F_{em}: Electromagnetic Flux from Target, from Planck's Equation, in W/m^2.

F_{AD}: Force on SC due to Atmospheric Drag, in *N.*

$F_{Th/md}$: Thruster Force Required to Create a Angular Momentum (*H*) Equal to the Angular Momentum Created by the Reaction Wheel since last Momentum Dumping, in *N.*

F_{E}^{Th}: Thrust Force Generated by Orbit Re-boosting Engine during Re-boosts, in N.

FOV_s: Suggested Sensor Field of View (FOV), in °.

FOV; Actual Sensor FOV, in °.

G_r: Gain of Receiving Antenna: $G_r=\pi^2 D_{ra}^{\ 2}\varpi_{ra}/\lambda_c^{\ 2}$.

G_t: Transmitting Antenna Gain: $G_t=\pi^2 D_{ta}^{\ 2}\varpi_{td}/\lambda_c^{\ 2}$.

ΔGS; Maximum Distance Between Ground Stations, in *km.*

$H_{AD/O}$: Angular Momentum Generated by SC per Orbit, due to the Torque created by the Atmospheric Drag, in *Nms.*

H_{rw}^{max}: Maximum Angular Momentum that can be generated by the Reaction Wheel between Momentum Dumpings, in *Nms.*

Isp: Specific Impulse for Launch Vehicle, in *sec.*

Isp_{Th}: Specific Impulse of SC Thrusters, in *sec.*

Isp_b: Specific Impulse of SC Thrusters used for Reboosting, in *sec.*

I_d: Inherent Degradation of Solar Array, in %.

I_{SC}: Moment of Inertia for SC along in the x,y, and z Directions, including MOI for SC body and Solar Arrays, in $kg*m^2$.

I_{mi}: Moment of Inertia for Scanning Mirror, Parallel to the Scan Vector, in $kg*m^2$.

I_{rw}: Moment of Inertia Produced by Reaction Wheel, in $kg*m^2$.

J_2: Gravitational Field Constant: $1.082*10^{-3}$.

$K_{\%}$: Compression of DR_s due to Data Processing, in %.

La_p; Latitude of SC's Launch Position, in ^oN.

La_f; Latitude of SC after k time units, in ^oN.

La_e; Latitude of SC at Equator, $La_e=0$, in ^oN.

L_a; Transmission Path Loss in Communication System.

L_l; Transmitter to Antenna Line Loss.

L_s; Space Loss.

Lo_e; Longitude Equivalent at Equator, in ^oE.

Lo_f; Longitude k time units after launch, in ^oE.

Lo_p; Longitude of Launch Position, in ^oE.

ΔLo; Change in Longitude k time units after Launch, in *rad*.

M; Mass of SC, in *kg*.

M_{dry}; Estimated Dry Mass of SC, $M- (M_{pTb}+ M_{pTmd})$, in *kg*.

M_{tbd}; Mass of Propellant Consumed by SC, $M_{pTb}+ M_{pTmd}$, in *kg*.

M_p; Propellant Mass Required to Achieve Orbit, in *kg*.

M_{pi}; Propellant Mass Required per Reboost, in *kg*.

M_{pi}^{max}; Maximum Propellant Mass Consumed by Any One Re-boosting Maneuver, in *kg*.

M_{pTb}; Total Propellant Mass required for all On-orbit Reboosts of the SC, in *kg*.

M_{rbpt}; Mass of Propellant Tank(s) Holding Propellant used for On-orbit Reboosting, in *kg*.

M_{rbpm}; Mass of the On-orbit Reboosting Propellant Management System, in *kg*.

M_{rbE}; Mass of the On-orbit Reboosting Engine, in *kg*.

M_{rbps}; Mass of the Whole On-orbit Reboosting Propulsion System, including, M_{rbpt}, M_{rbpm}, and M_{rbE}.

M_{pImd}; Propellant Mass Required per Momentum Dump, in *kg*.

M_{pTmd}; Total Propellant Mass required for all on-orbit Momentum Dumping, in *kg*.

M_{mdpt}; Mass of Propellant Tank(s) Holding Propellant used for Momentum Dumping, in *kg*.

M_{mdpm}; Mass of the Momentum Dumping Propellant Management System, in *kg*.

M_{mdT}; Mass of the Momentum Dumping Thruster, one per Axis, in *kg*.

M_{mdps}; Mass of the Whole Momentum Dumping Propulsion System, including, M_{mdpt}, M_{mdpm}, and $3 * M_{mdT}$.

M_{rw}; Required Mass for Reaction Wheels on Axes, to Compensate for Torque Produced by the Atmospheric Drag, in kg.

M_{rwe}; Mass of Reaction Wheel Drive Electronics, in *kg*.

M_{rws}; Mass of the Reaction Wheel System includes the Mass of Three Reaction Wheels, M_{rw}, (one for each axis), and the Mass of One Wheel Drive Electronics' unit, M_{rwe}, in *kg*.

M_{st}; Mass of Star Tracker/Scanner, in *kg*.

M_{es}; Mass of Earth Sensor/Tracker, in *kg*.

M_{gs}; Mass of Gyro Scope, in *kg*.

M_{PROP}; Mass of Propulsion System, in *kg*.

M_{GNC}; Mass of GN&C System, in *kg*.

M_{SENS}; Mass of Sensor System, in *kg*.

M_{mi}; Mass of SC Sensor Scanning Mirror, in *kg*.

M_{cc}; Mass of SC Sensor Cryogenic Cooler, in *kg*

M_{o}; Mass of SC Sensor Optics, in *kg*.

$M_{P\&S}$; Mass of DP&DS System, in *kg*.

M_{CPU}; Mass of DP System, in *kg*.

M_{DS}; Mass of DS System (SS, or T), in *kg*.

$M_{DS}{}^{SS}$; Mass of Solid State Recorder, in *kg*.

$M_{DS}{}^{T}$; Mass of Tape Recorder, in *kg*.

M_{COM}; Mass of Communication System, in *kg*.

$M_{ANT}{}^{m}$; Mass of Main Antenna (Parabola), in *kg*.

M_{FIL}; Mass of Communication System Filter, in *kg*.

M_{TR}; Mass of Transmitter and Receiver, includes two units for Redundancy, in *kg*.

M_{ANT}: Mass of Antenna System, in *kg*.

M_{HEM}: Mass of Hemispheric Antenna, in *kg*.

M_{ANT}^{o}: Mass of Other Antenna System Components, such as Waveguide, and Turnstile, in *kg*.

M_{POW}: Mass of Power System, in *kg*.

M_{PCU}: Mass of the Power System Power Control Unit, *kg*.

M_{CR}: Mass of the Power System Converter and Control Unit, *kg*.

M_{PW}: Mass of Power-Wiring, in *kg*.

M_{BAT}: Mass of Battery, in *kg*.

M_{SA}: Mass of Solar Array, in *kg*.

M_F^{SS}: The Fixed (Base) Mass of a Solid State Recorder, in *kg*.

M_{inc}^{SS}: Mass, for a Solid State Recorder, of each Incremental Storage Capacity Unit, in *kg*.

N_p: Number of Photons Bombarding Sensor Detector per time Unit, in Photons per *sec*.

N_{pix}: The number of pixels a scanner scans across-track, per scan.

For a staring system N_{pix}=1.

N_{ch}: Number of Channels, or frequency bands in which data is being acquired.

$N^{s/sc}{}_{pix/ch}$: Number of Pixels, Scanned Simultaneously.

$N^{sim}{}_{pix/ch}$: Number of pixels observed simultaneously, by SC sensor, per channel, or band: For a staring system, $N^{sim}{}_{pix/ch}$ represents the total number sensor detector elements; For a scanner, $N^{sim}{}_{pix/ch}$ represents the number of pixels scanned simultaneously ($N^{s/sc}{}_{pix/ch}$).

N_s^i: Number of Samples Generated per *sec* by House Keeping System *i*. in *(Samples/Sec) Hz*.

N_s^s: Number of Samples Generated per *sec* by Sensor, in *(samples/sec)Hz*.

$N_{b/s}^i$: Number of Bits per Sample Generated by House Keeping System *i*, in *bits/sample*.

$N_{i/s}^i$: Number of Instructions Required to Process Each Sample Generated by House Keeping System *i*, in *IP/sample*.

$N_{i/s}^{s}$; Number of Instructions Required to Process Each Sample Generated by Sensor. in *IP/sample.*

N_{TIPS}^{HK}; Total Number of Instructions Required per *sec* for Processing of House Keeping Data. in *KIPS.*

N_{IPS}^{s}; Number of Instructions Required per *sec* for Processing of Sensor Data, assuming On-Board Processing, in *KIPS.*

N_{TIPS}; Total Number of Instructions that Need to be Handled by the SC CPU per *sec,* in *KIPS.*

N_{O}; Communication System Noise Density, E_b/N_O, should be between 5 and 10.

NEP; Detector Noise Equivalent Power, the Value of the Signal Power (P_D) when it Equals The Noise Power, in *W.*

N_{SA}; Number of Solar Arrays Attached to the SC body.

N_{CS}; Total Number of Solar Cells Required to Power SC.

N_{GS}; Number of Ground Stations Along SC Track.

N_{rb}; Number of Orbital Descent and Ascent Cycles the SC goes through during its Life Time (T_{lt}).

P_{BOL}; Real Power Output from Solar Array at Beginning of Life (BOL), in *W/m²*.

P_{CPU}; Power Required by CPU, in *W.*

P_{DS}; Power Required for the DS Device (SS, or T), in *W.*

P_{DS}^{T}; Power Required for a Tape Recorder, in *W.*

P_{DS}^{SS}; Power Required for a Solid State Recorder, in *W.*

P_{D}; Power Generated in Detector, in *W.*

P_{EOL}; Real SA Power Output at EOL, in *W.*

P_{F}^{SS}; The Fixed (Base) Power Level for a Solid State Recorder, in *W.*

P_{inc}^{SS}; Power Required, for a Solid State Recorder, per Incremental Storage Capacity Unit. in *W.*

P_{ec}; Power Required During Eclipse, in *W.*

P_d; Power Required During Sun Light, in W.

P_o; Optimal Power Output by Solar Array, in W/m^2.

P_s; Incident Solar Radiation Power, in W/m^2.

P_{sa}; Power to be Generated by Solar Array during Sun Light, to Power Space Craft through the full Orbit, in W.

$P_{Th/md}$; Electrical Power to Fire El. Thrusters used for Momentum Dumping, in W.

P_{GNC}; Power Required by GN&C System, in W.

P_{rw}; Power Required to Produce Sufficient Torque to Counter the Torque Produced on the SC through Atmospheric drag, in W.

P_{SENS}; Power Required by Sensor System, in W.

P_{mi}; Power Required by Scanner's Oscillating Mirror, in W.

P_{cc}; Power Required by SC Sensor Cryogenic Cooler, in W.

$P_{P\&S}$; Power Required by DP&DS System, in W.

P_{COM}; Power Required by Communication System, in W.

P_{COM}^T; Power required by Transmitter to Transmit at DR_c, in W.

P_{COM}^R; Power required by Receiver, in W.

Q; Number of SC Orbits per One Sidereal Day.

R_E; Earth Radius, in km.

R_S; Radius of the Sun, in km.

R_P; Earth Radius (R_E) plus Altitude above Sea Level of Observed Position, in km.

S_w; Swath Width, at h km, in km.

S_i; Effective Sensor Swath Width at h km, in km.

S/N_{dnl}; The Ratio between Signal Strength and Noise Strength for a Detector Noise Limited System, here measured in W.

S/N_{pnl}; The Ratio between Signal Strength and Noise Strength for a Photon Noise Limited System ,here measured in $(N_p)^{(1/2)}$

S/N_s; SC Sensor S/N Ratio.

S/N_s^e: Effective SC Sensor S/N Ratio.

T: Orbital Period of SC, in *sec*.

T_E: Orbital Period of Earth, in Sidereal Time, in sec.

T_{ep}: Time from Equator to Launch Position, in *sec*.

T_{ES}: Orbital Period of Earth around Sun.

T_c: Time Available for Communication , in *min*.

T_{ca}: Time to set up Communication, set at *0.5 min*, in *min*.

T_{cb}: Buffer Time at the end of the Communication Segment, set at *0.5 min*, in *min*.

T_{co}: Orbital period for the Reboosting Transfer Orbit, in *sec*.

T_{Hc}: Time SC is in the Hohman Reboosting Transfer Orbit, in *hours*.

$T_{\Delta h}$: Time between required Reboosts, in *days*.

T_{bg}: Time interval shown on the SC altitude graph. Graph includes reboost corrections, in *days*.

$T_{\Delta md}$: Time between required Momentum Dumps, in *sec*.

$T_{\Delta GS}$: Time between SC passes over Ground Stations, in *sec*.

T_d: Dwell Time, the time it takes for an Image to Move Through a Pixel, in *sec*.

T_i: Integration time, detector data acquisition time, $T_i = \gamma T_d$, in *sec*.

T_{ec}^{max}: Maximum Eclipse Duration, in *min*.

T_{ec}^{min}: Minimum Eclipse Duration, in *min*.

T_{ec}^{eff}: Effective Eclipse Duration, in *min*.

T_{mi}: Time for Scanning Mirror to Travel the Full Scan Angle, in *sec*.

T_{lt}: Life Time of SC, in *years*.

$T_{b/P}$: Burn Time per Thruster Pulse, in *sec*.

T_E^{bt}: Burn Time or Pulse for the Re-boosting Engine, in *sec*.

V: Velocity of SC in Orbit, in *m/s*.

V_E: Velocity of Earth at the Latitude of a given Launch Site, and for a given *Az*. in *m/s*.

V_{tp}: SC Velocity at Perigee of the Transfer Ellipse, in *m/s*.

V_{ta}; SC Velocity at Apogee of the Transfer Ellipse, in *m/s*.

ΔV_{Ec}; ΔV Required to increase SC Velocity from V_E to V_{tp}, in *m/s*.

ΔV_{eo}; ΔV Required to increase SC Velocity from V_{ta} to V, to "Circularize" the transfer Ellipse, in *m/s*.

ΔV; ΔV Required to go from Earth Launch Site to a Defined Orbital Altitude, Assuming a Hohman Transfer Orbit, in *m/s*.

V_l; Velocity of SC at the lowest Orbit between Reboosts, r_l, in *m/s*.

V_{cp}; SC Velocity at Perigee of the Reboost Transfer Ellipse, in *m/s*.

V_{ca}; SC Velocity at Apogee of the Reboost Transfer Ellipse, in *m/s*.

ΔV_{lc}; ΔV Required to go increase SC Velocity from V_l to V_{cp}, in *m/s*.

ΔV_{co}; ΔV Required to increase SC Velocity from V_{ca} to V, to "Circularize" the Reboost transfer Ellipse, in *m/s*.

ΔV_c; ΔV Required to Reboost SC from r_l to r, using a Hohman Transfer Orbit, in *m/s*.

$\Delta V_{AD/O}$; Velocity Change per Orbit due to Atmospheric Drag, in *m/s*.

GREEK

Lower Case

α_h; Central Angle to Horizon, at *h km*, in *rad*.

α_c; Central Angle to Communication Horizon, in *rad*.

α_i; Central Angle of Sensor FOV.

β_h; Nadir Angle to Horizon.

β_c; Nadir Angle to Effective Communication Horizon, in *rad*.

β_i; Sensor Nadir Angle.

χ_E; The Inverse Specific Energy Density for a Battery, in *kg/Whr*.

χ_A; The Mass to Area Ratio for a Solar Array, in *kg/m²*.

δ_p; A digital constant that only can take values of *1* and *0*. δ_p is *0*. if there is no on-board processing, and *1*. if there is on-board processing.

ε_c; Nadir Angle reduction to avoid Communication problems, in °.

$\Delta\phi$: Angular Displacement of SC per Orbital Period, in *rad.*

γ; Degree of overlap between Pixels on the Ground. $\gamma=0$, indicates that no data is being recorded; $\gamma=1$, Indicates that there is no Overlap; $\gamma=0.5$, Indicates that there is a 50% overlap between Pixels. When the pixel radius is being used in the calculations of T_d. these values are 0, 2, and 1, respectively.

η; Solar Incidence Angle, Measured between the Vector Normal to the Surface of the Array, and the Sun.

$\Delta\varphi_{AD/O}$; Change in Pointing per Orbit due to Atmospheric Drag, along all Axes, in °.

$\Delta\lambda$; Sensor Bandwidth, in *m.*

λ; Sensor Wavelength, in *m.*

λ_c; Wavelength of Communication Down Link, in *m.*

μ; Geocentric Gravitational Constant: $3.986*10^{14}$ m^3/s^2.

$\Delta\theta$; Detector Plane Angle, also called IFOV (Instantaneous Field of View), in *rad.*

ρ; Atmospheric Density, in $kg*m^2$.

σ; Time unit.

τ_{AD}; Torque on SC as a function of Atmospheric Drag, and r_{cp}, in Nm.

τ_{Th}; Torque generated on SC, when the Momentum Dumping Thrusters are fired, in *Nm.*

υ_{mi}; Specific Density of Material in Scanning Mirror, in kg/m^3.

ϖ_{ra}; A Figure of Merit, between 0 and 1, for Receiving Communication Antenna, typical Value around 0.55. For Good Ground Antennas, values of 0.6-0.7 can be achieved.

ϖ_{ta}; A Figure of Merit, between 0 and 1, for Transmitting Communication Antenna, typical Value around 0.55.

ω; Angular Velocity of SC, in *rad/sec.*

ω_E; Angular Velocity of Earth, in *rad/sec.*

ω_{rw}^{max}; Maximum allowable Angular Velocity for Reaction Wheel, Specified by Manufacturer, in *rad/s.*

ξ_{rw}; Angular Acceleration of Reaction Wheel, in rad/s^2.

ξ_{mi}; Angular Acceleration of Scanning Mirror, in rad/s^2.

ψ; Constant that defines the Relationship Between δf_n and T_l. Values between o.5 and 3 are Used Depending on the Application.

ζ_{CPU}^{P}; Constant that defines the Proportional Relationship between N_{TIPS} and P_{CPU}, Values will vary with the Technology of the CPU being Applied, in *W/TIPS.*

ζ_{CPU}^{m}; Constant that defines the Proportional Relationship between N_{TIPS} and M_{CPU}, Values will vary with the Technology of the CPU being Applied, in *kg/TIPS.*

ζ_{COM}^{m}; Constant that defines the Proportional Relationship between P_{REC} and M_{TR}, Values are Dependent on the Type of Communication Band being Used, in *kg/W.*

ζ_{ANT}^{m}; Constant that defines the Proportional Relationship between D_{ta} and M_{ANT}^{m}, Values are Dependent on the Antenna Technology being Used. in kg/m.

ζ_{PCU}^{m}; Constant that defines the Proportional Relationship between P_d and M_{PCU}, in *W/kg.*

ζ_{CR}^{m}; Constant that defines the Proportional Relationship between P_d and M_{CR}, in *W/kg.*

ζ_{PW}^{m}; Constant that defines the Proportional Relationship between M_{dry} and M_{PW}, (Preferably dry mass of SC should be used in this calculation), in *kg/kg.*

ζ_{mpt}^{m}; Constant that Defines the Relationship between M_{pTmd} and M_{mdpt}, in *kg/kg.*

ζ_{mpm}^{m}; Constant that Defines the Relationship between M_{mdpt} and M_{mdpm}, in *kg/kg.*

ζ_{bpt}^{m}; Constant that Defines the Relationship between M_{pTb} and M_{rbpt}, in *kg/kg.*

ζ_{bpm}^{m}; Constant that Defines the Relationship between M_{rbpt} and M_{rbpm}, in *kg/kg.*

Upper Case

$\Gamma_c^{'}$; Angle Opposite Communication Nadir Angle.

$\Gamma^{'}$; Angle Opposite Sensor FOV (θ_s) Nadir Angle.

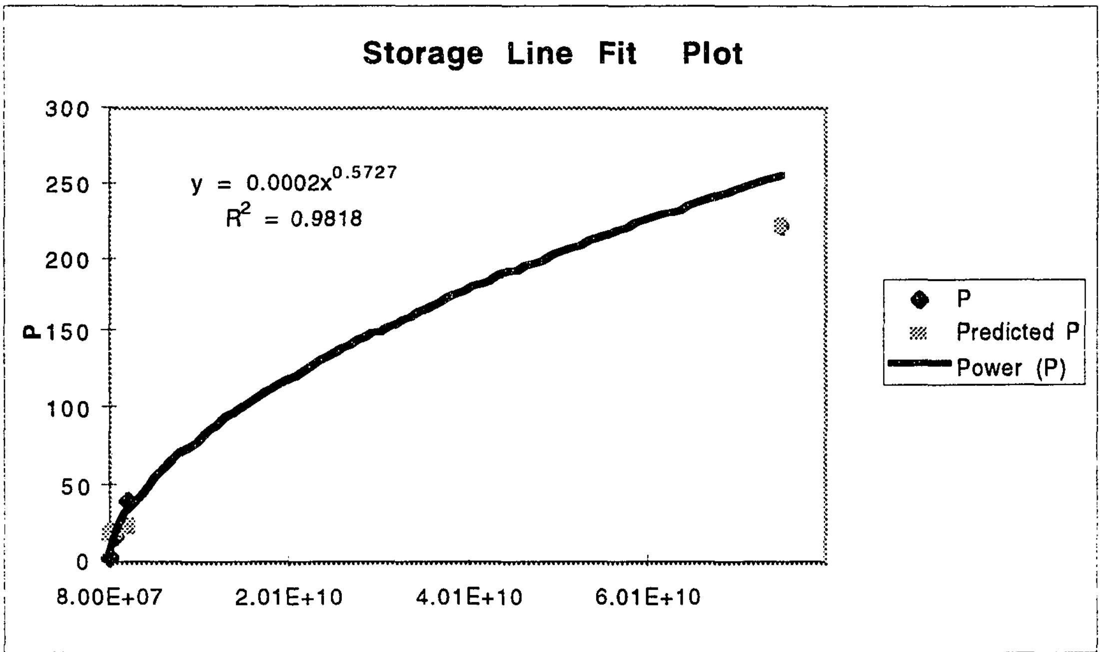
ϑ_T; Percentage time for Observations between Ground Stations, in %.

$\Delta\Omega_{ef}$; Regression of Nodes after k time units, in *rad.*

$\Delta\Omega_{ep}$; Regression of Nodes over T_{ep} *sec.*

Appendix C: EMPIRICALLY DERIVED EQUATIONS

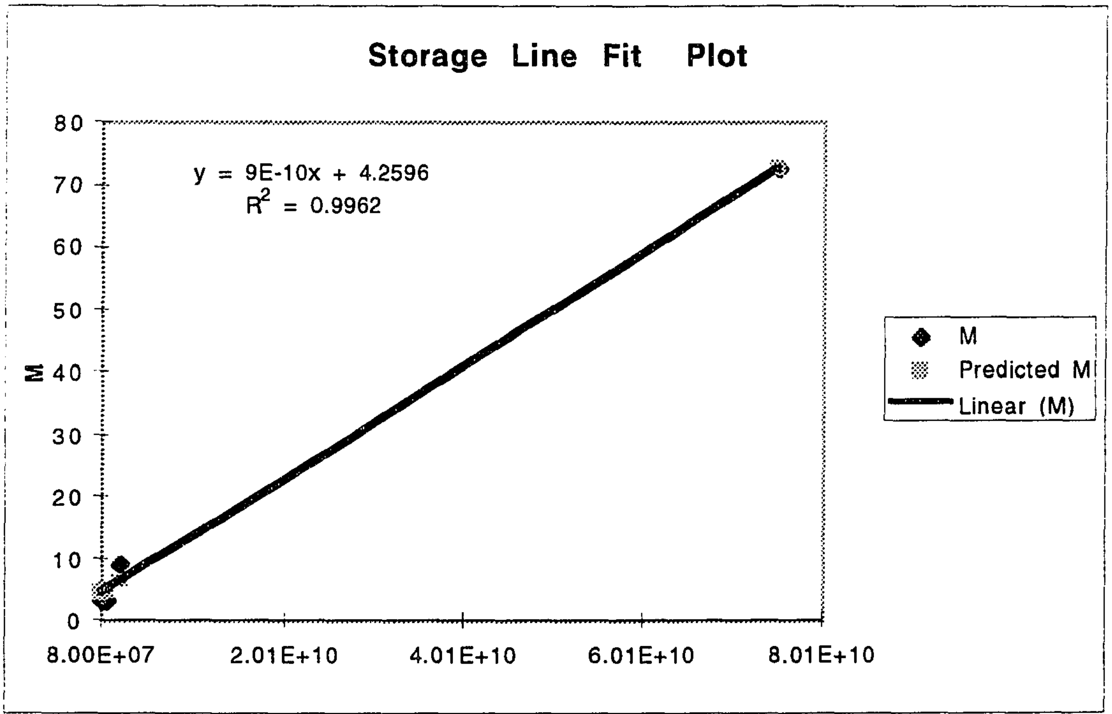
1. Data Storage Capacity vs. Power



Type	Storage	DR	P	M
4200	8.00E+07	0.512	4	2.95
STR 108	5.00E+08	2.56	17	3.18
DDS 5000	2.00E+09	3	40	9.07
DDS 6000	7.50E+10	100	220	72.6

Boatwright, J.E., Mueller, R.L., Command and Data Handling, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, Table 11.26, 348.

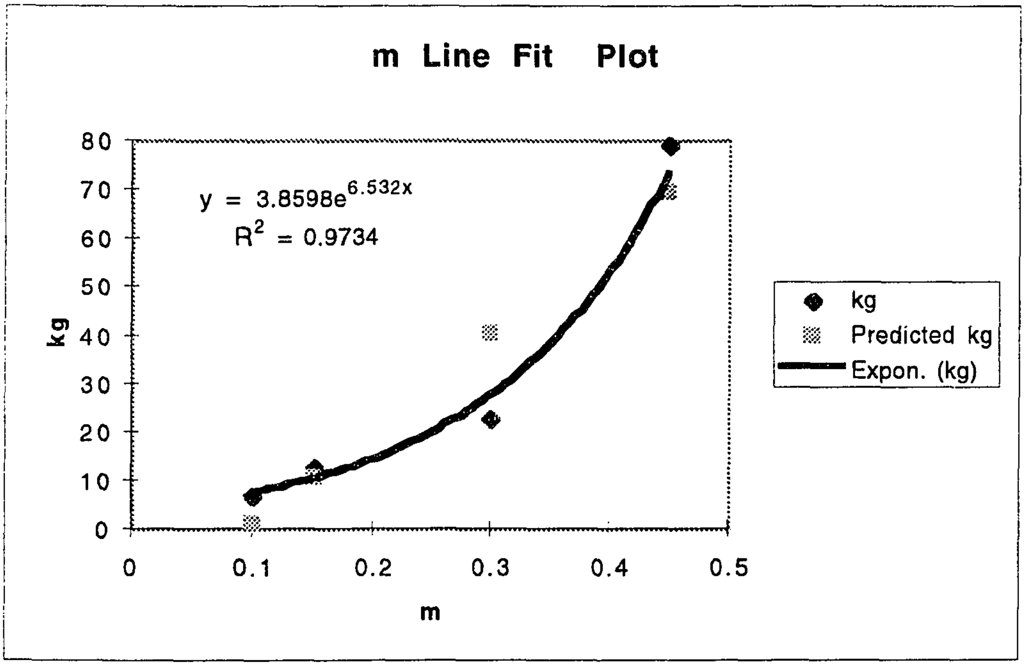
2. Data Storage Capacity vs. Mass



Type	Storage	DR	P	M
4200	8.00E+07	0.512	4	2.95
STR 108	5.00E+08	2.56	17	3.18
DDS 5000	2.00E+09	3	40	9.07
DDS 6000	7.50E+10	100	220	72.6

Boatwright, J.E., Table 11.26, 348.

3. Sensor Aperture Diameter vs. Mass



inch	m	kg
4	0.1	6.8
6	0.15	12.5
12	0.3	22.7
18	0.45	79

Telecon with F. Kingsley, Space Optics Research Lab. July 24, 1995.

Appendix D: DEFAULT VALUES FOR MODEL VARIABLES

Front Panel Input Variables

0.0 Rem.Sens.Sat.Analysis.vi

Q Value O/day: 15.5
Orb.Inclination,rad: 0.524
SC Life Time, y: 3
SC Mass kg: 200
SC Drag C: 2.2
Area SC m^2: 5

1.1 Two Body Motion.vi

1.3 Angular Displacement.vi

Lat of West AD °: 20

1.4. SC Horizon & Swath.vi

Altitude at Position m: 500

1.5 Communication Time.vi

Com.Corr.Ang. °*: 3

1.6 Eclipse Time.vi

2.0 SC Bus.vi

2.1 Propulsion.vi

2.1.1 Propulsion System.vi

Lai, N/S: 20-N
Isp s: 460

2.1.2 Prop for Reboost.vi

All.Orb.Dip, m: 3000
Mass Corr*, kg: 5
Isp b*: 302
rb Prop.Marg.,kg*: 3
rb cycl graph.:0
rb E Burn Time, s*: 2
Booster Eng. M,kg*: 1.2

2.2 Sensor.vi

Scan/Stare: stare

2.2.1 Apert.Diam;pnl;dnl.vi

Sensor Wavel., m*: 5.00E-7
Sens.Foc.Length, m: 0.9
D* *: 1E+9
Sens.Det.C/S: C
Sens.Det.Diam, m*: 1.50E-5
Sens.Bandw., m: 2.00E-7
Pix.Eff., m: 8
Bits per Smpl.*: 12
Const.dfn/Ti*: 1.57
Abs.Temp, K*: 6000
Pix.Overl.,%*: 45

2.2.2 Scanner Power.vi

% Tmi, a&d*: 5
Mirr.Rad.,m: 0.5
Mirr.Thick.,m: 0.01
Mirr.Breadth.,m: 0.1
Mirr.Dens..kg/m^3: 1.85E+3

2.2.3 Sensor H&S.vi

Sugg.FOV, deg: 0.5
Altitude at Position m: 500

2.2.4 Sens. DR.vi

No.Pix.Sc.Sim.*: 5
Smpl.perPix*: 1.4
Push Br./F.Stare*: PB
#Channels: 1
Sens.Fr.Eff.,%*: 95

2.3 DP & DS.vi

2.3.1 Data Pro.Instr.Smpl.vi

2.3.1 Data Pro.Smpl.Sec.vi

2.3.1 Data Processing.vi

Sens.Instr.Smpl.: 50
CPU Type*: SQ-AOSP
Sens.Proc.Y-N: N
Data Comp.,%*: 50

2.3.2 Data Sto.Bits.Smpl.vi

2.3.2 Data Storage.vi

GS: 3
Obs.betw.GS,%: 90
Storage Medium: Tape
SS Fix.P, W*: 3
SS inc.S,bits*: 6.400E+7
SS inc.P,W*: 4.376E-1
SS Fix.S.,bits*: 1.280E+8
SS Fix.M, kg*: 6.170E+0
SS inc.M,kg*: 9.000E-1

2.4 Communication.vi

Com.Wavel.,m*: 1.35E-1
T.Ant.Diam.,m*: 0.7

2.4.1 Comm. DR.vi

Set Up T,min*: 0.5
Buffer T,min*: 0.5

2.4.2 Comm Power.R.Gain.vi

R.Ant.Diam.,m*: 14.00
R.Ant.Eff*: 0.6

2.4.2 Comm Power.T.Gain.vi

T.Ant.Eff*: 0.55

2.4.2 Comm Power.vi

Rec.Com.En./bit*: 100
Com.Syst.n.Dens.*: 10
Com.Syst.Loss,%: 1
Space Loss,%: Now a Calculation
Tr.toAnt.L.L.,%: 1
Transm.Eff.,%*: 35

2.4.3 Comm.Mass.vi

Hemis. M,kg*: 0.25
Other Ant. M,kg*: 1.5
Antenna Type: Parabola Fixed-S(1.7)

2.5 GN&C.vi

Star Tr. M,kg*: 7.7
Star Tr. P,W*: 18
Earth Sens. M,kg*: 2.5
Earth Sens. P,W*: 8
Gyro M,kg*: 3
Gyro P,W*: 15

2.5.1 Atmosph.Drag Eff.vi

SC MOI,kg*m^2: 300
Dist.CP&CM.,m: 0.5

2.5.2 Sizing the RW.vi

Ang.Acc.RW, rad/sec^2: 8.70E-3
RW Rad.,m*: 0.3
RW El.Mass.,kg*: 2.2
RW max ang.Vel.,rad/sec*: 6.280E+2

2.5.3. Mom. Dump.vi

El vs. Chemical*: Chem.
Burn T per Th.Pulse,s*: 0.5
MD Th.Spec.Imp.,s*: 250
Dist.Th.to Prin.Ax.,m: 1
El.Prop.Eff.*: 0.9
Thruster M,kg*: 1.2

2.6 Power.vi

2.6.1 Solar Array, P&A.vi

Solar Cell Type: GaAs
P Regulation: PPT
Sol.Inc.Angle,°: 23.5
SC Sol.Arrays: 2
Inh.Sol.Cell.Degr.,%*: 77

Sol.Arr.Width.,m^2: 0.4
Sol.Cell Pack.Dens.,%*: 90
Sol.Cell.Width,m*: 0.02
Sol.Cell.Length.,m*: 0.04
SA Type: Substr.Foldo/Adv.Rollo

2.6.2 Battery.vi

Battery Class: NiH2ipv
Transm.Eff.,%*: 90
Batt.Avg.Chrg.,V*: 26

Appendix E: THE LABVIEW MODEL

Illustrations of front panels and (block) diagrams from the LabVIEW model developed for this research:

0.0 Rem.Sens.Sat.Analysis.vi [front panel]

2.1 Propulsion.vi [diagram]

2.2 Sensor.vi [diagram]

2.3 DP & DS.vi [diagram]

2.4 Communication.vi [diagram]

2.5 GN&C.vi [diagram]

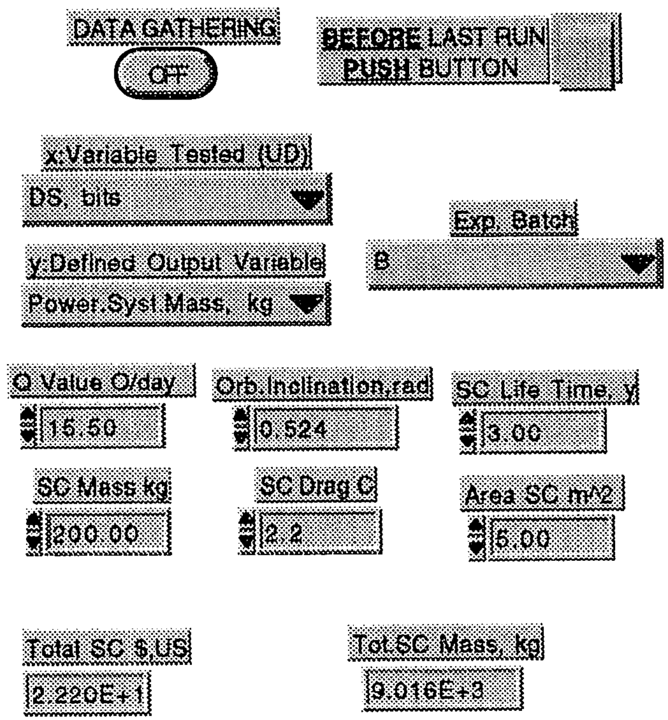
2.6 Power.vi [front panel and diagram]

2.6.2 Battery.vi [diagram]

"HELP" text from 2.6.2 Battery.vi

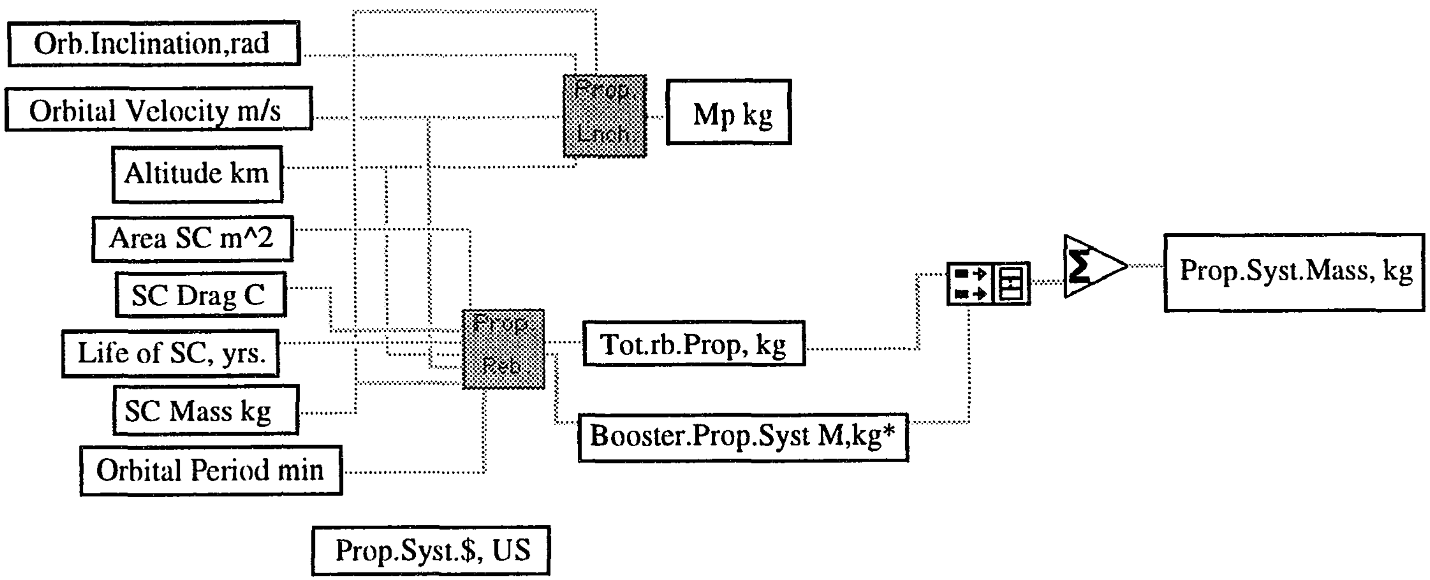
0.0 Rem.Sens.Sat.Analysis.vi

Front Panel (Controls & Indicators)



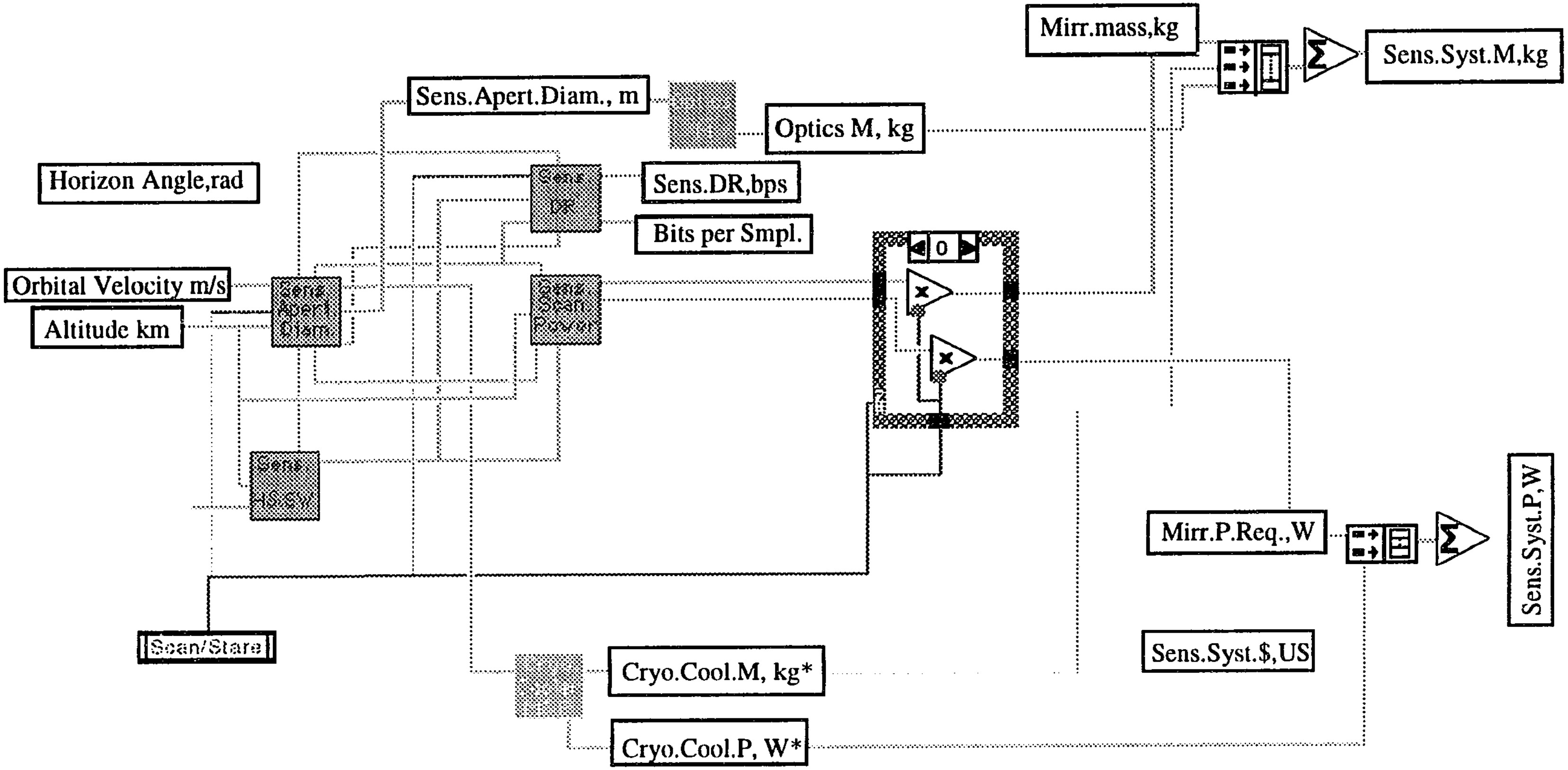
2.1 Propulsion.vi

Diagram



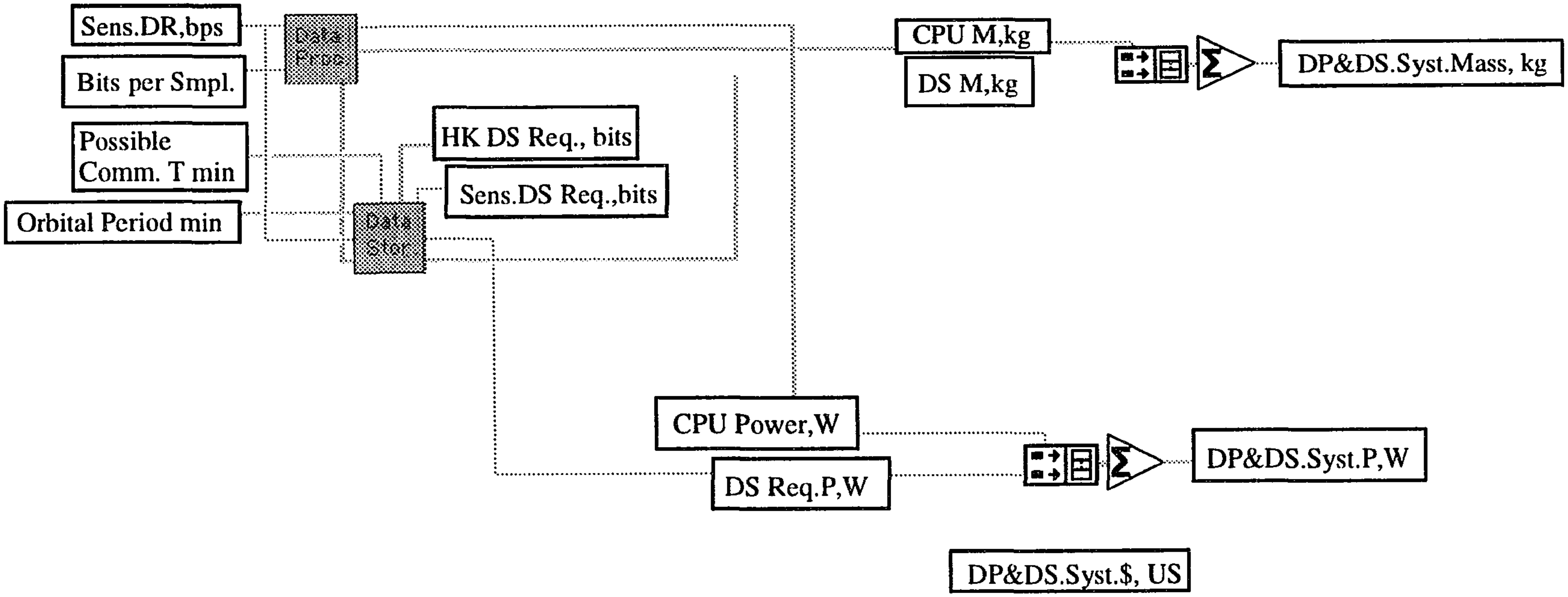
2.2 Sensor.vi

Diagram



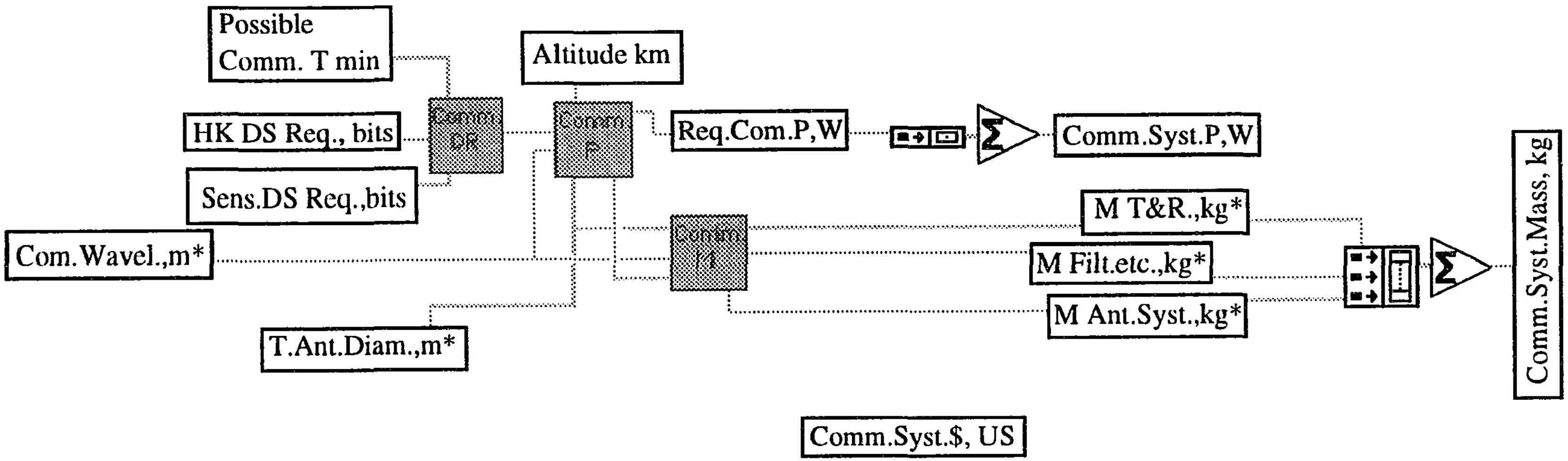
2.3 DP & DS.vi

Diagram

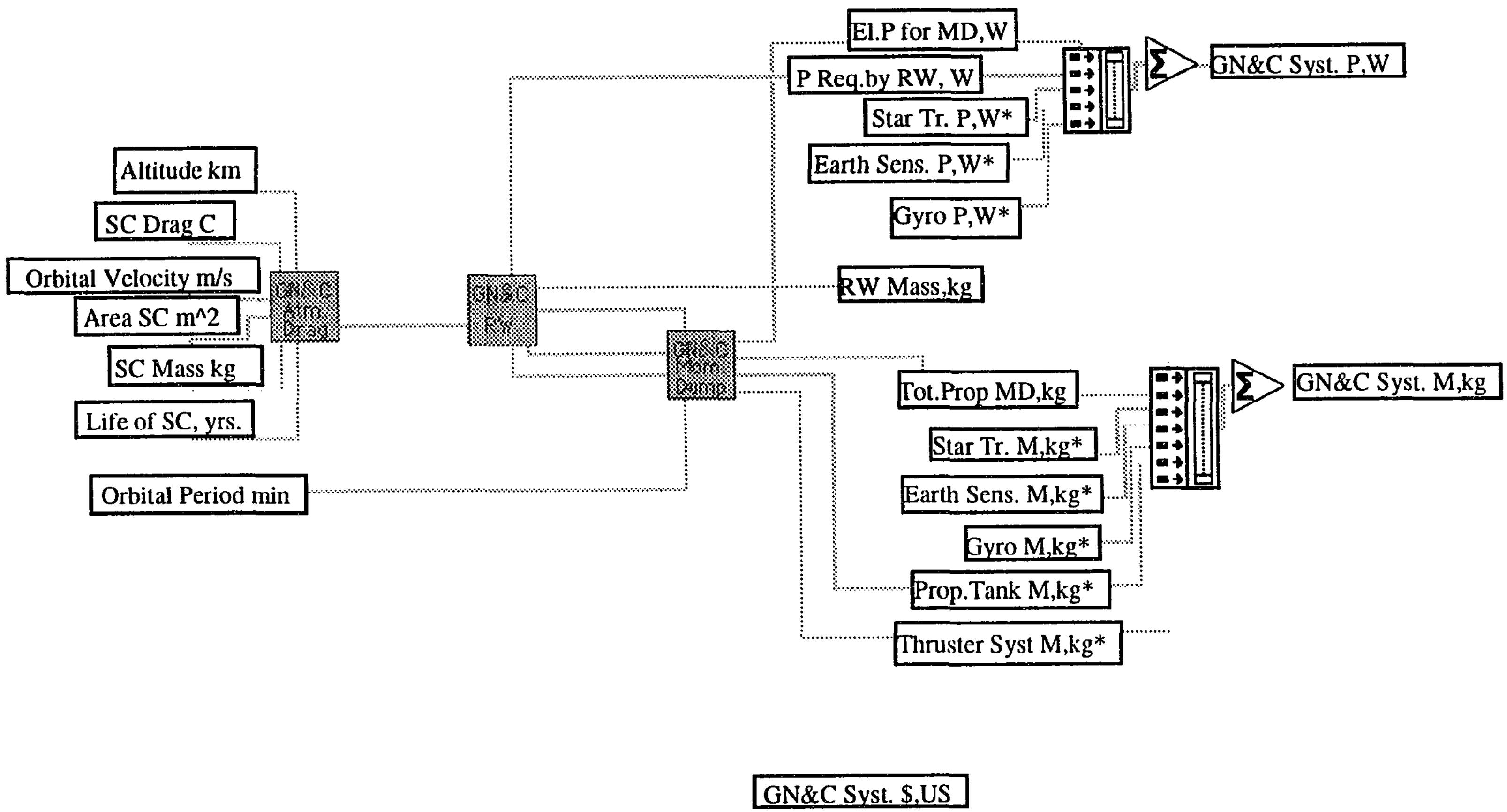


2.4 Communication.vi

Diagram

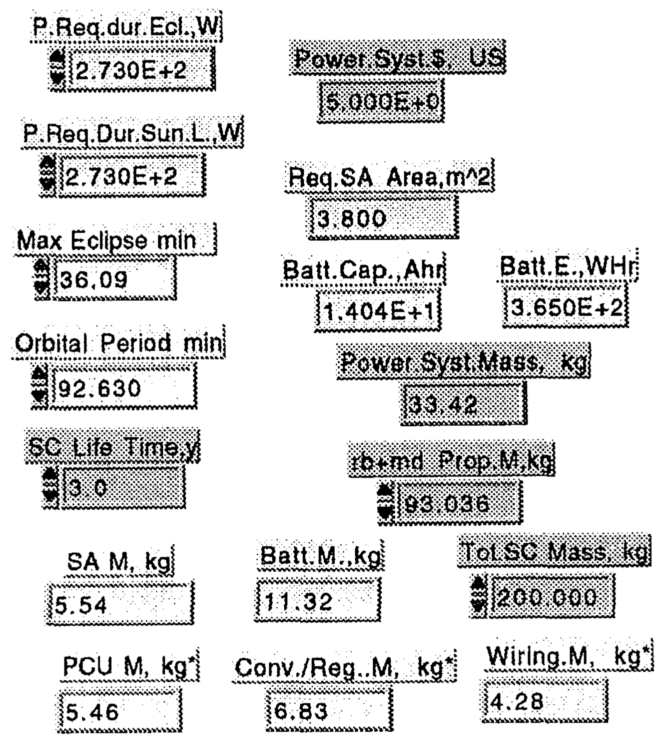


2.5 GN&C.vi.
Diagram

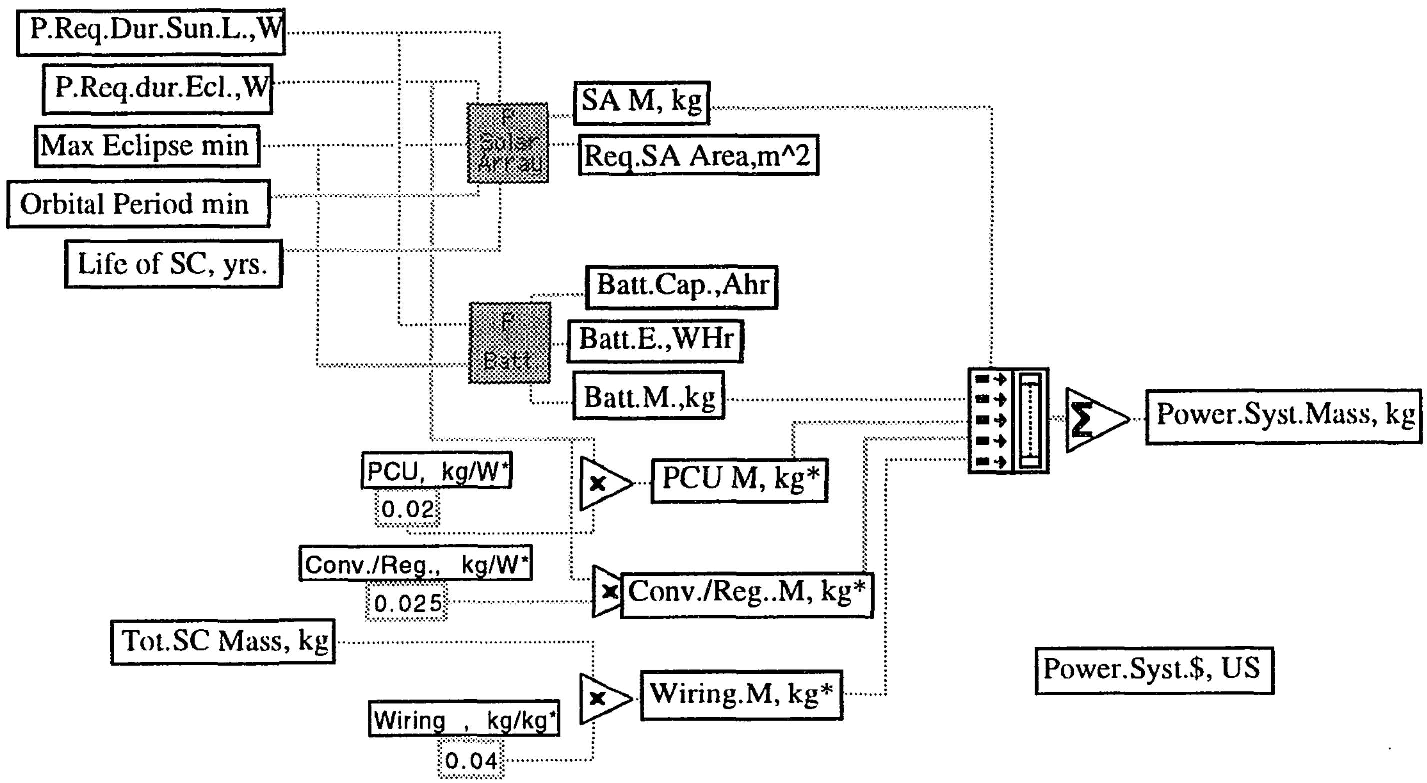


2.6 Power.vi

Front Panel (Controls & Indicators)

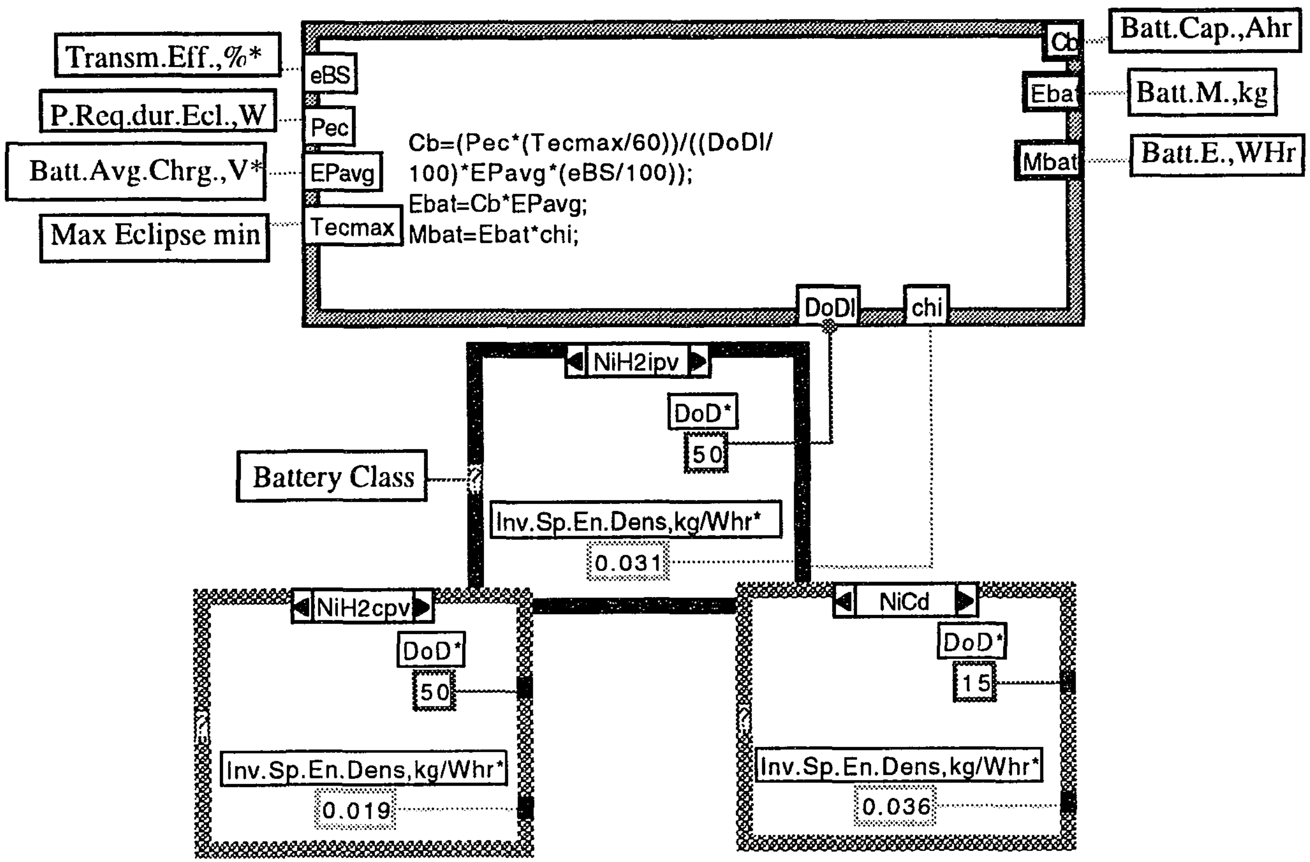


2.6 Power.vi
Diagram



2.6.2 Battery.vi

Diagram



"HELP" Text from 2.6.2 Battery.vi

Transm.Eff.,%*

McDermott, K., Table 11-36, 364.

Batt.Avg.Chrg.,V*

McDermott, K., Table 11-36, 364.

DoD*

NiH2ipv:DoD between 40 and 60%.

NiH2cpv:DoD between 40 and 60%.

NiCd:DoD between 10 and 20%.

McDermott, J.K., 364.

Inv.Sp.En.Dens,kg/Whr*

NiH2ipv:Inverse specific energy between (1/25) and (1/40), in kg/Whr.

NiH2cpv:Inverse specific energy between (1/45) and (1/60), in kg/Whr.

NiCd:Inverse specific energy between(1/25) and (1/30), in kg/Whr.

McDermott, J.K., 362.

Appendix F: COMMENTS

1. Comments from NASA Experts in Branch A.

Feedback from experts at Branch A at NASA Langley Research Center after an initial review of the dissertation draft.

The first three NASA experts work in the same branch and section at NASA Langley Research Center. Their responses have therefore been grouped together. Only one of them has seen the full model, an early version, demonstrated. The two others provided subsystem input.

Issue 1: What do you see as being new about the method and the suggested modeling approach?

NASA 1.

The modeling approach used for this work is similar to that being used in a number of other design and analysis tools. The unique element is the tool (LabView) that was used to perform the implementation of the method. When I first became acquainted with this work, the announced effort included a new and unique approach to performing a high level system design, but the current effort does not include the unique elements of that approach.

NASA 2.

Modeling interactions among subsystems for use in overall system optimization or resource control is not new. What appears new is carrying the interaction modeling deep into the actual subsystem.

NASA 3.

The method is consistent with modern methods of mission and spacecraft design. However, it is not new. Parallel development of subsystems is traditional systems engineering. Software tools, computer modeling, and computer networking create an environment that facilitates system engineering, and computerized modeling for space systems design taking into account subsystem interactions is common practice today. That which would have been new was the consideration of the entire life cycle and the application of this method to a financial analysis capability. A concurrent engineering capability would have then existed as well. That which might have been unique was lost in descoping the work.

Issue 2: How do you see the suggested method and modeling approach fitting into current design processes?

NASA 1.

This modeling approach can be used as a contributing element to a complete system level design effort, but does not comprise but a small part of that effort. The model would need to be extended to a more generic one before more than a narrow set of missions could be accommodated in even this small part of the total design effort,

NASA 2.

The software model could be used for quick modeling a satellite imaging system and, with additional software, could be used to automatically optimize the system to meet certain optimization objectives.

NASA 3.

The method implemented here lends itself to the demonstration of selected design principals and to conducting selected subsystem trade studies, which could be useful during the early design phases of a project. The method used here alone does not provide for a system design. For example, it was not intended to provide a hardware configuration which is required for thermal design, lifetime estimates, propellant trades, orbit selection, and launch mass estimates. As opposed to a design tool, this modeling capability might be more useful as an instructional tool or in a teaching environment.

Issue 3: What do you consider as being the advantages and disadvantages of applying this method and modeling approach to the design process?

NASA 1.

The advantage of performing design using the proposed method is that an integration of the requirements of the various subsystems can be automatically tracked and kept in synchronization. Also, allocations of resources to the various subsystems can be easily done and the consequences of those allocations can be easily shown. The disadvantage is that there is a great deal of information that must be given to the model before any of the advantages can be realized, and there is no indication of whether the model reflects the configuration that is needed to realize the requirements of the mission.

NASA 2.

The advantage is that better detail in the subsystems provides better accuracy in the overall system prediction results. The disadvantage is in the point design nature for each realization of the method: Each new concept requires a whole new system modeling effort. For already developed space systems, rules of thumb exist. For novel systems rules of thumb don't, but then models don't either. Thus, to be useful this method must rely on systems that are well understood. Well understood systems are already well in hand, and thus don't really need a method to help them.

NASA 3.

The implementation In LabView may provide convenience in demonstrating principals and trades and may be preferred by some users.

2. Comments from NASA Expert in Branch B.

Feedback from expert at Branch B at NASA Langley Research Center after an initial review of the dissertation draft.

> The fourth NASA expert also works at NASA Langley Research Center, but at a different branch. He was exposed to this work through a detailed one-to-one presentation of the method and the model. The presentation of the model included a discussion of the equations and programs developed for this research, the model's analysis capabilities, and the results of these analyses.

Issue 1: What do you see as being new about the method and the suggested modeling approach?

To determine what is new we must first examine the current NASA method and modeling approach. First, let us return to only 1990 to establish a base perspective. By 1990 NASA and its support contractors had analyzed space station for almost 10 years with existing methods and approaches and reported that all was go. Then, in July 1990, Fisher and Price reported that the complete design of the space station was flawed because no one had previously determined that the amount of extra vehicular activity required to maintain the space station would absorb virtually all available astronaut time. The problem was that insufficient disciplines were involved in the analysis. To that point, maintainability had not been included in the analyses. Even today, the inclusion of maintainability is a laborious non automated process just as it was for the Fisher-Price task force.

Today, with a few exceptions, the typical spacecraft analysis includes the serial application of a number of independent codes by contractors who run the codes, interpret the code outputs, and input these interpretations into the next code, and so on until a single analysis is complete. The collection of these codes does not represent the totality of the disciplines and considerations necessary to perform a systems engineering analysis. NASA is engineering systems, but, with the exception of the Jet Propulsion Laboratory, is not performing systems engineering and analysis. When disciplines are integrated within analysis, evaluability, designability, prototypeability, testability, produceability, reliability, maintainability, supportability, operability, evolvability, retireability, manageability, quality, scheduling, cost, and risk are not included. Even the inclusion of cost with sizing and trajectories is an exceptionally difficult and expensive code integration task, usually requiring extensive contractor support to supply the knowledge to accomplish the task.

The fact that the disciplines within NASA are not integrated within analyses has been recognized at the NASA Langley Research Center which established a new Branch last year with the sole function of integrating multiple disciplines within multidisciplinary design optimization processes. This Branch will not be at the level of integrating all necessary disciplines within a system level analysis for many years yet.

JPL, in their design-for-cost facility, is the only NASA organization to have successfully integrated all subsystem functions for a system at the system level. This facility was funded approximately a year ago and is still in its infancy. It now permits groups to make design changes and see the effects on all functions of the system in a concurrent engineering format. It is integrated around Excel using Visual Basic, a visual programming language. In January 1995, problems at the design-for-cost facility were in the range of 50 to 150 variables or equations to simulate a whole mission. These equations had to be established uniquely for each planetary or orbital mission. This is the difficult part because mathematical models must be developed to model each mission function with reasonable accuracy. In response to the faster, better, cheaper edict of the NASA Administrator, a group of only 5 professionals develops, produces, and evolves both the facility and the models. The capabilities of this facility represent the state of the art in space systems engineering within the NASA community.

Where does the proposed method and modeling approach stand with respect to this facility?

The LabVIEW modeling framework is quite different from the Excel/Visual Basic framework. It is more applicable to the dynamic simulations needed for the future. For example, the design, manufacturing, operations, and support models required for the future are of the dynamic multiobject Markov model type which can be approximated as differential equations within LabVIEW far easier than within Excel.

The mathematical modeling should be about equal for static simulations. The capacities for LabVIEW are probably greater because of the limited cell capability in Excel as opposed to the extensibility of LabVIEW to use more memory as available.

Given the above, the best description of the newness of the approach is that JPL and the author of this research have been in a technology race, neither aware of the research of the other. Using a metaphor, a group at JPL has developed the equivalent of Taguchi Methods with the support of substantial funding. The author has independently developed the equivalent of response surface methodology on meager funding. Both lead the rest of NASA in concept and elegance of implementation. Both are neck and neck in terms of the current state of the art. The two distinct implementations require similar modeling and do about the same thing today, but the dynamic LabVIEW approach of the author should lead further into the future, given the same amount of support.

At this point in time, there is no known NASA application of multidisciplinary design optimization of a satellite at the system level. The show stopper has been the nonexistence of a simple but adequately descriptive system of equations at the system level. Based on the reported simulation results, it appears that the equation system integrated by the author is adequate to be used as the first such example.

Issue 2: How do you see the suggested method and modeling approach fitting into the current design processes?

To answer this we must first examine the current design processes. They are not the same as those of a year ago because of the new NASA Management Instruction which dictates that NASA shall contract future major projects and that the requirements for those contracts must be functional requirements, as opposed to the design requirements NASA has used historically.

In the past, NASA has accomplished much of the design in house and levied design requirements on the contractor, e.g., space station. Most of the analysis tools which exist today support the now eliminated process described above. NASA, with the exception of the design-to-cost facility at JPL, does not have the analysis tools to support implementation of independent functional requirements. They all assume a rather specific implementation. Hence, they are of little value to the current, but as yet largely unperceived by NASA personnel, NASA design process of providing functional requirements for a contractor to do the design.

The current, but as yet largely unimplemented, NASA design processes require that the new NASA design at the functional level, which is best described by abstract mathematical models which are largely independent of implementation. The tool developed by the author is at this level. It also permits the necessary excursions to lower levels to determine size, complexity, and reliability estimates necessary for cost and schedule estimates. It also permits a simple extension to the process and dynamic models which will be needed in the future.

Issue 3: What do you consider as being the advantages and disadvantages of applying this method and modeling approach to the design process?

Under the new, and hence current, NASA design processes, the ability of the mathematical models within this method and approach to model, integrate, and determine the behavior of the system functional requirements is a necessity which does not currently exist within NASA, outside of the design-to-cost facility at JPL.

The approach to building a system of system level equations is valuable for application within multidisciplinary design optimization.

The fact that models must be built for each system is an advantage, rather than a disadvantage, because the system being modeled will be truly understood by the group using the approach. The process will facilitate the communication necessary within the group to ensure success. It also permits the simple inclusion of new knowledge as it is acquired.

A potential disadvantage relates to the fact that engineers do not like to make visible mistakes in their own discipline. This approach tends to reduce "computer" mistakes so that most of the mistakes will be within the engineering disciplines. Experience indicates that this may turn off many engineers and discourage them from using this approach. But then, do we really want "any" engineer doing the design? Maybe it is really an advantage in disguise.

3. Comments from External Expert

Feedback from external expert after an initial review of the dissertation draft.

> This external expert has provided technical support to NASA Langley Research Center, NASA Headquarters and other NASA centers. He has 10 years of experience in space systems engineering, requirements analysis, and spacecraft mission analysis. During his career he has provided support to space project efforts, such as the Space Station redesign analysis, Space Shuttle failure scenario analysis, Challenger lessons learned analysis, and the development and implementation of the Technical and Management Information System (TMIS) for the Space Station. For the model developed for this research he provided input and comments. He has also seen the model demonstrated.

Issue 1: What do you see as being new about the method and the suggested modeling approach?

This modeling approach represents a new effort to develop a tool of the appropriate scale, complexity and flexibility for the conceptual design process. Unlike other industries, many spacecraft must be designed for requirements that are unique for that mission. This means that detailed, inflexible computer models that require large efforts to develop have resulted in tools that were unable to adapt to new problems. This modeling approach seeks to capture important subsystem interactions and accommodate changes in parameters due to technology improvements. But just as important as demonstrating the tool on a particular problem is for the tool to be flexible to be adapted to a different problem. The use of a graphical programming environment allows the function of each module to be easily understood. It also allows modules to be modified for different spacecraft missions in the future. Smaller and simpler tools would miss important subsystem interactions. Larger and more complex tools would be less flexible to support the conceptual design process. By choosing a programming environment that allows the appropriate ease of use and flexibility, and then developing a model of subsystem interactions for a class of spacecraft, this approach demonstrates a tool that would be useful for conceptual design.

Issue 2: How do you see the suggested method and modeling approach fitting into current design processes?

The modeling approach is appropriate for a small design team working on conceptual design. The adoption of a common tool requires a strong, results driven management. The introduction of a new common tool, requiring each team member to adapt, will be extremely difficult in an existing design team. The total systems model requires each expert to accept responsibility for the content of their subsystem. The expert must 'put on display' the analysis procedures they use for their subsystem. The use of this kind of tool would be easier with a new team, or a new project, where the adoption of the modeling approach is accepted. As this modeling approach is used over a period of time, the total systems model would capture more of the expertise of the engineers. The engineers could concentrate on tracking new technology changes, and let the model handle routine analysis. While the depth of the model could increase gradually, it is important that the scale of the model remain manageable so that the assumptions within the model are understood. The use of the model would also help provide continuity when team members change. The graphical programming environment would help new team members understand the content of their subsystem modules, and be prepared to accept responsibility for those sections.

Issue 3: What do you consider as being the advantages and disadvantages of applying this method and modeling approach to the design process?

The advantage in the conceptual design process will be the ability for each team member to explore many design alternatives rapidly. The total systems model can provide each team member with the approximate response from other subsystems as they explore alternatives within their own subsystem. The common programming environment will allow analysis procedures to be updated and communicated in a functional form. The ease of programming will allow the model to respond to changes with a flexibility appropriate to the conceptual design process. Exploring more alternatives early will improve the selection of a point design. The model will allow the evaluation of sensitivity of the design to minor changes. The model will also allow the team to preserve the reasons for the point design, in case a change in requirements forces a redesign and a return to the conceptual design process. The computer model can also be adapted to compute different parameters (mass, power, cost) and allow the design to be optimized for different objectives. A disadvantage of applying the method is that the benefits of the tool will only be fully demonstrated after the tool is used by a design team on a real problem. For the individual member of a design team, this first requires learning the graphical programming environment and how to implement their analysis procedures. But more importantly, it requires the engineer to have a commitment to make the tool work. They must expose each calculation they make in conducting an analysis, and remove the ambiguity from where they apply "engineering judgment". A person who has built their career on the value of their expertise will not be motivated to reduce their expertise to a handful of equations. Yet somehow the team must share the motivation to make the model work. The introduction of such a tool into a design team can become a problem of psychology and politics.

Appendix G: VI TITLES

Below is a list of the main VI titles used in the text. Those titles that are not self explanatory are spelled out in full next to the title.

1.1 Two Body Motion.vi

1.3 Angular Displacement.vi

1.4. SC Horizon & Swath.vi: Spacecraft horizon and swath width.

1.5 Communication Time.vi

1.6 Eclipse Time.vi

2.1 Propulsion.vi

2.1.1 Propulsion System.vi

2.1.2 Prop for Reboost.vi: Propulsion system mass (including propellant) for the reboosting system.

2.2 Sensor.vi

2.2.1 Apert.Diam;pnl;dnl.vi: Aperture diameter for photon noise limited and noise limited sensor systems.

2.2.2 Scanner Power.vi

2.2.3 Sensor H&S.vi: Sensor horizon and swath width.

2.2.4 Sens. DR.vi: Sensor data rate.

2.3 DP & DS.vi: Data processing and data storage.

2.3.1 Data Processing.vi

2.3.2 Data Storage.vi

2.4 Communication.vi

2.4.1 Comm. DR.vi: Communication data rate.

2.4.2 Comm Power.vi: Communication system power.

2.4.3 Comm.Mass.vi: Communication system mass.

2.5 GN&C.vi: Guidance Navigation and Control.

2.5.1 Atmosph.Drag Eff.vi: Atmospheric drag effect.

2.5.1 Atmosph.Drag;Atm.Dens.vi: Atmospheric drag from atmospheric density.

2.5.2 Sizing the RW.vi: Sizing the reaction wheel.

2.5.3. Mom. Dump.vi: Momentum dumping.

2.6 Power.vi

2.6.1 Solar Array, P&A.vi: Solar array, power and area.

2.6.2 Battery.vi

Appendix H: UNITS OF MEASURE

Below is a list of some of the abbreviated units of measure used in the text.

bps; Bits per Second
Hz; Hertz
IPS; Instructions per Second
J; Joules
Js; Joule-Seconds
K; Kelvin
kg; Kilograms
KIPS; Thousand IPS
km; Kilometers
m/s; Meters per Second
min; Minutes
MIPS; Million IPS
N; Newtons
Nm; Newton-Meters
rad; Radians
rpm; Revolutions per Minute
s; Seconds
sec; Seconds
V; Volts
W; Watts
Whr; Watt-Hours
y; Years

BIBLIOGRAPHY

AEC-Able Engineering Company, Inc., Company Literature, P.O. Box 588 Goleta, CA 93116-0588, or 93 Castilian Drive, Goleta, CA 93117.

Acetta, J.S., Infrared Search and Track Systems, in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Co-published by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington, 1993, 209-344.

Agrawal, B. N., *Design of Geosynchronous Spacecraft*, Englewood Cliffs, NJ, Prentice-Hall, 1986.

Aguilar, R.J., *Systems Analysis and Design in Engineering, Architecture, Construction, and Planning*, Englewood Cliffs, NJ, Prentice-Hall, 1973.

Aslaksen, E., Belcher, R., *Systems Engineering*, Englewood Cliffs, NJ, Prentice-Hall , 1992.

Ballou, D.P., Pazer, H.L., A Framework for the Analysis of Error in Conjunctive, Multicriteria, Satisificing Decision Processes, *Decision Sciences*, 1990, 21, 752-771.

Bell, L., Bilby, C., *The Lunar Base Model and its Role in Systems Optimization*, Proceedings of the Annual Invitational Symposium, Space Mining and Manufacturing, UA/NASA Space Engineering Research Center for Utilization of Local Planetary Resources, Tucson, Arizona, 1989.

Blanchard, B.S., Fabrycky, W.J., *Systems Engineering and Analysis*, Englewood Cliffs, NJ, Prentice-Hall, 1981.

Boatwright, J.E., Mueller, R.L., Command and Data Handling, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 342-348.

Boden, D.G., Introduction to Astrodynamics, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 113-140.

Brodsky, R.F., Defining and Sizing Space Payloads, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 213-254.

Brooks, D. R., *An Introduction to Orbit Dynamics and Its Application to Satellite Based Earth Monitoring Missions*, NASA Reference Publication 1009, NASA Langley Research Center, 1977.

Brown, C.D., *Spacecraft Mission Design*, AIAA Education Series, American Institute of Aeronautics and Astronautics, Washington, D.C., 1992.

Casani, K.E., Thomas, N.W, The Flight System Testbed, SPIE-94, Conference.

Cantella, M.J., Staring-Sensor Systems, in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Copublished by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington, 1993, 157-207.

Chamberlain, R.G., Fox, G., Duquette, W.H., *A Design Optimization Process for Space Station Freedom*, JPL Publication 90-23, NASA, June 15, 1990.

Chetty, P. R. K., *Satellite Technology and its Applications*, 2nd ed., Blue Ridge Summit, PA, Tab Books, 1991.

Clarkson, M., Tools on the Fly: Virtual Instruments Take Off, *Desktop Engineering*, September/October 1995, 1, 1, 20-28.

Colton, J.S., An Intelligent Evaluation of Design for Manufacturing, in Kusiak, A. (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993.

Conte, D., *Payload Overview, System Design Audit (SDA), LightSats*, TRW, August, 1994, 170-210.

Costogue, E.N., Hasbach, W.A., *Solar Array Technology Developments and Design Evolutions for Electric Propulsion Applications*, Princeton/AIAA/DGLR 14th International Electric Propulsion Conference, Princeton, NJ, 1979, 5.

Cutler, A.H., *Space Manufacturing 5, Engineering with Lunar and Asteroidal Materials*, Proceedings of the Seventh Princeton/AIAA/SSI Conference, May 8-11, 1985.

Dahlstrom, E.L., *Commercial Remote Sensing Elevation Requirements*, Lockheed Engineering and Sciences Company, January 1994.

Davies, R.S., Communications Architecture, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 441-484.

Dornheim, M.A., New Sensors Show Two Paths to Reconnaissance, *Aviation Week and Space Technology*, A McGraw-Hill Publication, July 10, 1995, 42-46.

Doukas, P.G., McCandless, J.R., Sarafin, T.P., Britton, W.R., Structures and Mechanisms, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 387-426.

Durand, R.L., Technology Services Corporation, LabVIEW for Windows Cuts Development Time for Radar Simulator, Instrumentation Newsletter, *Technical News from National Instruments*, Vol. 7, Number 2, Summer 1995, 13.

Dyer, J.S., Fishburn, P.C., Steuer, R.E., Wallenius, J., Multiple Criteria Decision Making, Multiattribute Utility Theory: The Next Ten Years, *Management Science*, 1992, 38, 5, 645-645.

Fertig, K.W., Gonda, M., Computer Environments for Design and Analysis Design Sheet: An Engineer's Spreadsheet, Rockwell International Science Center, Palo Alto Laboratory, December 8, 1993.

First Lunar Outpost System Effectiveness Report, EXPO Internal Document No. XE-92-005, Exploration Programs Office, NASA-JSC, May 1992.

Fleeter, R., Warner, R., Design of Low-Cost Spacecraft, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 711-738.

Flynn, M.J.O., Ahmad, M.M., Computer-Based Concurrent Engineering Systems, in Parasei, H.R., Sullivan, W.G. (Editors), *Concurrent Engineering: Contemporary Issues and Modern Design Tools*, 1st ed., London, New York, Chapman and Hall, 1993, 184-206.

Ford, J., Communications, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 329-341.

Fortesque, P. W., Attitude Control, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 229-253.

Fortesque, P. W., Dynamics of Spacecraft, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 33-54.

Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992.

Franklin, J.L., Dean, E.B., *Interactive Graphics for Computer Aided Network Design*, Proceedings National Computer Conference 1973, 677-683.

Franklin, J.L., Dean, E.B., Computer-Aided Design with Interactive Graphics (CANDIG) System, *SID Journal*, May-June 1974, 5-13.

Friend, J.K., Jessop, W.N., *Local Government and Strategic Choice*, 2nd ed., London, Pergamon Press, 1977.

Gerschenson, J., Ishii, K., Life Cycle Serviceability Design, in Kusiak, A. (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993, 363-384.

Glaseman, S., Hansen, L.J., Pollock, C.H., Thimlar, M., Spacecraft Computer Systems, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 541-577.

Gorney, D.J., Blake, J.B., Koons, H.C., Schultz, M., Vampola, A., Walterscheid, R.L., The Space Environment, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 183-197.

Griffin, M. D., French, J. R., *Space Vehicle Design*, AIAA Education Series, American Institute of Aeronautics and Astronautics, Washington, D.C., 1991.

Hall, A.D., Systems Engineering from an Engineering Viewpoint, in Sage, A.P. (Editor), *Systems Engineering: Methodology and Applications*, New York, IEEE Press, 1977, 13-17.

Hansen, L.J., Pollock, C.H., Spacecraft Computer Systems, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, 2nd ed., London, Kluwer Academic Publishers, 1992, 603-635.

Hopper, G.S., Forward-Looking Infrared Systems, in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Co-published by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington, 1993, 103-155.

Hundal, M.S., Designing for Cost, in Parasei, H.R., Sullivan, W.G. (Editors), *Concurrent Engineering: Contemporary Issues and Modern Design Tools*, 1st ed., London, New York, Chapman and Hall, 1993, 329-351.

Jain, R.K., Weinberg, L., *Status and Future Directions of InP Solar Cell Research*, NASA-TM-1055426, NASA Lewis Research Center, 1992.

Jamieson, J.A., McFee, R.H., Plass, G.N., Grube, R.H., Richards, R.G., *Infrared Physics and Engineering*, New York, McGraw-Hill Book Company, 1963.

JTEC (Japanese Technology Evaluation Center)/WTEC (World Technology Evaluation Center) - Loyola College, *NASA/NSF Panel Report on Satellite Communications and Systems Technology* - Volume I Analytical Chapters, NASA/NSF Conference - February 1992 , Washington, D.C., May 1993.

JTEC (Japanese Technology Evaluation Center)/WTEC (World Technology Evaluation Center), *Satellite Communication in Europe, Russia and Japan*, NASA/NSF Conference - Viewgraphs, Washington, D.C., February 1992.

Kaufman, B.E., A New Theory for Satisificing, *The Journal of Behavioral Economics*, 1990, 19, 1, 35-51.

Keller, R., *Production of Oxygen from Lunar Soil by Molten Salt Electrolysis*, Annual Invitational Symposium, Space Mining and Manufacturing, UA/NASA Space Engineering Research Center for Utilization of Local Planetary Resources, Tucson, Arizona, 1989.

Klement, M.A., Design for Maintainability, in Kusiak, A. (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993, 385-400.

Kramer, H.J., *Observation of the Earth and Its Environment, Survey of Missions and Sensors*, 2nd ed., Berlin, Springer Verlag, 1994.

Koelle, D.E., *TRANSCOST - Statistical-Analytical Model for Cost Estimation and Economic Optimization of Space Transportation Systems*, MBB-Report No. URV-185(91), March 1991.

LabVIEW for Macintosh, *User Manual*, National Instruments, December 1993 Edition, Part Number 320591-01.

Law, A.M., Kelton, W.D., *Simulation Modeling and Analysis*, 2nd ed., New York, McGraw-Hill, Inc., 1991.

Marino, K., The New Millennium Program, *JPL Universe*, February 10, 1995.

McCracken, W.L., Infrared Line Scanning Systems, in Campana, S.B. (Editor), *The Infrared and Electro-Optical Systems Handbook, Volume 5, Passive Electro-Optical Systems*, Co-published by Environmental Research Institute of Michigan, Ann Arbor, Michigan, and Spie Optical Engineering Press, Bellingham, Washington, 1993, 23-101.

McDermott, J.K., Power, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 349-369.

McMordic, R.K., Thermal, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 370-386.

Meredith, B.D., *Mission Data Package for Earth Science and Commercial Remote Sensing*, OACT Small Spacecraft Technology Workshop, NASA LaRC, 1993.

Moss, B.J., Propulsion Systems, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 106-143.

NASA/Office of Advanced Space Concepts and Technology (prepared by KPMG Peat Marwick), Remote Sensing Strategic Action Plan, Washington, D.C., January 1993.

Newnan, D.G., *Engineering Economic Analysis*, 4th ed., Engineering Press, Inc., San Jose, CA, 1988.

Noble, J.S., Economic Design in Concurrent Engineering, in Parasei, H.R., Sullivan, W.G. (Editors), *Concurrent Engineering: Contemporary Issues and Modern Design Tools*, 1st ed., London, New York, Chapman and Hall, 1993, 352-372.

Noble, J.S., Tanachoco, J.M., Design for Economics, in Kusiak, A. (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993, 401-436.

Oxnevad, K.I., *An Investment Analysis for Space Mining Ventures*, International Astronautical Federation (IAF) Conference, Montreal, 1991.

Oxnevad, K.I., *Evaluating and Optimizing Large Infrastructure Space Projects*, International Space University, Huntsville, AL, July 1993.

Oxnevad, K.I., Space and Offshore, *Norwegian Oil Review (NOR)*, August 1992, No.7, 101-114, 169-170, 176-178.

Pidgeon, D., Price, K. M., Tsao, A., *Mass and Power Modelling of Communication Satellites*, NASA Lewis Research Center Contract # NAS3-25092, Space Systems/LORAL, December 1991.

Princeton Synergetics, Inc., *Technical Description of TECHSAT, A Computer Aided Spacecraft Design System for Earth Observing Satellites*, NASA Contract No. NAS23-26313, Cleveland, August 5, 1992.

Project Design Center, Jet Propulsion Labiratory (JPL), Publication JPL 400-533, July 1994.

Ramohalli, K., Kirsch, T., A "Figure of Merit" Approach to Extraterrestrial Resource Utilization, Proceedings of the Annual Invitational Symposium, Space Mining and Manufacturing, UA/NASA Space Engineering Research Center for Utilization of Local Planetary Resources, Tucson, Arizona, 1989, vl-v11.

Reeves, E., Spacecraft Design and Sizing, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 255-300.

Reeves, R. G., Anson, A., Landen, D. (Editors), *Manual of Remote Sensing*, Vol.1, American Society of Photogrammetry, Falls Church, VA, 1975.

Rosenhead, J., Elton, M.E., Gupta, S.K., Robustness and Optimality as Criteria for Strategic Decisions, *Operational Research Quarterly*, 1972, 23, 4, 413-431.

Rosenhead, J., Gupta, S.K., Robustness in Sequential Investment Decisions, *Management Science*, 1968, 15, 2, B18-B29.

Sackheim, R.L., Wolf, R.S., Space Propulsion Systems, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 579-606.

Sage, A.P. (Editor), *Systems Engineering: Methodology and Applications*, New York, IEEE Press, 1977.

Schenkerman, S., Use and Abuse of Weights in Multiple Objective Decision Support Models, *Decision Sciences*, 1991, 22, 369-378.

Schrage, D.P., Concurrent Design: A Case Study, in Kusiak, A. (Editor), *Concurrent Engineering: Automation, Tools, and Techniques*, New York, J. Wiley, 1993, 535-581.

Short, N.M., *The LANDSAT Tutorial Workbook, Basics of Satellite Remote Sensing*, NASA Reference Publication 1078, NASA Goddard Space Flight Center, 1982.

Silva, L. R., Radiation and Instrumentation in Remote Sensing, in Swain, P. H., Davis, S. M. (Editors), *Remote Sensing. The Quantitative Approach*, New York, McGraw-Hill, Inc., 1978, 21-133.

Simmons, K.P., SEA CORP., LabVIEW Slashes Development Time of Submarine Sonar Test System, Instrumentation Newsletter, *Technical News from National Instruments*, Autumn 1995, Vol. 7, Special Academic Edition, 12.

Simonds, C.H., Ballin, M., Schlater, N., *NASA TECHNOLOGY INVESTMENT: An Application of Decision Analysis*, Presented to the Exploration Programs Office, November 17, 1992.

Simonds, C.H., *Decision Analysis for Space Systems: Overview and Summary of Pilot Study*, Presentation as Part of Subcontract to Bionetics Corp., Subcontract: 906-317, Prime Contract NAS 2-13, 273, 1993.

Skinner, D.L, QUICK: An Interactive Software Environment For Engineering Design, 1989, http://pdc.jpl.nasa.gov/QUICK/QUICKA5.html.

Skjong, R., Lerheim, J., Economic Risk of Offshore Field Development, AACE (American Association for Cost Engineers) Proceedings, 1988.

Smith, H., Telecommunications, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 303-346.

Space Station Redesign Team, Final Report to the Advisory Committee on the Redesign of the Space Station, NASA, Washington, D.C., June 1993.

SPOT Image, *The Catalogue of SPOT Products and Services*, Spot Image, Toulouse, France, 1989.

Stansbery, E., *Architecture Effectiveness Model Summary*, Lunar and Mars Exploration Program Office, NASA - JSC, April 18, 1991.

Stark, J. P. W., Mission Analysis, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 83-106.

Stark, J. P. W., Celestial Mechanics, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 59-81.

Stark, J. P. W., Electrical Power Systems, in Fortesque, P. W., Stark, J. P. W. (Editors), *Spacecraft Systems Engineering*, New York, J. Wiley, 1992, 255-273.

Throckmorton, A., Tadikonda, K., Leath, K., *SmallSat Accommodation of Precision Payloads*, NASA Langley Research Center Contract # NAS 1-19244, McDonnell Douglas, April 1994.

Turino, J. L., *Managing Concurrent Engineering: Buying Time to Market*, New York, Van Nostrand Reinhold, 1992.

User Systems, Inc., *WRAP A Space-Based Remote Sensor and Satellite Data Base for Your Personal Computer*, Chesapeake Beach, MD, 1993.

Valle-Riestra, J.F., *Project Evaluation in the Chemical Process Industries*, New York, McGraw-Hill, 1983.

Waldron, R.D., Processing Options for Lunar Propellant and By-Products, Annual Invitational Symposium, Space Mining and Manufacturing, UA/NASA Space Engineering Research Center for Utilization of Local Planetary Resources, Tucson, Arizona, 1989.

Wertz, J.R., Space Mission Geometry, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 79-112.

Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991.

Whitworth, G. G., Ground System Design and Sizing, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 517-540.

Wise, R.M., Department of Surgery, Thoracic and Cardiovascular Division, School of Medicine, University of Maryland, LabVIEW our Choice for Cardiothoracic Research, Instrumentation Newsletter, *Technical News from National Instruments*, Autumn 1995, Vol. 7, Special Academic Edition, A-4.

Wong, R., Cost Modelling, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 657-682.

Woodcock, G., *Space Transfer Concepts and Analysis for Exploration Missions*, Contract NAS8-37857, Executive Summary, Phase 1, Boeing Defense and Space Group, Huntsville, February 1991.

Zermuehlen, R.O., Zimbelman, H.F., Guidance, Navigation, and Control, in Wertz, J.R., Larson, W.J. (Editors), *Space Mission Analysis and Design*, London, Kluwer Academic Publishers, 1991, 303-328.

Zuppero, A., Whitman, P., *Discovery of New Resources Near Earth*, Draft Workshop Report, Space Energy Resource Utilization Conference, University of Southwestern Louisiana, March 1993, 26-27.

VITA

Knut I. Øxnevad
Born: November 16, 1959, Bergen, Norway
Mailing address: Sandved Terrasse 27, 4300 Sandnes, Norway

Knut I. Øxnevad received his Ms. in Economics ("siviløkonom" title) in 1984 from Norges Handelshøyskole (the Norwegian School of Economics and Business Administration). From 1985 to 1989 he held positions at the International Division of Den norske Bank gaining experience within the areas of international finance, stocks and bonds trading, country risk analysis, and in developing large and complex transaction and exposure control systems for the banking industry. In the fall of 1989 he refocused his career towards space systems engineering with emphasis on space system design and was accepted to the Ph.D. program in Engineering Management at the Old Dominion University. From that time he has worked on issues ranging from planetary bases and asteroid mining, to earth observation systems. He is a graduate of the International Space University (ISU) and has on a number of occasions given lectures at their programs. Some of his work was also presented at the International Astronautics Federation (IAF) conference in 1991. His publications include:

1. *A Study of Petri-Nets for Modeling and Analysis of Complex Systems*, NASA Contract NAS1-18584--Task Assignment No. 105 (181051), NASA LaRC, Hampton, VA, co-author Dr. Resit Unal, December 1992.

2. *Space and Offshore - New Opportunities for the Norwegian Offshore Industry*, the Norwegian Oil Review - NOR, No. 7, 101, 18 pages, August 1992.
3. (Special Section on the Parallels between Offshore and Space Technologies).

4. *An Investment Analysis Model for Space Mining Ventures*. Presented at the IAF Conference in Montreal, Canada, 1991.

5. *International Asteroid Mission (IAM)*, the International Space University (ISU), co-author, Toronto, Canada, 1990.

6. *A study of the Structure and the Dynamics in the Market for Aviation Services.*(Thesis required for the Ms. in Economics at Norges Handelshøyskole, the Norwegian School of Economics and Business Administration.), NHH, Bergen, Norway, 1984.