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The Preliminary Design of a Standardized Spacecraft Bus for Small Tactical Satellites

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STANDARDIZED SPACECRAFT BUS FOR
SMALL TACTICAL SATELLITES

THESIS

Written by GSO & GSE team

AFIT/GSE/GSO/ENY/96D-1

DEPARTMENT OF THE AIR FORCE

AIR UNIVERSITY

AIR FORCE INSTITUTE OF TECHNOLOGY

Wright-Patterson Air Force Base, Ohio

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Disclaimer

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

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THESIS

Presented to the Faculty of the School of Engineering

of the Air Force Institute of Technology

Air University

in Partial Fulfillment of the Requirements for the

Degree of Master of Science

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Approved for public release; distribution unlimited

Preface

The following document is the culmination of the work performed by a team of eight graduate engineering students assigned to the Air Force Institute of Technology (AFIT). The students compiled this document while performing a systems engineering design study to create a small standardized tactical satellite bus for the Phillips Laboratory. This document is divided into three separate volumes. Each volume is an integrated element of the student thesis but it can also serve as a stand alone document.

The first volume is the Executive Summary. The purpose of the Executive Summary is to present a synopsis of the design study results to the sponsor at the Phillips Laboratory. This volume includes information on the methods employed during the study, the scope of the problem, the value system used to evaluate alternatives, tradeoff studies performed, modeling tools utilized to create and analyze design alternatives, recommendations and implications of the alternatives, and areas where future research should be considered.

The second volume is a detailed account of the design process. The steps of the team's innovative design process and the team organization are initially presented. Each phase of the design study is discussed in subsequent sections. Phase I provides accounts of the team's initial attempt to apply a well known systematic approach to satellite design. Efforts concentrate on defining the problem posed by the sponsor. "First cuts" at developing analysis tools and models are performed. Additionally, different alternatives are generated as possible solutions to the problem. An initial analysis and evaluation is performed to define an initial solution space, and to verify the analysis tool. Phase II is an iterative step in the design process and serves as a reservoir for the team's most meaningful work. The team realized that a new systematic approach had to be applied to the study. This phase provides the results of the application of that innovative approach. It is here that the understanding of the problem is further refined and decisions are made that limit the scope of the study. The objective hierarchy is further developed and a value system is created as a method for measuring each design alternative. Information is collected on satellite designs and satellite subsystems. Tradeoffs are performed to determined the best methods and components to be used in the alternatives. A model is created and design alternatives are generated. System analysis is performed on the alternatives using the value hierarchy, and results are generated. Sensitivity analysis is performed on the alternatives, and implementation recommendations are provided to the sponsor.

The third volume provides details on the tools developed to build a satellite and to analyze the design. There are three sections to this volume. The first section describes the model's philosophy and presents details on the purpose and operation of each module of the model. Mathematical formulae and module architecture are also described in this section. The second section is a user's guide to operating the model. Specific details of the sequence to be used and information required to run the model are provided in this discussion. The final section of this volume is the actual code of the model. The code is contained in an annex and is maintained by AFIT's Aeronautics Department at Wright-Patterson AFB, Ohio. The code can be provided to allow future modelers to understand and refine the work that has been accomplished.

Acknowledgments

The systems engineering design team would like to thank all the people who have provided their guidance, support, instruction, and personal time to ensure that the design study was a success. Special thanks go to the team's advisors, Lt Col Stuart Kramer, Maj Ed Pohl, and Dr Chris Hall. The team also realizes that it took more than just the team members to make the study meaningful and complete. Therefore, we wish to acknowledge the efforts of Lt Col Stan Correia, Maj Brad Prescott, Maj Scott Thomason, Doug Holker, Edward Salem, Lt Mike Rice, Capt Joel Hagan, Dave Everett, Col (ret) Edward Nicastri, Lt Col Brandy Johnson, Richard Warner, Linda A. Karanian, and the Space Warfare Center for the assistance and expertise they provided throughout our study. Finally, we would like to thank our families, who supported this effort with their patience and understanding: Sheri Carneal; Sedef and Sena Cokuysal; Rebecca, Austin and Travis From; Donna Krueger; and Coleen Robinson.

The Systems Engineering Team

Satellite Skunkworks

VOLUME I: EXECUTIVE SUMMARY

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Symbols

a	The semi-major axis (km)
Asa	Solar area (m2)
Asc	Surface area of component
CG	Center of gravity
C_r	Capacity required
d	Density (gr/cm^3)
D	Degradation per year
Fet	Configuration factor
g	Gravitational constant (9.806 m/sec^2)
GaAr	Gallium arsenide
H	Angular momentum
Het	Earth's emitted IR
Hsu	Solar constant
I_d	The inherent degradation
I_{sp}	Specific impulse (m/sec)
I_t	Total impulse (m/sec)
I_original	Original inertial matrix for a given sphere, cylinder, cone, etc.
I_total	Overall satellite inertial matrix
k	Specific heat ratio
LC^3	Linear, charge-current-control
L_d	Lifetime degradation
m_o	Initial vehicle mass (kg)
m_p	Mass of propellant consumed (kg)
M_{sa}	Mass of solar array
O/F	Mixture ratio for bi-propellants (m_{ox} / m_{fuel})
P	Pressure (Pa)
P_0	Power output (W/m^2)
PAS	Projected solar area
Qds	Incident solar energy on the satellite (α*PAS*Hsu)
Qer	Reflected solar energy (sun-earth-satellite)
Qet	Earth emitted radiation (ε*Fet*Asc*Het)
Qint	Heat output of the subcomponent
r	Radius
r_x	The transformation matrix converting the object's local frame of reference to the satellite's launch cg frame of reference

R	Gas constant (J/(Kg.K))
R_b	Blow-down ratio (V_{gf}/ V_{gi})
R_m	Mass ratio (m_o / m_f)
SA	Solar array
Si	Silicon
SL	Satellite design life
SR	Shunt regulator
t	Thickness
T	Temperature (K)
Tsc	Temperature of subcomponent
TP	The orbital period (min)
TP_d	The minimum daylight period (min)
TP_e	The maximum eclipse period (min)
X_d	The efficiency of the path directly from the arrays to the loads.
X_e	The efficiency of the paths from the solar arrays through the batteries to the individual loads.
α	Solar absorptivity of the subcomponent
DI	Required specific impulse during mission of satellite (m/sec)
DV	Required delta velocity during mission of satellite (m/sec)
ε	IR emittance of the subcomponent
ϕ	The eclipse rotation angle(deg)
μ	The gravitational parameter (3.986012 x 105)
ρ	Angular radius of the earth (deg)
σ	Stefan-Boltzman constant (5.67x10-8 W-m-2-K-4)
s	Allowable stress

Abbreviations

AF	Air Force
AFIT	Air Force Institute of Technology
AFSCN	Air Force Satellite Control Network
ADCS	Attitude Determination and Control System
ARPA	Advanced Research Project Agency
ASAT	Anti satellite
ATSSB	Advanced Technology Standardized Satellite Bus
BER	Bit error rate
BOL	Begin of life
BW	Band width
CAD	Computer aided design
CCD	Charge coupled device
C&DH	Command and data handling
CDM	Chief decision maker
CEO	Chief executive officer
CTO	Chief technical officer
CER	Cost estimation relationship
cmds	commands
CMG	Control-moment gyro
COTS	Commercial off the shelf
CTA/STEP	Space Test Experiment Program
dB	Decibel
DET	Direct energy transfer
DOD	Depth-of-discharge
EO	Electro-optical
EOL	End of life
EM	Electro-magnetic
FMC	Financial Management and Comptroller
EPDS	Electrical Power and Distribution Subsystem
FY96$M	Fiscal year 1996 million dollars
GERM	General-Error Regression Model
GUI	Graphic User Interface
HAPS	Hydrazine Auxiliary Propulsion System
HETE	High Energy Transit Experiment
HTML	Hyper Text Markup Language
IFOV	Instantaneous field of view

IMU	Inertial measurement unit
IR	Infrared
IRQ	Interrupt request
KISS	Keep it short and simple
LASER	Light Amplification by Stimulated Emission of Radiation
LEO	Low Earth orbit
LEV	Launch Equipment Van
LIDAR	Light Detection and Ranging
LMLV	Lockheed Martin Launch Vehicle
LOWTAC	Low Tactical
LSE	Launch Support Equipment
LV	Launch vehicle
LWIR	Long wave infrared
m	Mass
MAXTAC	Maximum Tactical
MB	Megabyte
MIDTAC	Medium Tactical
MOE	Measure Of Effectiveness
MML	Mean mission life
MPE	Minimum percentage error
MSI	Multispectral-imaging
MSTI	Miniature Sensor Technology Integration
MTBF	Mean Time Between Failure
MUPE	Minimum unbiased percentage error
NASA	National Aeronautics and Space Administration
NIR	Near infrared
NiCd	Nickel - Cadmium
OLS	Ordinary least squares
OSC	Orbital Science Corporation
PPT	Peak-power tracking
PCSOAP	Personal Computer Satellite Orbital Analysis Program
RAM	Random access memory
RDT&E	Research, development, test and engineering
RER	Requirements estimation relationships
SAR	Synthetic aperture radar
SOH	Spacecraft state-of-health
SGLS	Space to ground link
SMAD	Space Mission Analysis and Design
SMC	Space and Missile System Center
SNR	Signal to noise ratio

SOS	Satellite operating system
SOW	Statement of work
SPIG	Spacecraft-To-Payload Interface Guideline
SSCM	Small Satellite Cost Model
SSLV	Single stage launch vehicle
SSRM	Single stage rocket motor
TFU	Theoretical first unit
TSS/SPI	Tactical Support Satellite/Standard Payload Interface
TT&C	Telemetry, Tracking and Commanding
TWTA	Traveling Wave Tube Amplifiers
U.S.	United States
USCM	Unmanned Space Vehicle Cost Model
UV	Ultraviolet
VIS	Visible
V	Volume
VSD	Value System Design

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Abstract

A PRELIMINARY DESIGN OF A STANDARDIZED SPACECRAFT BUS FOR SMALL TACTICAL SATELLITES

Current satellite design philosophies concentrate on optimizing and tailoring a particular satellite bus to a specific payload or mission. Today's satellites take a long time to build, checkout, and launch. Space Operations planners, concerned with the unpredictable nature of the global demands placed upon space systems, desire responsive satellite systems that are multi-mission capable, easily and inexpensively produced, smoothly integrated, and rapidly launched. This emphasis shifts the design paradigm to one that focuses on access to space, enabling tactical deployment on demand and the capability to put current payload technology into orbit, versus several years by today's standards, by which time the technology is already obsolete. This design study applied systems engineering methods to create a satellite bus architecture that can accommodate a range of remote sensing mission modules. System-level and subsystem-level tradeoffs provided standard components and satellite structures, and an iterative design approach provided candidate designs constructed with those components. A cost and reliability trade study provided initial estimates for satellite performance. Modeling and analysis based upon the Sponsor's objectives converged the designs to an optimum solution. Optimum design characteristics include a single-string architecture, modular solar arrays, an internet-style command and data handling system, on-board propulsion, and a cage structure with a removable frame for easy access to subsystem components. Major products of this study include not only a preliminary satellite design to meet the sponsor's needs, but also a software modeling and analysis tool for satellite design, integration, and test. Finally, the report provides an initial implementation scheme and concept for operations for the tactical support of this satellite system.

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1. Report Overview

This document provides the results of a group design study performed at the Air Force Institute of Technology. The team of eight graduate engineering students examined the design of a generic satellite bus for small tactical satellite applications. The project was sponsored by LtCol James Rooney of the United States Air Force's Phillips Laboratory in Albuquerque, New Mexico. Similar design studies have been completed by various companies and laboratories, but to date success has been limited. Phillips Laboratory's goal was to seek a "clean-sheet" approach to the design of a cost-effective satellite bus. Several design characteristics were suggested by the sponsor and were considered throughout the project. These characteristics included modularity, flexibility, robustness, and operability. These characteristics have been treated as guidance in developing objectives and alternative design architectures and were not treated as hard requirements.

This is the first volume of a three volume report. Volume I is an Executive Summary of the work performed by the design team. Volume II provides greater detail of the work and includes the theory and analysis behind the team's approach to the problem. Volume III is an in-depth explanation of the modeling performed for the project and includes the associated code.

This volume provides a high level discussion of the results of the team's efforts. The first section discusses the design process that evolved during this study. The remaining sections document the results in each of the steps of the systematic process.

A majority of the work was performed in the second iteration of the design study. This volume presents this information and contains a discussion on the scope of the problem, the value system design, the decisions made in the tradeoffs section, and an overview of the modeling efforts. Different design alternatives are presented in system synthesis and the analysis of the alternatives are documented in the system analysis section. Sensitivity analysis is included as part of decision making and the implementation plan discusses how the selected alternative can be integrated into space operations. The final section of this report provides a discussion on future technologies and areas where further research can enhance the products of this study.

2. Design Process

The design team recognized the need for a well-defined, iterative, systematic design process to approach the problem logically. The design team was familiar with two well-known systematic approaches, Hall's seven-step process to systems engineering (Hall, 1969:156) and the space mission design approach described in the Space Mission Analysis and Design (SMAD) textbook (Wertz and Larson, 1992:1).

Hall's systematic process has been a standard systematic approach for almost four decades. This process is well understood and can be applied to many different engineering problems. The Hall method is an iterative seven-step process (refer to Figure 2-1). These steps are: problem definition, value system design, system synthesis, system analysis, optimization, decision-making and implementation (Hall, 1969:157).

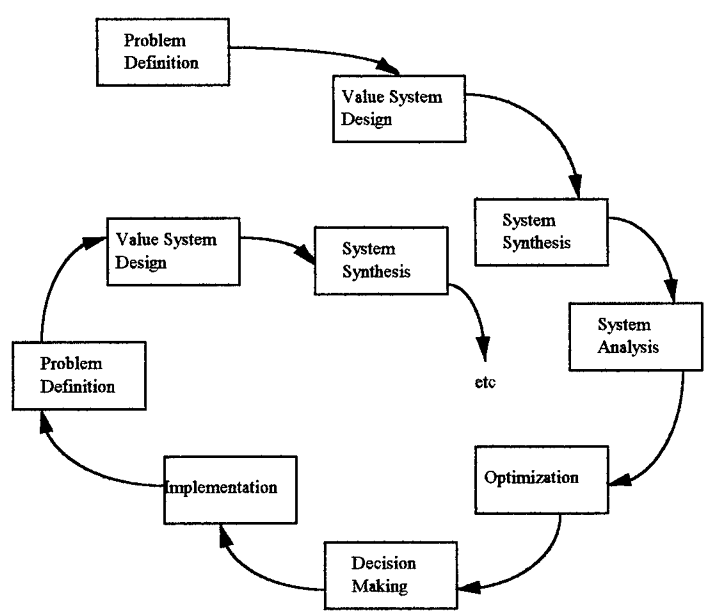


Figure 2-1: Hall's Seven-step Approach

Each step of Hall's approach is influenced by the actions taken in the other steps. The process' iterative nature forces refinement in each step as the process continues. Hall's fundamental framework follows a logical sequence that allows the user to define and constrain the problem, create an evaluation tool using the decision-maker's values, and generate possible solution alternatives. The framework also permits the user to create models and perform simulations as a means of quantifying aspects for each alternative. The quantified values serve as an input into the evaluation tool. Once the basic modeling is accomplished, different aspects of each possible solution are further refined in an attempt to optimize each alternative. Hall's process also allows the user to perform sensitivity analysis on each of the alternatives before the decision-maker is presented with the results of the system evaluation. In the decision-making step, the decision-maker applies his subjective values and risk preferences to select an alternative. With an alternative selected, a plan for implementation is created. The Hall process is complete once an adequate implementation strategy is accepted by the decision-maker.

The SMAD approach is well-known to contemporary satellite designers (Warner, 1996). The SMAD text and the process it describes is a compilation of the first thirty years of satellite design experience. In general terms, the SMAD process can be considered the classic approach to satellite design because the approach is based on the premise that the satellite's mission drives the design of the satellite bus. The SMAD approach is iterative and consists of four broad areas. These broad areas are 1) define objectives, 2) characterize the mission, 3) evaluate the mission, and 4) define requirements (Wertz and Larson, 1992:2).

Table 2-1: Space Mission Analysis and Design Process

Step	Sub-steps
Define Objectives	A. Define broad objectives and constraints B. Estimate quantitative mission needs and requirements
Characterize the Mission	C. Define alternative mission concepts D. Define alternative mission architectures E. Identify system drivers for each F. Characterize mission concepts and architectures
Evaluate the Mission	G. Identify driving requirements H. Evaluate mission utility I. Define mission concept (baseline)
Define Requirements	J. Define system requirements K. Allocate requirements to system elements

The first step in the SMAD process is to define the broad mission objectives and constraints. Additionally, quantified estimates of how well one wants to achieve the broad mission objectives are developed with respect to the needs, constraints, and technology available. These estimates become initial system requirements. A unique feature of the SMAD process is that these quantified estimates are subject to trades as the process continues. Characterizing the mission involves a number of steps. These steps include defining alternative mission concepts and architectures, identifying system drivers for each alternative, and describing in detail what the system is and what it does. Power, weight, and pointing budgets are developed in this step. Evaluating the mission forces the designer to return to the initial system requirements to determine which requirements become driving requirements. Driving requirements are the items principally responsible for determining the cost and level of complexity of the system. Mission utility analysis is also part of this step and this analysis quantifies how well the satellite design meets the system requirements and objectives as a function of design choices. Evaluation of the mission ends by choosing a baseline system design. The SMAD process ends by defining requirements. Broad objectives and constraints are translated into well-defined, specific system requirements. These numerical requirements are allocated to specific components of the overall space mission (Wertz and Larson, 1992:3-90).

The traditional approaches are not suited to designing a satellite bus that will support a variety of missions. This was recognized as the study evolved and initial iterations of the applied processes failed to narrow the scope of the study. Specifically, Hall's approach does not provide an effective, streamlined method for converging on viable satellite design alternatives. Time is wasted performing numerous iterations of the process to achieve the desired focus. Likewise, the SMAD process concentrates too much on using the satellite's payload (mission module) as the key upon which the satellite bus is designed. Consequently, neither of these methods is adequate for designing a generic, standardized satellite bus. A new, customized approach was developed that permitted the team to converge quickly on a satellite bus design without regard to a particular mission module type. The systematic process that was created is a synthesis of the methods described by Hall and SMAD. The process is called the Modsat approach.

Table 2-2: Modsat Systems Approach

Step	Action
Problem Definition	Scope nature of problem
Value System Design	Capture decision maker's needs and goals; create evaluation structure for alternatives
Trade Studies	Link broad design decisions directly to the study's goals and objectives
Modeling	Formulate predictive or descriptive tool(s) to represent activities; analyze various configurations
System Synthesis	Create alternative solution sets
System Analysis	Score each alternative against problem's evaluation structure
Decision Making	Perform sensitivity analysis on solution sets
Implementation	Develop plans for fielding the selected alternative(s)

The iterative approach is comprised of eight steps. The steps, in order, are problem definition, value system design, trade studies, modeling, system synthesis, systems analysis, decision-making, and implementation. The majority of these come directly from Hall's seven-step process. The items that distinguish this approach from Hall's approach are the inclusion of a trade studies step and the reordering of the system synthesis and modeling steps. Additionally, the design team's approach does not include an optimization step. This process distinguishes itself from the SMAD process in two ways. The systematic approach does not commit its focus to the requirements of one mission module as the key factor for satellite bus design. Secondly, the process specifically includes a method for evaluating the merits of each design alternative.

Problem definition is a fundamental first step of any systematic process. The Modsat problem definition step closely follows that of Hall. The purpose of this step is to define and constrain the problem. A result of this step is a succinct statement that identifies the goal and focus of the study. The value of the problem definition step is that it serves as a mechanism to define the system boundaries, identify the system needs, alterables and constraints, and to identify the system actors.

The system boundaries define the environment affecting the system. A distinction can be made between those items contained within an internal environment and those items contained in the external environment. Items within the internal environment are factors that the design team can control. Items that exist in the external environment influence the study but cannot be controlled by the design team. The distinction between the internal and external environment is paramount to understanding the scope and focus of the project. The focus of the study can be narrowed further by performing iterations on the system boundaries. Needs are the fundamental requirements that the decision-maker and users levy on the system and are crucial in determining the broad objective of the study. As is the case in the SMAD process, some needs serve as driving requirements for satellite bus designs. Other needs may be traded off against each other. Alterables are those items that can be influenced or changed by the design team and are contained within the internal environment of the system's boundaries. Constraints are those items that the team cannot control but have a major impact on focusing the study. Problem definition also identifies the actors in the study. Actors are simply the persons/groups who influence the design and evaluation of possible alternatives. Different tools such as concept maps, waterfall diagrams, and interaction matrices can be used to assist in defining the system boundaries, needs, alterables, constraints, and actors.

The value system design step is similar to Hall's respective step. The purpose of this step is to capture the decision-maker's values and goals. Ultimately, these values and goals are used as a means for evaluating the effectiveness of design alternatives. Capturing the decision-maker's values and goals is accomplished by creating an objective hierarchy. Broad values and goals are translated into broad objectives. The broad objectives are decomposed into more specific subobjectives until meaningful measures of effectiveness can be determined. The study's objectives and subobjectives are related to the needs, alterables, and constraints defined in the previous step. As part of defining the study's objectives and measures of effectiveness, major premises and assumptions are explicitly articulated.

Once the objective hierarchy is in place the decision-maker's preferences for each objective have to be incorporated into the structure. It is common to have competing objectives for a problem or study. A score, or weight, is assigned to each objective per level in the hierarchy to capture the importance the decision-maker places on a particular objective. The weights are normalized and the resulting weighted objective hierarchy eventually serves as the evaluation structure for each solution alternative generated in the problem.

The trade studies step is a new and innovative step. This step evolved out of the SMAD process. The purpose of the trade studies step is to make broad design decisions that can be directly linked to the study's goals and objectives. The emphasis is on decisions which can be made without having detailed descriptions of the alternatives. Trade studies serve as an efficient and effective means to narrow the study's scope and provide clearer focus early in the design process. The step is efficient because a manageable study focus can be reached without the need for extra iterations of the process. The step is effective because it reduces the number of possible design alternatives that would have to be evaluated to determine a solution to the study.

The trade studies occur on two levels: the system level and the subsystem level. Trades performed on the system level have broader effects on the design of a satellite bus. These system level trades add definition to the external environment by providing constraints on the system's boundaries. System level trade decisions also impact the trades performed on the subsystem level.

A satellite bus is a system comprised of smaller subsystems. Each subsystem can be designed in a variety of configurations using different qualities and types of components. Some choices can be made independent of choices in other subsystems. A subsystem design decision that is traceable to the study's goals and objectives increases the possibility that system design alternatives will meet the goals of the study. Defining a subsystem configuration or specifying a particular quality or type of component reduces the number of iterations a designer may have to perform to create a viable design alternative. An additional benefit of including a trade studies step early in the process is that it forces team members to focus efforts on gaining insight into subsystem design while simultaneously refining the problem.

Although the trade studies step evolved from SMAD, it differs from the SMAD process in two ways. System level trades in SMAD occur late in the process (the "evaluate the mission" step). This results in the study's focus and system boundaries not being fully defined until late in the process. Subsequently, time is wasted early in the process by identifying the principal cost and performance drivers for each mission concept and mission architectures alternative before the system's boundaries are defined. Secondly, SMAD does not specifically mention that subsystem level trades would occur in the process. It can be inferred that the subsystem level trades would occur after the baseline concept is determined.

The next step in the approach is modeling. Modeling is the development of a descriptive or predictive model representing a set of activities or the entire system in order to allow analysis of alternative configurations of the system (Mosard, 1982:86). The modeling step precedes the system synthesis step, unlike Hall's traditional approach. This reordering of process steps is because the creation and development of satellite bus design alternatives is tedious and complex. Satellite design is an art because many satellite components have to be strategically placed within the confines of a satellite structure to meet stringent heat dissipation, thermal shielding, center of mass, volume, mass, and size constraints. Modeling provides a tool that permits the three-dimensional visualization of the placement and performance of components. Components can be placed, moved, and resized quite easily using a model when compared to physically connecting, disconnecting, or replacing components on an actual satellite. Time and cost savings can be easily realized through the use of a model, especially if design requirements or assumptions change.

In addition, the model builder can take advantage of the decisions that have already been accomplished during the process. Desired subsystem configurations and component selection from the trade studies step can be easily loaded into the model before alternatives are created. If component selection changes, the new information can be easily loaded into the model. Modeling must also be able to quantify the performance characteristics of each design alternative. Different subsystem characteristics can be emulated using mathematical models that can be programmed into the tool. The team's modeling section currently uses the first order estimates and relationships that are found in the SMAD process. Refinements to these relationships can be loaded into the model as the design develops. As a minimum, the quantified performance values must be those values necessary for input into the value system's measures of effectiveness.

The model must allow analysis of alternative configurations of the system. The evaluation structure developed in the value system design is incorporated into this stage of the process. This puts an evaluation structure in place before any design alternatives are generated. With effective use of the modeling tool, it is possible to create new designs and perform evaluation on those designs in a timely fashion.

The system synthesis step is similar to most systematic approach steps for creating alternatives. Accordingly, alternatives can be existing designs, modifications to existing designs, prepackaged designs, or entirely new designs (Pohl, 1995). The difference between the system synthesis step and traditional steps is its placement after the modeling step, for the reasons discussed above.

The systems analysis step follows system synthesis. The purpose of systems analysis is to score each of the design alternatives against the problem's evaluation structure. The problem has been defined and the weighted objective hierarchy is in place. Each alternative's input to the objective hierarchy's measures of effectiveness is evaluated and each solution alternative receives a score commensurate with its performance to the competing objectives.

Decision-making is a step that permits the team to perform sensitivity analysis on the design alternatives. Sensitivity analysis is performed by varying one variable at a time. This variable is usually a weight associated with an objective in the objective hierarchy. The results of the sensitivity analysis provide insight as to how an alternative will perform given different preferences of the decision-maker. Including the sensitivity analysis results allows the decision-maker to make a subjective decision as to which design alternative will meet the goals of the study.

Implementation is the final step of the systematic approach. The purpose of this step is to develop plans for fielding the selected alternative. The plan is presented to the decision-maker and reflects the team's view on how the alternative can be best put to use in the operational setting. It provides recommendations for improvements to the selected alternative and to the associated elements that affect the alternative. The implementation step also addresses the possible architectures in which the alternative can be deployed and it covers the organizational structure necessary to support that architecture.

The approach described above is an innovative approach to satellite bus design. It provides a logical sequence which deliberately allows the design to evolve from one stage to another while documenting the decisions and assumptions made along the way. The method permits continual improvements to the design as the design matures. This approach provides a method for determining if a design alternative is the best design possible by incorporating design decisions made throughout the process. The systematic approach used in this design study provides a holistic view of the problem and allows the team to capture all important aspects affecting the design. This iterative, systematic approach ensures that these aspects are correctly integrated throughout the design process.

3. Team Organization

This combined approach permitted the team to be easily divided into major areas of responsibility within a matrix organization. The team concentrated on three areas: the major steps of the combined Hall/SMAD systematic approach, particular satellite subsystems, and specific areas of research. Each team member's responsibility included taking the lead in charting the group's direction for the steps of the Hall/SMAD (reference Table 3-1) while maintaining a focus on the team's limited time, resources, and budget. Decisions made in one area of the design or a step in the process had to be properly documented and presented to the group to prevent conflicts between satellite subsystems and maintain the direction of the project. Team members also provided the group with the information necessary to understand each satellite subsystem and to realize the influence and impact each subsystem had on the other. The subsystem assignments are listed in Table 3-2. Each member also made contacts with aerospace companies or organizations that had been involved with the development of satellites within the project's weight class (see Table 3-3). Extremely valuable information was gained by examining the successes and failures of other organizations. The following three tables depict the structure of the team's matrix organization.

Table 3-1: System Steps Responsibility Matrix

Steps Of Hall/SMAD Approach	Member(s) Responsible
Problem Definition	From/Krueger
Value System Design	Cokuysal
Trade Studies	All
Modeling/Analysis	Carneal/Ashby
System Synthesis	Buck
Systems Analysis	Carneal/Ashby
Decision Making	Donmez
Implementation	Donmez

NOTE: Robinson served as a "floater" throughout the Hall/SMAD approach.

Table 3-2: Subsystem Expertise Responsibility Matrix

Subsystem Area	Member(s) Responsible
Structures/Mechanisms and Thermal Control	Ashby
Electrical Power Generation and Distribution	Krueger
Attitude Determination and Control	Robinson
Propulsion	Cokuysal
Telemetry, Tracking, and Commanding/Data Handling	Carneal/From
Mission Modules	Buck
Launch Systems/Command, Control and Communications/Operations Concepts	Donmez

Table 3-3: Similar Projects Research Responsibility

Research Area	Member(s) Responsible
Spectrum Astro/MSTI	Cokuysal
TRW and CTA/STEP	Ashby
Lockheed-Martin/Iridium	Carneal
Orbital Sciences Corporation/Pegasus	Donmez
AeroAstro/HETE	From
Ball Aerospace	Buck
Naval Research Laboratory\Clementine	Robinson
Phillips Laboratory/MightySat	Krueger

4. Problem Definition

4.1 Definition

Problem definition was the first step of the systematic approached. The purpose of the problem definition step was to evaluate the proposed problem and establish a succinct problem statement. Defining the problem required careful examination of the sponsor's tasking statement and the factors influencing the proposed problem. Identification of the system's boundary, needs, alterables, constraints, and actors were important to understanding the scope of the problem.

The system's boundary defined those elements of the problem, and its potential solution space, that could or could not be controlled or manipulated by the design team (Athey, 1992:13). Through careful examination and identification of the problem's boundary, the design team determined the factors that influence and affected the problem.

Needs were the driving factors behind the existence of the problem. Needs were referred to as requirements. Without the needs, there would have been no problem. By identifying the needs of the chief decision maker (CDM), the team understood why the problem existed, what the problem was, and what some of the possible solutions to the problem were. Needs also served as a means for measuring the success of potential solutions.

Alterables were those factors the CDM had control over. Identifying those factors provided the team with a method of opening the potential solution space to the problem.

Constraints, on the other hand, were factors that the CDM and design team had no control over. These factors limited the number of potential solutions to the problem.

Actors were the people who had an influence on the problem and the possible alternatives. The most influential actor was the chief decision maker. Capturing and incorporating the decision maker's needs, values, and constraints was paramount to producing the best solution possible. The decision maker provided information necessary to determine the framework by which all possible alternatives were measured.

Gaining insight into satellite design and probing the different aspects of the proposed problem were the main focus of the first iteration. In the second iteration, the team focused its effort on studying and understanding the functions of the satellite subsystems and examining the factors that influence the satellite bus design. This led to a more detailed examination of the tasking statement and the factors that influence the problem.

The resulting problem statement was referred to throughout the design process. This ensured that the team's efforts remained focused.

4.2 Problem Statement

The problem statement reads:

Design a rapidly deployable, tactically oriented, satellite bus to enhance theater operations. This satellite bus is to support missions in the Pegasus and Lockheed-Martin Launch Vehicle (LMLV) weight class.

4.3 Concept Map

A concept map was employed to help define the problem. The concept map provided a graphic representation of the design team's interpretation of the problem. Concept mapping is based on the premise that all knowledge can be represented by relationships between more fundamental concepts (Kramer,1990:652-654). The concept map consists of two primitives: concepts and linkages. As an example, refer to Figure 4-1. The satellite bus is the central concept. The motherboard architecture is another concept. The device which connects the two is the linkage. The linkage in this case is "has a". This method ties the two concepts together into a meaningful structure.

The team developed the concept map in Figure 4-1 by carefully evaluating the concepts and linkages suggested through the chief decision maker's tasking statement. This graphic represented the design team's interpretation of the decision maker's problem.

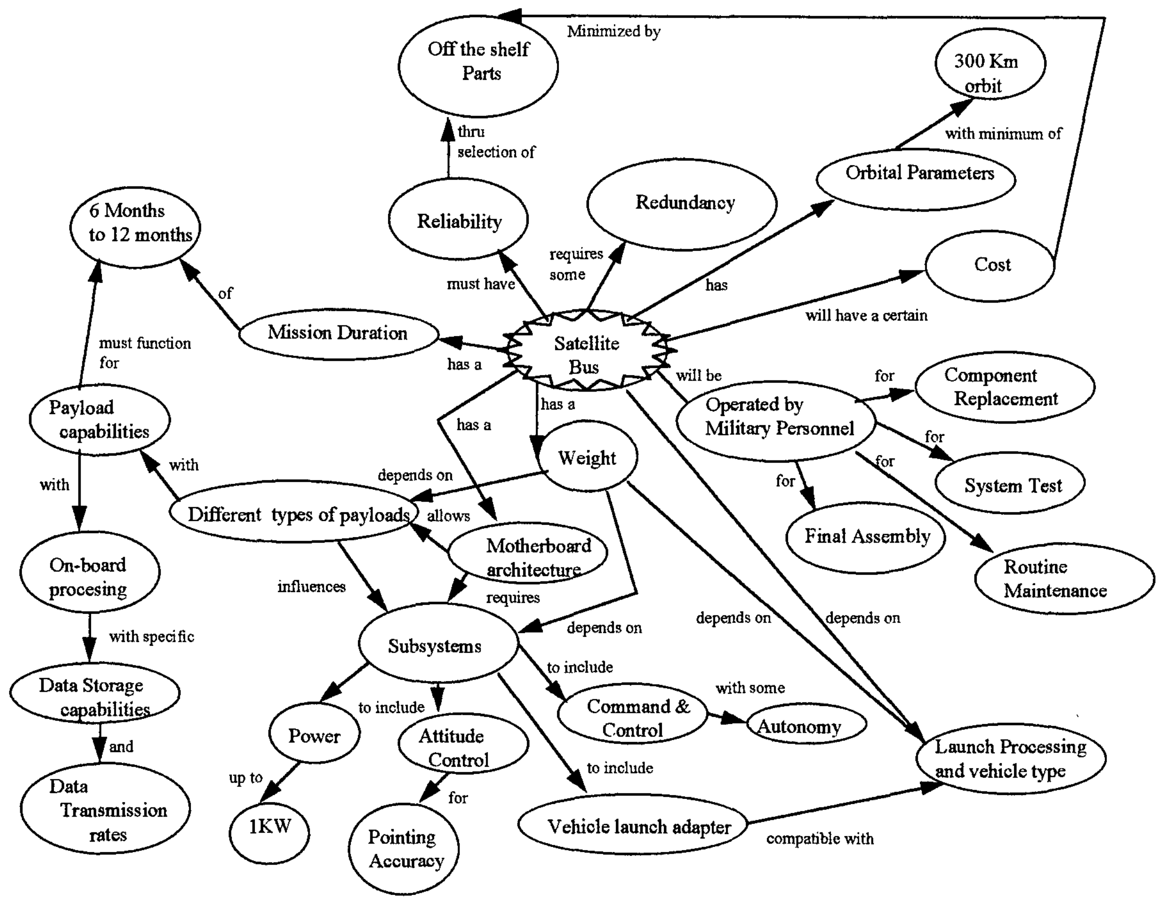


Figure 4-1: Concept Map of Problem

The use of the concept map provided many benefits. It helped the team understand how various factors affected the problem. The team immediately realized that the problem was highly complex. The concept map generated many questions the team needed answered before a concentrated approach to the solution could be given. The team began to question how the operations concept affected a satellite design. How would integration and launch processing occur? Was launch vehicle selection an area to be explored? Another question centered on what type of components were "off-the-shelf" and what reliability did they have. The team also wanted to know what effect orbit selection might have on vehicle life time.

The concept map helped the team identify issues that needed to be considered when examining candidate solutions. The team began to question how much modularity is needed in a satellite bus design and whether modularity is necessarily good. Other questions focused on how much autonomy a satellite bus needs and what reliability is required for a one year life time.

The use of the concept map was only a starting point. The team realized that much research was needed to fully understand the problem. Questions prompted through the use of the concept map were instrumental in identifying areas where research needed to be performed. These areas included researching similar projects, satellite subsystems, launch vehicles, satellite design concepts, command, communication and control architectures, orbital mechanics, and potential mission modules. The concept map offered yet another benefit. This representation of the problem provided a potential mechanism for the team and the decision maker to fully discuss what the problem was and what it was not. The definition of the problem was further enhanced by establishing system boundaries.

4.4 System Boundary

Since the system boundary defined the elements of the problem that could be controlled or manipulated, the team decided that the best way to narrow the focus and scope of the study was to refine this area. The boundaries were divided into two distinct environments; an external environment and an internal environment. Items that existed in the external environment influenced the solution space for the problem, but were not items that the design team could control. These items were considered outside the team's scope with respect to redesigning components or changing concepts of operation. Items contained within the internal environment were aspects that could be controlled by the design team and were subject to trade studies Figure 4-2 provides a graphical representation of the problem's system boundaries.

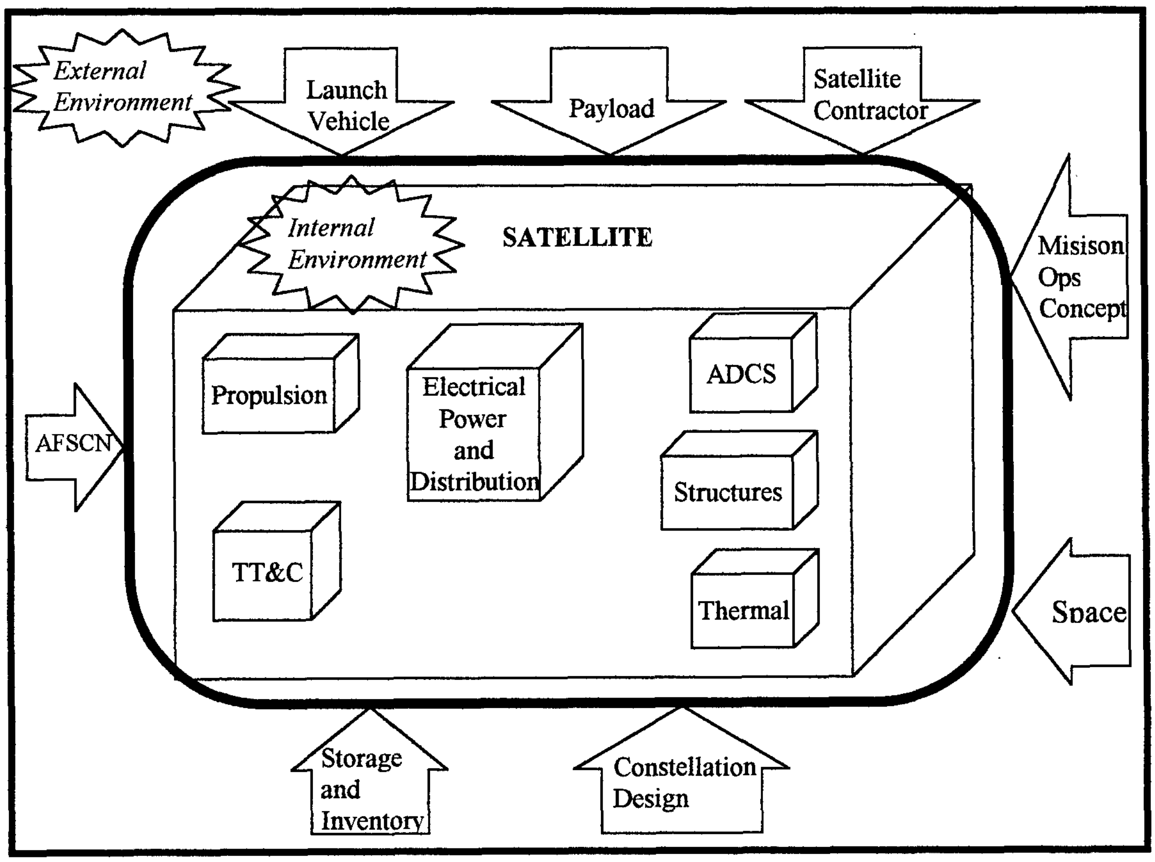


Figure 4-2: System Boundary

The external boundary was comprised of the following items: the launch vehicle, the payload, the satellite contractor, the mission operations concept, the space environment, the constellation design, the storage and inventory concept, and the Air Force Satellite Control Network. Aspects of each element are described below.

- The launch vehicle: The design team examined the system requirements for placing the vehicle into orbit. The team decided to use the Pegasus XL launch vehicle for this design study. Aspects of the launch vehicle that had an influence on the satellite bus design were launch preparation time, mass-to-orbit performance, satellite-to-launch vehicle integration constraints, and fairing constraints. Launch vehicle development and integration of launch vehicle stages were outside the realm of the design team's control and were not subjected to trades or redesign.

- The mission module: A number of different mission modules were examined. These included remote sensing payloads such as multispectral imaging (MSI) systems, synthetic aperture radar (SAR) systems, infrared (IR) systems, and laser designators. Specific mission modules were not designed by the team to be integrated onto the generic bus. The bus design was influenced by the mission module's data storage/health and status requirements, thermal loading, power requirements, mission requirements, and required pointing accuracy's.

- The satellite contractor: A satellite company has a definite influence on the design when it comes to manufacturing the actual satellite bus. Additionally, different companies have different manufacturing processes. Due to the preliminary nature of the design study, the team considered items that would create manufacturing difficulties, but did not perform extensive research into actual satellite manufacturing.

- The mission operations concept: The methods the United States Air Force (USAF) employs to perform its satellite missions had an influence on the satellite design. The design team did not attempt to change or modify the way the USAF does business. However, understanding the constraints and requirements needed to perform satellite operations was a prerequisite for this design effort.

- The space environment: This was the actual operating environment in which the satellite bus would perform its mission. It was important that the design team understood what effects space could have upon the satellite bus. The team had to design the vehicle in such a way that it accounted for the space environment.

- The constellation design: The actual use of the vehicle and constellation deployment was left to the satellite user. The orbit choice and number of satellites to be placed into orbit will be contingent upon the user's need. The team provided a satellite bus design that attempted to maximize its utility to the user.

- The storage and inventory concept: It was taken as an assumption that the satellite and its components would be stored and maintained in an appropriate clean room environment while the satellite was on the ground. Therefore, the team did not design any aspects of the storage and inventory process. However, consideration was given to this storage and inventory process when design alternatives were developed.

- The Air Force Satellite Control Network (AFSCN): Since compatibility with the AFSCN was required by the decision maker, aspects associated with this satellite control network had a major influence on the design. Satellite components such as receivers and transmitters had to operate at Space Ground Link System (SGLS) frequencies. The team made no attempts to redesign any aspect of the AFSCN.

Items contained within the internal environment included the satellite bus subsystems, the launch vehicle interface, and the mission module interface. Operational concepts directly related to the use of the bus were considered within the boundary of the system. The team had the freedom to modify concepts such as on-orbit command and control, mission module and launch integration, and sensor data processing. The team decided that the concept of having two different on-orbit command and control systems to the satellite was well within the scope of the bus design.

4.5 Needs

Revision were made to the needs. This was accomplished because the team wanted a clearer understanding of the problem. The categorized lists created in the first iteration only provided generalized groupings of the needs. A deeper understanding of the problem required more definition be given to the problems needs. It was believed that if a needs definition could be tied to the system's boundaries or the decision maker's views (Rooney, 1996), it would offer a better understanding of the problem. The refined needs definitions are provided below and incorporate the sponsor's views or the system's boundary considerations as appropriate.

- <u>Mass</u>: The satellite design had to be optimized to support as many mission module types as possible. Mission modules with masses between 23 and 114 kilograms had to be supported and fit within the constraints of a Pegasus XL launch vehicle. It was thought that better designs would provide more mass and volume to the mission module yet still supply ample power and interfaces.

- Responsiveness: Possible design alternatives had to consider the rapid launch of a satellite constellation. The bus/mission module combination had to be easily integrated to meet the need for rapid deployability and tactical applications.
- Sensors/mission modules: Different mission requirements were considered; i.e., electro-optical (EO), infra-red (IR), laser designators.
- Pointing accuracy: The question of pointing accuracy had to be considered when trying to achieve 1 meter resolution during an imagery pass of 5-10 minutes per orbit.
- Power: The satellite design had to support peak power requirements up to 1 kilowatts and average power requirements of 300-500 watts.
- Orbital maintenance: The satellite had to operate at a minimal orbital altitude of 300 kilometers for a mean mission duration of 12 months.
- Telemetry, Tracking, and Command: Satellite design alternatives had to be compatible with the AFSCN. Data downlinks had to support near real-time transmission of 1 meter resolution imagery data. Encryption and deception provisions were also required.
- Data storage: Designs had to support on-board storage of up to 100 images.
- Data Processing: Design alternatives had to support minimal on-board processing and data compression algorithms for transmission of images to ground station.

4.6 Alterables and Constraints

Alterables were those elements of the system and its environment that could be controlled by the chief decision maker. The constraints were those items which could not be changed by the decision maker. The team had to manipulate the alterables to achieve a solution, provided the constraints were satisfied. The items contained within the internal environment were considered the alterables. The items in the external environment were constraints on the alternatives for the design study.

4.7 Actors

The actors were all the people and agencies who were involved with some aspect of the system or project. It was important to understand and consider the impact the system has on all actors. The principle actor for any project is the chief decision maker (CDM). The CDM generates the requirements and objectives for the system, and is the approving and implementing authority for the solution. The CDM for this project was LtCol James Rooney of Phillips Laboratory, Kirtland Air Force Base. The design team was comprised of the engineers and analysts who will work together to develop the system. This project's design team consisted of space operations and systems engineering masters candidates at the Air Force Institute of Technology (AFIT). The team was advised by members of AFIT's Department of Aeronautics/Astronautics.

The eventual user of bus design was also an actor in this design study. This was the warfighter who depended on tactical space assets to wage effective information warfare. In order for the warfighter to receive his information, the project's satellites would be commanded and controlled by Air Force space operators. Air Force launch personnel would integrate the satellites to launch vehicles and launch them into low earth orbits. Prior to launch, mission modules would be integrated with their satellite busses by qualified personnel. Prior to being required for missions, ready-to-integrate busses would be stored and maintained. All personnel required to complete these activities were important actors in the development of this system, and their needs were considered.

4.8 Mission Module Overview

The problem involved with the design of a "generic" satellite bus is that, because the bus is to be generic, it cannot be designed to one specific payload or mission. It must support the requirements demanded by all foreseeable missions within the scope of the overall design.

4.8.1 Background and Scope

The payload or mission equipment of any spacecraft is generally considered to be that particular spacecraft's reason for existing. The payload is, after all, comprised of the equipment which the spacecraft owners and users desire to employ (from the space vantage) for the collection or distribution of very specific mission information. Consequently, satellite designs in the past have always focused on this specialized equipment, functionally separating it from the rest of the vehicle (satellite bus). Within this paradigm, payloads tended to be as large, expensive, and/or as powerful or capable as possible. The bus was basically designed and built to support that particular payload (i.e., the bus was built up "around" the payload). Thus, because of the specific nature of a spacecraft's payload equipment, as well as owing to the fact that all satellites are basically manufactured by hand, individual satellites tended (and continue) to be unique. Similarities in design and equipment among satellites of the same constellation or "family" are more numerous, but even these satellites have been and continue to be dissimilar in some areas, due to the addition of features, change of specifications, or flight experience from earlier designs. All of these factors, in addition to the slow historical launch rate, tended to drive up costs. A "vicious circle" of spiraling costs ensued and continues today, driving designers to build fewer, more reliable, more capable, and larger satellites. The larger vehicles compounded the launch availability problem due to the fact that larger boosters became necessary for the larger vehicles -- larger boosters take longer and are more costly to integrate.

Shifting equipment and design focus AWAY from the payload components forces a shift in the spacecraft design paradigm. It focuses on the vehicle itself, the bus, as a starting point for employment of special sensors or other equipment from space. In this paradigm, the payload simply becomes yet another "component" which must be integrated into the vehicle as a whole -- the payload specializes or tailors a standard vehicle to a specific mission or purpose. This paradigm is analogous to a multi-role fighter aircraft being outfitted with a particular weapons load for the performance of a specific mission. The fighter is the "standard vehicle" which can then be used for a variety of missions. This particular paradigm requires the payload designer to produce payload equipment packages which can seamlessly interface with and "take a ride" on a satellite bus which has already been designed (or built) and can provide all the payload support functions necessary.

The aspects of the satellite-to-mission module interface which will be most important to the mission module designer will be mass, volume, power, and data storage budgets available for the mission module. These budgets will provide the mission module designer with limits within which the mission-specific equipment must operate.

Similarly, the most important considerations for the bus design will be the support of those baseline power, mass, stability, pointing, data handling, data storage, and thermal isolation requirements necessary to accommodate all of the baseline mission module types. These types include applications spanning basic electro-optical radiometers, multispectral imagers, LASER/LIDAR systems, and synthetic aperture RADARs. These mission module types were chosen for their diversity and their applicability to tactical space applications. Because of the generic nature of this study, and due to the fact that specifications for military systems (within these categories) are either classified or unavailable at this time, estimates for mass, power, volume, and other specific requirements had to be generated from experience, remote sensing class notes, SMAD, and the few analogous commercial, scientific, and civilian applications available for inclusion. For purposes of this design study, however, which focuses on the design of a specific satellite bus (not the mission modules), lack of specificity of mission module designs will not impact the overall design of the bus. The purpose of a discussion of mission module requirements will provide, in many cases, valuable performance requirements to be met by the specific subsystems of the satellite bus (e.g., pointing accuracy requirements will drive decisions made about attitude control system components). In all cases, extrapolation of estimates was conservatively overestimated in order to provide sufficient design margins.

4.8.2 Specific Mission Module Types

4.8.2.1 Electro-Optical Imaging (EO)

The least expensive, lightest weight, lowest power, and probably the widest used payload type for tactical missions is the simple yet capable, high-resolution camera system. The military utility of this type of imagery dates back to the first days of placing observers in balloons and later placement of cameras in reconnaissance aircraft. The EO mission module is very closely constrained to a Sun-synchronous orbit for optimal orbit selection, due to the radiometric equipment's dependence upon reflected sunlight for illumination of a target.

The following table summarizes some estimated EO mission modules and their characteristics. For the EO mission module, a central wavelength of 0.5 microns is assumed, and ground resolution is calculated based on a 350 km circular orbit.

Table 4-1: Electro-optical (EO) Mission Module Estimations

Aperture (cm)	Diffraction Resolution (mrad/m)	Mass (kg)	Volume (m3)	Power (w)
30	2.03/0.71	23	0.035	7.5
40	1.53/0.53	41	0.082	10.0
50	1.22/0.43	64	0.158	12.5

4.8.2.2 Multispectral Imaging (MSI)

Advances in image processing as well as improvements in detector performance over the past few years have made MSI a high-demand payload. The MSI mission module uses several different arrays of detectors (CCD arrays), each optimized to detect a specific band of EM radiation. Image processing produces simultaneous images of a target area characterized at various regions of the EM spectrum. Intensity levels at specific wavelengths may indicate, through analysis, a particular activity, characteristic, or object within the field of view. By overlaying and comparing the levels of intensity at specific wavelengths from a single target, many characteristics of the target and subsequent target identification may be determined by comparing the received spectra to known spectra (predetermined spectra for specific substances -- a particular type of vehicle paint, for instance). Due to its wider range of detectable radiation bands, the MSI mission module is not as closely constrained to the Sun-synchronous orbit, although this type of orbit is still very advantageous. Specific orbit selections for MSI mission modules will vary, in accordance with varying detector types and specific mission objectives.

Physical characteristics for the MSI mission module are similar, but more massive and more power-hungry, than the EO mission module. MSI payload characteristics will vary according to specific design and performance requirements.

Estimates for MSI mission modules and their vital characteristics are included in the following table. Resolutions are calculated from a 350 km circular orbit.

Table 4-2: Multispectral Imaging (MSI) Mission Module Estimations

Aperture (cm)	NIR (1.5μm) Diffraction Resolution (μrad/m)	MIR (4.0μm) Diffraction Resolution (μrad/m)	LWIR (10.0μm) Diffraction Resolution (μrad/m)	Mass (kg)	Volume (m^3)	Power (w)
30	6.1/2.135	16.3/5.69	40.7/14.23	28.5	0.039	60.0
40	3.75/1.3125	12.2/4.27	30.5/10.68	50.3	0.089	80.0
50	3.0/1.05	9.76/3.42	24.4/8.54	78.7	0.169	100

4.8.2.3 LASER/LIDAR Applications

Using optics similar to the EO package (and in some "functionally dense" mission modules, the VERY same optics as the visible camera/detector), the LASER imaging payload adds a LASER head and power supply (LASER pump) in order to illuminate a target with a specific wavelength of EM radiation. These payloads can produce very accurate three-dimensional imagery, making them well-suited for topographical missions and atmospheric/meteorological (cloud system) observations.

The following table summarizes some estimated LASER/LIDAR mission modules and their characteristics.

Table 4-3: LASER/LIDAR Mission Module Estimations

Aperture (m)	LASER Power (w)	Mass (kg)	Volume (m^3)	Power (w)
30	300	38.7	0.044	318
40	250	56.7	0.089	274

4.8.2.4 Synthetic Aperture Radar (SAR)

By far the payload with possibly the greatest potential tactical "payoff" is the Synthetic Aperture RADAR or SAR mission module, which can produce (through intensive image processing) very high resolution images. SAR mission modules, like LASER-based mission modules, are active sensing systems and, as such, generally require an order of magnitude greater power to operate than passive systems (EO and MSI). Day and night, all-weather operations are possible with the SAR mission module, releasing it from a "most desired" orbit type.

Some estimated SAR mission modules (with stowed antenna -- launch configuration) are summarized in the following table.

Table 4-4: Synthetic Aperture RADAR (SAR) Mission Module Estimations

Antenna Dimensions (m x m)	Mass (kg)	Volume (m^3)	Power (w)
8.0 x 1.5	78.4	0.318	800
10.0 x 2.0	86.5	0.564	450

4.8.3 Generally Specified Mission Module Support Requirements

Though the aforementioned mission module types are varied (not to mention those mission modules which may be further differentiated (based on specialized application) within a given general type), there are certain requirements on the bus which may be specified, allowing components in many of the spacecraft subsystems to be chosen for all candidate designs. The requirements specifiable through mission module consideration include stabilization control, pointing accuracy, attitude knowledge, thermal isolation, operating power, data handling, data storage, and data down-link.

All design requirements which the bus must meet will be driven by the most demanding mission module type in all cases. Furthermore, because of the lack of specificity of mission module designs, those driving design requirements must necessarily be interpreted as "ranges" of values, as opposed to exact quantities. The following table summarizes the requirements necessary for support of all mission module types.

Table 4-5: Estimated Mission Module Support Requirements

Bus Performance Criteria	Mission Module Support Requirement
Pointing Accuracy	0.2-0.1 degrees or better
Attitude Knowledge	0.07-0.05 degrees or better
Data Compression	4:1 minimum
Data Storage Capacity	2Gbytes minimum; modular unit
Data Handling Capacity	150 Mbytes/s or better
Thermal Environment	thermally isolated from mission module
Available Mission Power	peak power from 500-900 watts
Available Mission Launch Mass	120 kg or better
Available Mission Launch Volume	0.6 m^3 or better

5. Value System Design

5.1 Overview

A systems engineering approach to the development of a system considers the values and objectives of the chief decision maker (CDM). The requirements and values expressed by the CDM must be expressed as an organized set of system objectives. This set of objectives should drive all design efforts, and it must serve as the standard by which alternative solutions are evaluated. Often, the established objectives are in conflict; that is, positive performance for one objective may imply negative performance for another. An example of this is the use of cutting-edge technology, which may deliver high performance while admitting higher cost and technological risk. The objectives must be organized in such a way that the engineer may judge alternative solutions against all the objectives, and perform trade-off analyses where necessary.

Value system design translates CDM values into a hierarchy of objectives, where objectives flow down from the top level in a well-structured manner. Each bottom-level objective has a corresponding measure of effectiveness (MOE), by which the performance of that objective is measured. In addition, each objective is weighted, in terms of importance, relative to the other objectives at the same level. Based on the performance of each objective, and the priorities of those objectives, alternative solutions receive a utility score, which can be compared to the scores of other alternatives. In this way, the value system allows the engineer to analyze trade-offs between competing objectives.

Each bottom-level sub-objective has a unique measure of effectiveness. The ideal MOE is a natural scale which can be directly measured or computed, such as speed in meters per second. However, the true MOE is often difficult and impractical to obtain. This is especially true for studies which occur early in the life-cycle of a system, where modeling and testing is limited. In this case, two options are available (Clemen, 1996: 79). The first is to use a proxy measurement. The proxy should be closely related to the objective under consideration. The second option is to "construct an attribute scale for measuring achievement of the objective" (Clemen: 79). This requires the definition of levels of performance for the objective, with levels ranging from best to worst.

Several of the MOEs for this study are actually combinations of proxy measurements and attribute scales. All attribute scales used for this study have six levels, from zero (worst) to five (best). It was felt that six levels provided enough detail to make intelligent judgments.

The MOEs for each bottom level objective are given in section 5.2. The contributing factors that aided in the construction of attribute scales are explained in Vol-II.

Although most of the MOEs are constructed attribute scales, future design efforts for this program must convert these to natural scales wherever possible.

It should be noted that for several of the objectives, the evaluation of the performance of the alternative solutions proved to be challenging. Members of the team combined their knowledge and experience to rate the performance of such objectives.

5.2 Objectives

The team determined that the overall objective of this study is to:

"DEVELOP THE BEST STANDARDIZED BUS FOR A SMALL TACTICAL SATELLITE."

It is important to understand what the words in this objective mean in order to prevent any misinterpretations between team members, advisors and the chief decision maker. The key words are defined below:

1. <u>STANDARDIZED:</u> The most important idea of the study. All other objectives must support this. The idea is that the bus will support many different mission types.
2. <u>THE BEST:</u> Implies an open-minded approach to developing the best system, free from the bias of favored technologies and approaches.
3. <u>SMALL:</u> The satellite must be compatible with a light weight launch vehicle, such as the Pegasus or the Lockheed Martin Launch Vehicle.
4. <u>TACTICAL</u>: The project statement emphasizes tactical missions and a short life.

The set of objectives used for this study is intended to fully capture the values of the CDM. Thus, all of the objectives discussed below were used to guide the development of subsystem and system-level solutions. Throughout the design of the overall system, many trade-offs were performed at the system and subsystem levels, in an effort to form a reasonably sized solution space within which alternative system solutions could be evaluated.

The top-level objectives are shown in Figure 5-1. The following sections explain each objective.

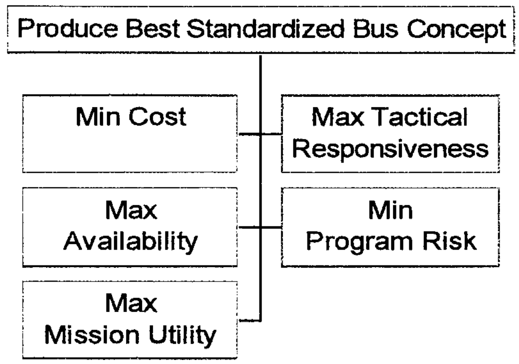


Figure 5-1: Top-level objectives

5.2.1 Minimize Cost

The cost is made up of two main elements. The first element is monetary cost. The performance of each monetary cost objective may not be measured in actual dollars; for some objectives a proxy utility scale will describe the cost. For others, cost estimating relationships (CERs) may be used where such relationships are available. CERs yield a cost in dollars, but all such costs must be stated for the same year (i.e., FY96 dollars). The second main element of cost is time.

There are 9 bottom level objectives and MOEs under the cost objective.

O-1. Minimize Time to Full Rate Production (MOE: attribute scale)

- **Minimize Monetary Cost**

  O-2. Minimize Research, Development, Test and Engineering (RDT&E) Cost (MOE: Cost estimating relationship)

  O-3. Minimize Bus Production Cost (MOE: Cost estimating relationship)

O-4. Minimize Retirement Cost (MOE: Attribute scale)

- **Minimize Operations Cost**

  O-5. Minimize Cost of Telemetry, Tracking, and Commanding (MOE: Attribute scale)

  - **Minimize Cost of Pre-Launch Operations**

    O-6. Minimize Cost of Mission Module Integration and System Test (MOE: Attribute scale)

    O-7. Minimize Cost of Maintenance (MOE: Attribute scale)

    O-8. Minimize Cost of Storage, Handling, and Transportation (MOE: Attribute scale)

    O-9. Minimize Cost of Launch Integration and Test (MOE: Attribute scale)

5.2.2 Maximize Tactical Responsiveness

Modsat is intended for tactical applications, as has been stressed by the CDM. Thus, responsiveness is a primary objective. Modsat satellites must be able to respond quickly to rapidly generated needs and mission requirements. One major driver of responsiveness is the availability of a launch vehicle; however, it was assumed for this study that launch vehicles are continuously available.

The bottom level objectives are:

O-10. Minimize Preparation Time to Launch (MOE: Attribute scale)

O-11. Minimize Data Latency (MOE: Attribute scale)

O-12. Maximize Capability For Tactical Maneuvers (MOE: Attribute scale)

5.2.3 Maximize Availability

A sound system design for Modsat must attempt to maximize on-orbit availability. The bus must endure both natural and man-made hazards. Natural hazards are considered under reliability, while man-made hazards are considered under survivability.

O-13. Maximize Reliability (MOE: Attribute scale)

O-14. Maximize Survivability (MOE: Attribute scale)

5.2.4 Minimize Program Risk

Program risk refers to the potential for elements of the program to fail to come together as planned. Risk can be assessed in the areas of cost, schedule, and performance.

O-15. Minimize Cost Risk (MOE: Attribute scale)

O-16. Minimize Schedule Risk (MOE: Attribute scale)

O-17. Minimize Performance Risk (MOE: Attribute scale)

5.2.5 Maximize Mission Utility

Mission utility refers to the ability of Modsat to accommodate a range of different mission modules. In other words, it is a way of quantifying how well the bus performs its role of being generic and standard. The larger the range of possible missions, the higher the mission utility. This objective was difficult to construct, since it is hard to envision how such utility can be measured. It was decided that mission utility is supported by the various aspects of spacecraft bus performance, such as pointing accuracy and available power. In other words, if more performance capability is built into the bus, more mission types can be accommodated. Thus, several performance sub-objectives were adopted, in addition to the obvious desire to maximize the available weight and volume for the mission module.

O-18. Maximize Pointing Accuracy (MOE: Degrees of pointing accuracy)

O-19. Maximize Data Storage (MOE: Attribute scale)

O-20. Maximize Average Mission Module Power (MOE: Watts of available average power)

O-21. Maximize Allowable Mission Module Weight (MOE: Kilograms of available weight)

O-22. Maximize Adaptability (MOE: Attribute scale)

O-23. Maximize Orbital Accuracy (MOE: Attribute scale)

O-24. Maximize Data Down-link Rate (MOE: Mbytes/sec)

O-25. Maximize Peak Mission Module Power (MOE: Watts of available peak power)

O-26. Maximize Allowable Mission Module Volume(MOE: cm^3 of available volume)

O-27. Minimize Thermal Transfer (MOE: Attribute scale)

5.3 Utility Functions

In order to provide overall utility scores for each competing system solution, all MOEs must be converted to a common utility scale. The team chose a scale from zero to one for convenience, but the endpoints of the scale could be any numbers. The translation from MOE to utility for a given objective is referred to as the utility function of that objective. A utility function is essentially a model that represents the preferences of the CDM for an objective (Clemen: 473). Feedback from the CDM enables the analyst to determine how much utility to assign a given level of performance.

Ideally, each objective would have a unique utility function that reflects the risk attitude of the CDM. However, in the absence of participation from the CDM, the team took a generalized approach. The validity of the results of this study could be improved with feedback from the CDM with regard to utility functions.

5.4 Priority Weighting of The Objectives

The objectives on the same level in the hierarchy were assigned priority weights, as shown in Figure 5-2. These weights were determined based on the results of a preference chart survey. This survey was completed by the CDM, members of the team, and other subject matter experts, with feedback from the CDM being more heavily weighted. The actual survey is attached as Appendix A.

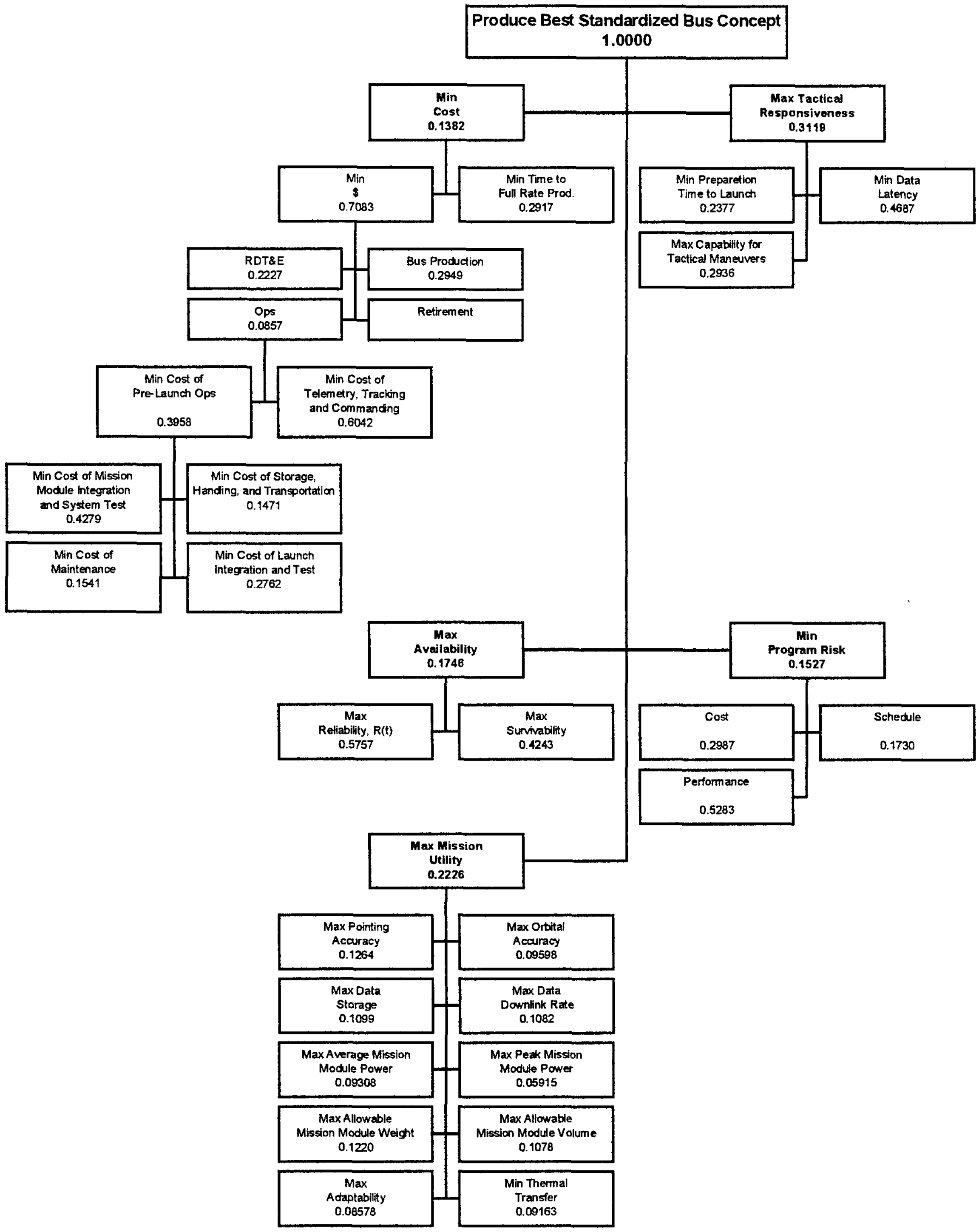


Figure 5-2: Phase two objective hierarchy

From Figure 5-2, it is clear which objectives are more important than others in the current value system. It should be noted that this set of weights is a reflection of personal opinions, solicited at a certain time and under a given set of technological, political, and economic conditions. Major changes in any of these areas could cause the relative priorities to change. This potential for change does not present a significant problem, since the overall scoring function (see section 5.5) can easily be re-calculated with new weights. In fact, a sensitivity analysis was performed on the alternative solutions by varying the weights of each of the top-level objectives.

A comparison of the priorities of the top-level objectives is shown in Figure 5-3. Tactical responsiveness is the highest-rated objective, while mission utility has the second highest priority. It is interesting to note that the cost objective received the lowest rating, as opposed to its prominence in most system studies.

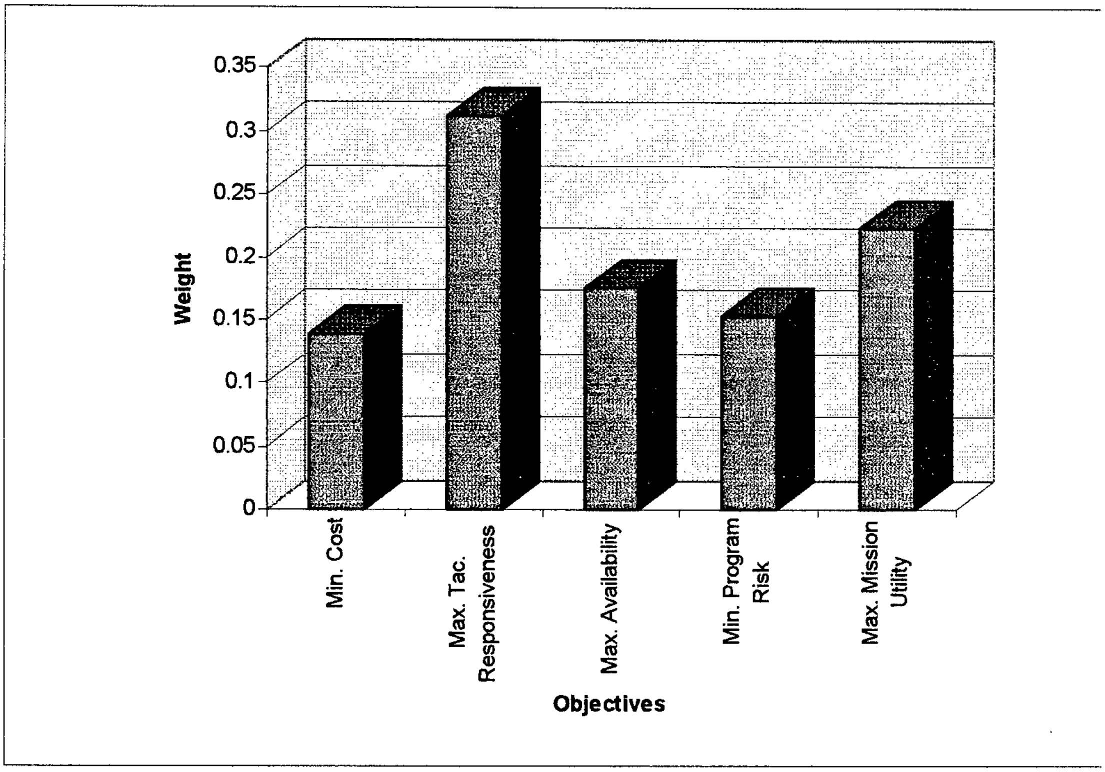


Figure 5-3: Top-level objective weights

5.5 Scoring Function

The utility values and objective weights must be combined to form an overall utility function. This function yields an overall utility score for a given alternative solution, based on its performance of each objective and the relative importance of those objectives. Alternative system solutions can be compared by their overall utility scores.

Since this study was intended to produce concept-exploration design characteristics, and not detailed design recommendations, the team decided to avoid modeling the interaction among the objective attributes.

5.6 Flexibility of the Value System

Since the value system drives all design efforts, changes in any of the elements of the value system can lead to different results. The objectives, weights, and utility functions used for this study could be modified upon further engineering efforts. It was not the intent of the team to create a rigid, unchanging value structure, but rather to create a robust framework within which intelligent decisions could be made. Now that the framework exists, it can be modified according to the changing desires of the CDM and the analysts who conduct further research on this topic.

6. Tradeoffs

6.1 System Level

Many important tradeoffs were analyzed at the system and subsystem level, and several critical design decisions were made. The purpose of this section of the report is to document these tradeoffs and decisions. Throughout the system study, the team has encountered many variables and options. Some of these have remained as variables, to be specified as elements of alternative system solutions for judgment against the value system criteria. However, many design decisions were made during this study, in order to narrow the solution space to a reasonable set of alternatives. Thus, the Modsat bus concept gained its shape throughout the study, as key decisions have built on each other. These decisions were made after performing tradeoffs between the alternatives, judging each against the Modsat objectives and constraints.

Much work was done at the subsystem level, and this is documented in section 4.5. This section discusses the system level tradeoffs.

6.1.1 One Satellite Per Launch Vehicle

Rather than attempt to design very small satellite buses for a multiple-payload launch package, the team chose to focus on one vehicle that will fill the payload bay of the chosen launch vehicle (LV). A key objective of the study is to allow for as much mission module capability as possible per bus. It would be inconsistent with this objective to force mission module designers to integrate with a microsat, or to make them spread their mission capability over a multiple-satellite configuration.

6.1.2 Launch Vehicle

The team originally chose the Pegasus XL, from Orbital Sciences Corporation (OSC). This LV is common for small low earth orbit (LEO) satellites. It is attractive for several reasons, among them being the ability to respond rapidly to mission needs and the flexibility of launch integration and location.

6.1.3 Basic Spacecraft Configuration

Modsat requires three axes of pointing control, since it has articulated solar arrays (see section 6.5.6) and must accommodate nadir pointing mission modules. According to aerospace consultant Emery Reeves,

> "The spacecraft configuration must provide two axes of control for each item that is to be pointed. The spacecraft body has three axes so the body alone can satisfy one pointing requirement; for instance, one body axis (i.e., the yaw axis) can be pointed toward nadir by control about the other two axes (roll and pitch). It two items are to be pointed, then the spacecraft must be configured with at least one articulated joint between the two items. For illustration, a body mounted antenna can be pointed nadir by controlling two axes of body attitude. A solar array can then simultaneously be pointed toward the Sun by using the third body axis and providing a single solar array drive to control the solar array attitude relative to the body...3-axis-controlled spacecraft generally use articulated panels" (Reeves, 1992:297).

Although two-axis systems are usually cheaper, lighter, and less complex than three-axis systems (Reeves, 1992:305), it was determined that the pointing requirements of both the solar arrays and the nadir pointing mission modules mandate the use of three-axis control.

6.1.4 Data Delivery Architecture

The users in the field would like to have their mission data as rapidly as possible. By requirement, Modsat must have S-band SGLS (space to ground-link subsystem) antennas which are compatible with the AFSCN. The SGLS link is limited to a maximum data rate of 1.024 Mbits/sec. Any tactical, warfighting satellite would benefit greatly from having its own high data rate downlink antenna(s), in addition to the SGLS antennas. Thus, the design team acknowledged that almost all Modsat satellites would require a high data rate antenna in addition to the required SGLS antennas. It was decided by the team to avoid selecting and integrating a particular type of antenna. Rather, the mission planners will have the flexibility to choose antennas that suit their needs. Thus, all Modsat bus designs accounted for the placement of a high data rate package, whether in the bus itself (as a standardized add-on) or as part of the mission module.

6.1.5 Adaptability and Modularity

As mentioned in the Problem Definition section, the CDM desires Modsat to have a flexible, adaptable design, such that additional capability can inserted, or excess capability can be removed, depending on the needs of each mission module. A modular approach will be used, whereby all busses are manufactured with all foreseeable component options attached via removable interfaces. Of course, center of mass and attitude control constraints may preclude the removal of some excess components, depending on the characteristics of the mission module in question.

6.1.6 Maximum Capability

The Modsat bus must accommodate a wide variety of mission modules, and must therefore cater to the most demanding customers. This implies that there will be excess capability on many missions. Some of this may be mitigated by the modular architecture discussed above, but certainly not all excess capability can be removed. Some built-in excess is unavoidable with a mandated, standardized interface.

6.1.7 On-Board Propulsion

Magnetic torquers are often used in place of the bulkier propellant-thruster configurations to reduce spacecraft momentum. However, magnetic torquers cannot perform ΔV maneuvers (Everett, 1996). In section 6.4, it will be shown that Modsat requires ΔV capability to maintain its altitude at LEO. Moreover, thrusters are required to perform tactical repositioning ΔV maneuvers, which may very well be required as part of Modsat's tactical mission profile (see Problem Definition).

6.1.8 Data Processing

Most space sensor applications require some degree of processing of the mission data, prior to its transmission to the ground. In any given custom-built satellite, it is possible for such data to be handled by the main spacecraft processor, along with all of its other processing functions. However, each mission type has its own very unique processing requirements, and it would be impossible to satisfy all with one processor. In fact, many instruments have their own mini-computers. Thus, the team decided that mission data processing must be performed by the mission module. However, data storage will be available in the bus processor, as stated in the requirements.

6.1.9 The Bus-To-Mission Module Interface

Another key part of this study is the specification of a standardized interface between the bus and all mission modules. This very issue has been addressed by the Aerospace Corporation, and the results are documented in "An Approach To Rapid Payload Integration: The Spacecraft-To-Payload Interface Guideline (SPIG), Version 1" (Aerospace Corporation, 1996). According to this document, "the SPIG is intended to serve as a core building block on which the payload-to-spacecraft interface can be designed."

In the production of the SPIG, much systems engineering work has been accomplished to suggest an optimum standardized interface. The team decided to use the SPIG interface in lieu of designing their own interface.

6.2 Reliability Analysis

This study was undertaken to determine, at a minimum, an optimal starting point for the cost vs. reliability tradeoff. Fault tolerance via three different subsystem redundancy levels (single-string, double-string, triple-string), and fault avoidance via three different component quality classes (commercial-grade, Class B, Class S) were compared. Educated assumptions were made regarding the relative costs of commercial, Class B, and Class S components. The cost assumptions were changed twice to determine the effect on the optimum cost/reliability option. A baseline cost of $30M, calculated from the Modsat cost model, was used in the study. The failure data used was extracted from a study of over 300 spacecraft failures from the early 1960s to 1984, dividing failure rates by an "improvement factor" to account for advances in technology.

The study showed that the optimum point to begin a detailed cost vs. reliability trade study was with a Class B, single-string spacecraft. Varying the cost assumption did not significantly affect the result. Double and triple-string spacecraft with commercial parts also fared well, but when the weight penalty associated with redundancy was considered, Class B/single-string stands apart. Optimality was determined by fitting a curve to the data and looking for the point(s) closest to the "knee" of the curve. An even more optimal starting point might be achieved by designing a spacecraft that is Class B/Class S and/or redundant where warranted (e.g. the Telemetry, Tracking, and Commanding subsystem). The availability of Class S and B components in an era of decreased government space and defense-related cuts, as well as the rapid growth in the quality of commercial-oriented components, was not quantified in this study.

6.3 Launch Vehicle Considerations

The sponsor's guidance required the satellite bus to be within the Pegasus and Lockheed Martin Launch Vehicle (LMLV) weight class (Rooney, 1996). Orbital Science Corporation's Pegasus XL was chosen as the baseline LV, based on its success and experience. The air-launched nature of the Pegasus offers a tactical advantage over other conventional ground-based LVs. The capability of being rapidly deployed and launched into any inclined Low Earth Orbits (LEO) from any latitude and longitude is another distinct advantage of the Pegasus. More specifically, the Pegasus XL was chosen over the standard Pegasus vehicle because the XL version offers more mass-to-orbit performance.

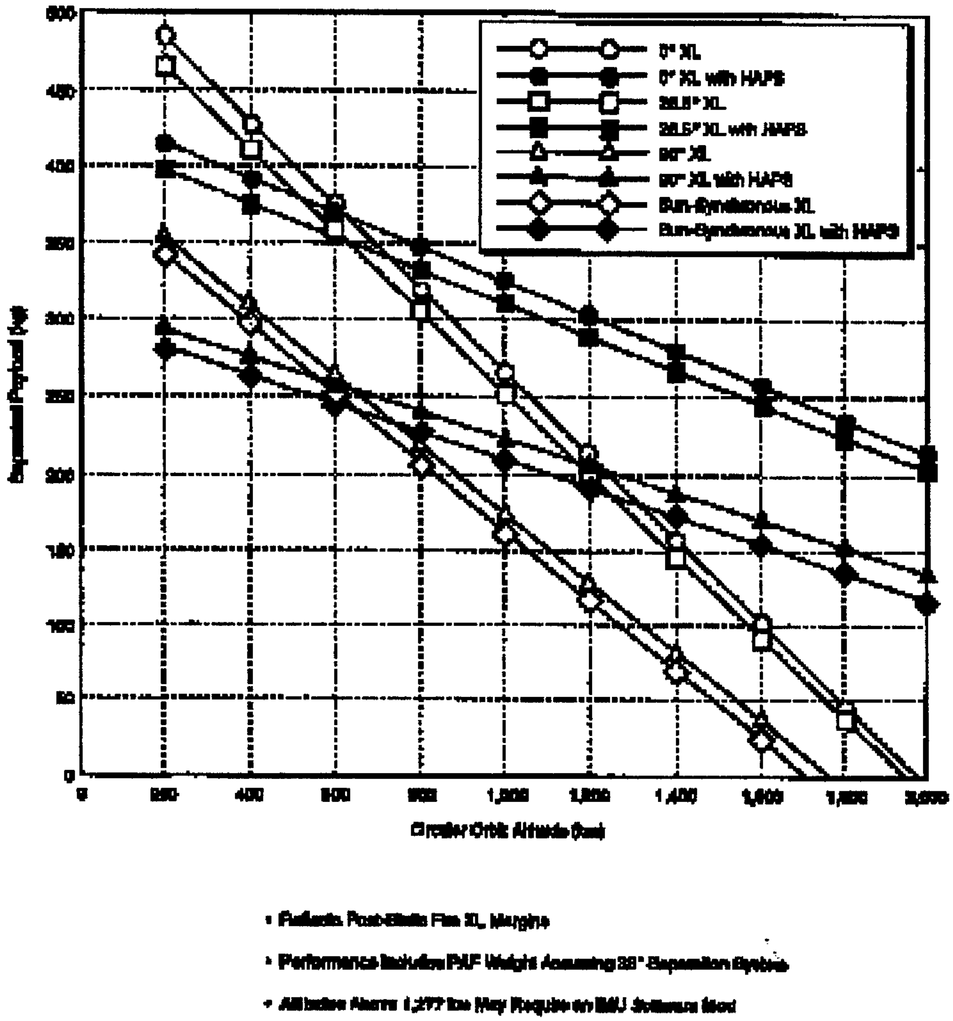
6.3.1 Pegasus XL

The Pegasus XL is a winged, three-stage, solid rocket booster that weighs approximately 22,680 kg (50,000 lbm) and measures 16.9 m (55.4 ft) in length and 1.27 m (50 in) in diameter. The rocket is lifted by a carrier aircraft, usually a Lockheed L-1011, to a level flight condition about 11,580 m (38.000 ft) and Mach 0.79 before it is released for launch (Orbital Science Corporation, 1993:2-1).

There are two major Pegasus XL features that impose restrictions on potential satellite bus designs. One is the Pegasus XL mass-to-orbit performance capability; the other is the booster's payload fairing dimensions.

6.3.2 Mass-to-Orbit Performance

An important characteristic of any launch vehicle is the amount of mass it can place into orbit. The mass that the Pegasus XL can deliver depends on the altitude and inclination of the desired orbit, and whether the booster uses an optional Hydrazine Auxiliary Propulsion System (HAPS, explained below). The mass-to-orbit performance capability for the Pegasus XL is provided in Figure 6-4.

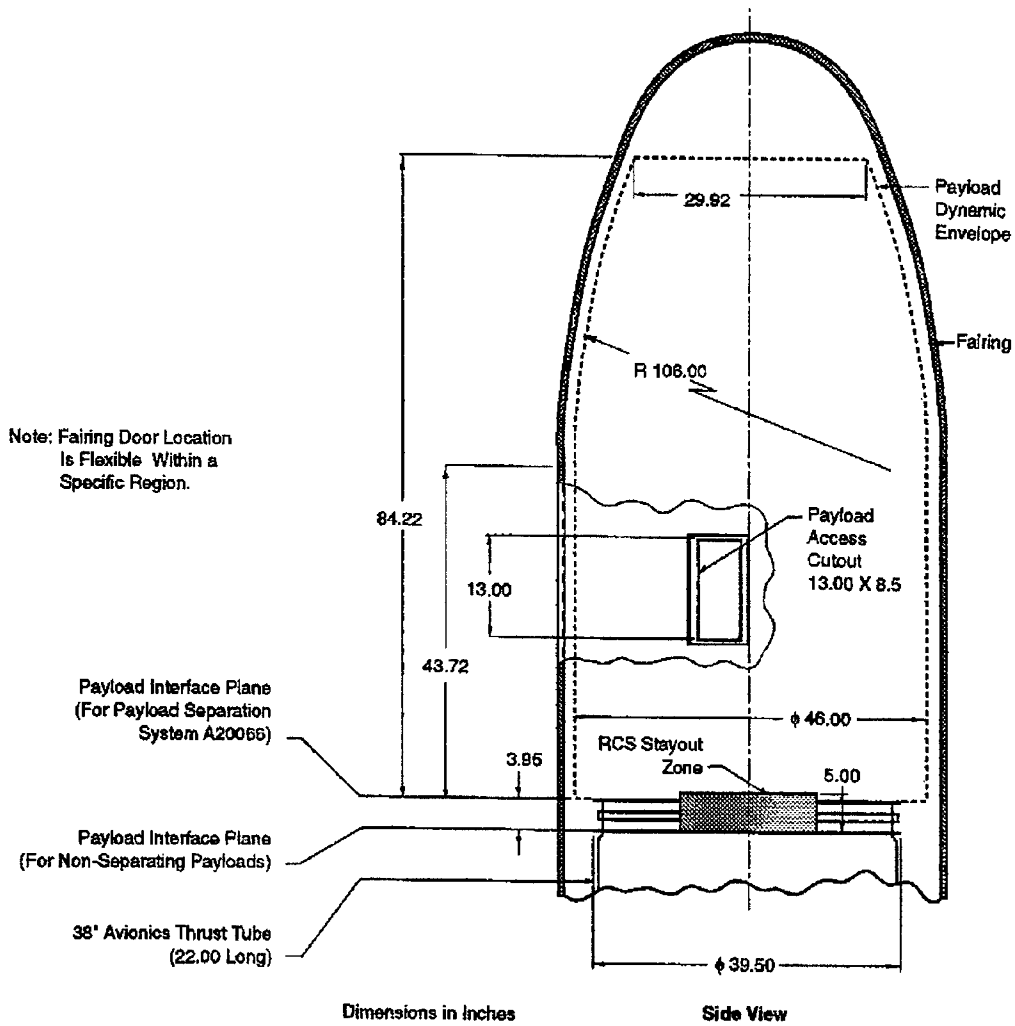


Source: OSC, 1993:3-3

Figure 6-4: Pegasus XL Performance Capability

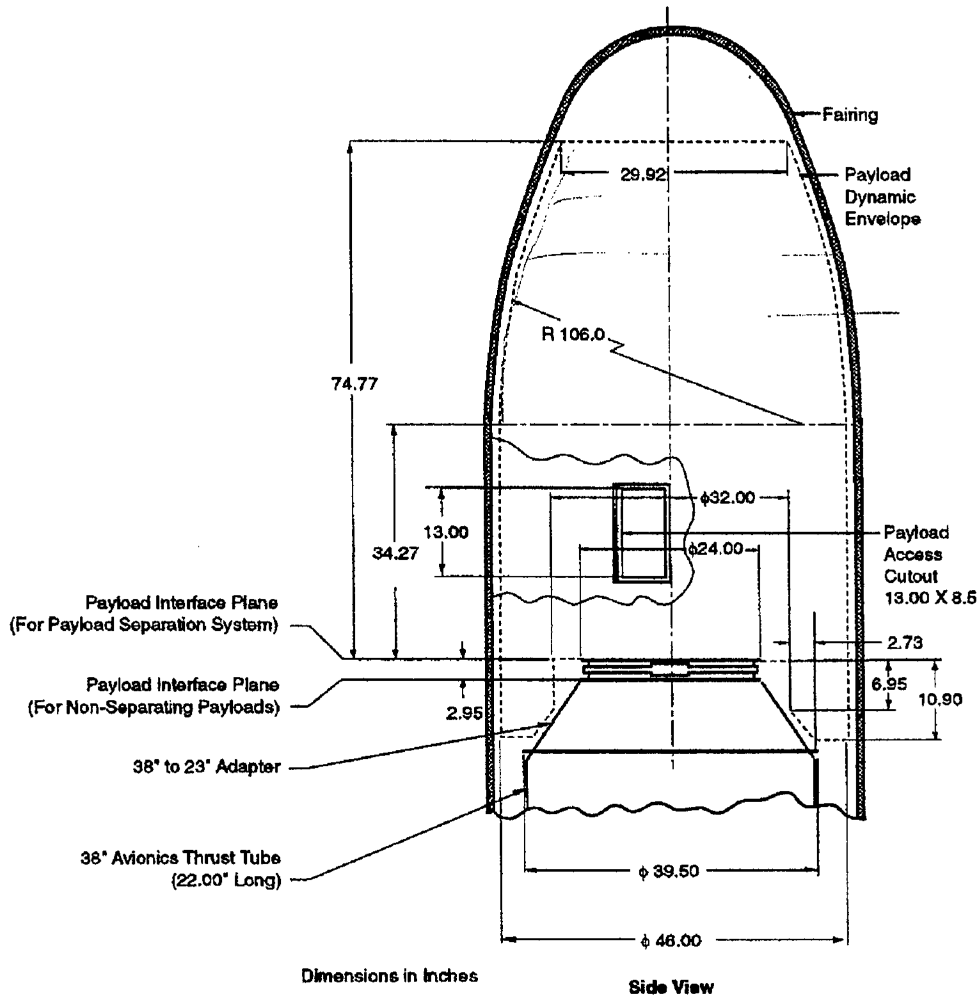
6.3.3 Payload Fairing

Pegasus XL offers two payload interfaces: a 38 inch diameter interface plate and a 23 inch diameter interface plate. Both of these interface plates can be used with or without HAPS. Figure 6-5 shows the payload fairing with the 38 inch interface. If this configuration is used with the optional HAPS, the spacecraft design will lose 14.65 inches from the available 84.22 inch height. Figure 6-6 depicts the 23 inch configuration without HAPS. When the HAPS is used with the 23 inch interface, 4.2 inches are deducted from the 70.57 inch height.



Source: OSC, 1993:5-3

Figure 6-5: Payload Fairing with 38 inch Interface



Source: OSC, 1993:5-4

Figure 6-6: Payload Fairing with 23 inch Interface

6.3.4 Hydrazine Auxiliary Propulsion System

The purpose of the HAPS is twofold. It improves orbital injection accuracy and increases mass-to-orbit capability for satellites placed into altitudes greater than approximately 600 kilometers. The inclusion of a HAPS adds weight to the LV and mandates additional restrictions on the height, mass and volume of the spacecraft, in a trade for added accuracy and/or higher orbit insertion. It was decided that the vast majority, if not all, of the Modsat missions would not require orbital altitudes above 600 kilometers. Additionally, orbital insertion accuracy is mission specific and depends on mass, targeted orbit, and the particular guidance strategy adopted for the mission (OSC, 1993:3-5). Thus, the team decided to exclude the HAPS configuration from consideration.

6.3.5 Other Considerations

Many other characteristics of the Pegasus XL influenced and constrained the design effort, including:

- Spacecraft center of mass location constraints
- Spacecraft stiffness/vibrational frequency constraints
- Spacecraft mass limitations due to critical shear stress at the LV interface
- Axial and lateral loads during launch

To ensure that the design alternatives do not violate the Pegasus XL constraints, launch vehicle compatibility testing was incorporated into the Modsat model through 3-D surface rendering visualization (see Volume III).

6.4 Baseline Orbit and Allowable Launch Mass

6.4.1 Baseline Orbit-Type

Section 6.3 demonstrated that sun-synchronous orbits are the most weight restrictive orbits for a launch vehicle. Many Modsat missions may require a sun-synchronous orbit. Therefore, the bus should be light enough to allow for such missions. But in order to establish a baseline launch mass, a baseline orbital altitude must be determined. This altitude will not be dictated to mission planners; in reality, they have the latitude to select an appropriate orbit, provided their spacecraft meet the launch mass constraint.

Nevertheless, the team needed a maximum mass limit for design purposes. Recall from Value System Design that one of the measures of effectiveness for this study is allowable mission module mass. For the baseline orbital altitude, this mass is the allowable launch mass less the mass of the bus. The challenge was to find the optimum sun-synchronous orbit for design, and therefore, the spacecraft mass limit. The optimum orbit is defined as that which maximizes the allowable launch mass.

6.4.2 Mass to Altitude Tradeoff

The mass-to-orbit performance of a launch vehicle decreases as the orbital altitude increases. Thus, lower altitudes are desirable for large payloads. However, as the orbital altitude decreases, the requirement for ΔV (thrust) to maintain altitude increases due to increased atmospheric drag.

Since ΔV is provided through the use of propellant, the amount of propellant on-board the spacecraft is proportional the ΔV required to maintain altitude. Thus, this analysis seeks to maximize the allowable launch mass for a sun-synchronous orbit, less the mass of the propellant reserved for altitude maintenance. Analysis shows that this mass increases with altitude, until 350 km. Beyond this, the ΔV requirement decreases but the allowable launch mass decreases dramatically. Thus, the baseline design altitude was chosen to be 350 km. At this altitude, approximately 308 kg can be launched into sun-synchronous orbit.

6.5 Subsystem Tradeoffs

Tradeoff analyses were performed at the subsystem level, in order to further narrow the solution space. The subsystems of Modsat are described below:

Table 6-1: Spacecraft Subsystems

Subsystem	Principal Functions	Other Names
Attitude Determination and Control System (ADCS)	Provides determination and control of attitude and orbit position, plus pointing of spacecraft and appendages	Attitude Control System (ACS), Guidance, Navigation and Control (GN&C) System, Control System
Propulsion	Provides thrust to adjust orbit and attitude, and to manage angular momentum	Reaction Control System (RCS)
Structures and Mechanisms	Provides support structure, booster adapter, and moving parts	Structure Subsystem
Thermal Control	Maintains equipment within allowed temperature ranges	Environmental Control System
Telemetry, Tracking and Command (TT&C)	Communicates with ground and other spacecraft; spacecraft tracking	Communication (Comm)
Electrical Power System (EPS)	Generates, stores, regulates, and distributes electric power	Power

Source: Reeves, 1992:287

6.5.1 Attitude Determination and Control

The purpose of the Attitude Determination and Control System (ADCS) is to (1) stabilize the spacecraft against both external and internal torques that would disturb its desired orientation, and (2) maneuver and point the spacecraft in response to commands. An ADCS must meet several stability and agility requirements, including pointing accuracy and satellite slewing rate. An ADCS must operate in several control modes, each of which is employed during the various phases of the satellite's mission, each having different requirements. Since normal operations and satellite slewing control modes are the most common, they tend to be the requirements drivers. Defining the various control modes is the first step in designing an ADCS.

The second step is selecting the particular type of the attitude control method. Control methods range from passive to active, spinning to three-axis stabilized, momentum-biased via constantly spinning rotors internal to the spacecraft, to zero-momentum biased (no spinning rotors). Modsat employs a three-axis stabilized system, which offers the greatest amount of stability.

The next step in ADCS design is to quantify the disturbing torques the spacecraft will be subject to. Internal torques can be controlled through careful design. External torques must be quantified and compensated for. External torques generally have four sources: gravity-gradients, magnetic effects due to the Earth's magnetic field, solar radiation, and aerodynamics. At the low altitudes Modsat will operate in, disturbances due to aerodynamics outweigh the others by two to three orders of magnitude. Thus Modsat's ADCS is designed primarily against aerodynamics-based disturbances.

Next, ADCS hardware must be selected. This includes attitude sensors (Sun sensors, Earth sensors, star sensors, inertial measurement units, magnetometers), and actuators (reaction wheels, momentum wheels, control-moment gyroscopes, magnetic torquers, thrusters). The sensor suite chosen for Modsat was an Earth sensor, a star sensor, and an IMU. This combination best satisfied the requirements of the different control modes. The actuators selected were thrusters and reaction wheels, which were best suited for Modsat's requirements and operating environment.

The final step in ADCS design is the design of the control algorithms that tie the sensors, actuators, and user-commands together. Although detailed control-law development requires a detailed knowledge of the satellite's predicted dynamic behavior (which itself requires detailed knowledge of the satellite's overall design), the designer can estimate the amount of control authority needed by considering the estimated worst-case disturbance together with the spacecraft's pointing accuracy and stability requirements.

6.5.2 Propulsion

The team did not set out to design new concepts in spacecraft propulsion, since a major emphasis has been placed on the use of competitively priced, proven technology. Instead, the design effort was limited to making use of currently available, off-the-shelf technology.

The Modsat propulsion subsystem performs the following functions:

- Orbital corrections.
- Tactical re-deployment.
- Acquisition of Sun, Earth, or star.
- On-orbit back-up mode control with 3-axis stabilization.
- Momentum management.
- 3-axis control during ΔV.

6.5.2.1 Propulsion Options For Standard Satellite Bus Alternatives

Propulsion subsystem options for Modsat were modeled and chosen with the aid of the Modsat computer model. The primary alterables for the propulsion subsystem are shown in Table 6-2. The design and placement of the Modsat propulsion subsystem is constrained primarily by the volume and weight limitations of the Pegasus Launch Vehicle.

Table 6-2: Propulsion alterables

Alterable Elements	Possible Options	
Propulsion Type	Liquid mono-propellant	Liquid bi-propellant
Pressure System	Blow-down	Regulated
Propellant type	Many	
Shape of Tanks	Spherical	Cylindrical

6.5.2.2 Propulsion System Decisions

The following decisions were made regarding the elements and configuration of the propulsion subsystem.

1. <u>The entire propulsion subsystem should reside on the bottom level of the satellite bus</u>.

- Launch vehicle considerations deem that the center of gravity of the spacecraft be as low as possible on the LV/spacecraft z-axis.
- Since the specific mission modules that will be flown on Modsat are unknown, it is wise to keep any thrusters well below the bus-to-mission module interface.
- The effects of thruster exhaust plume and other possible contamination from the propulsion system are reduced by locating it on a designated bottom plate.

- The propulsion subsystem does not fit in well with the modular nature of the bus, with its removable components and structural assemblies. It would be extremely difficult to locate any of the big components and systems anywhere other than on a fixed plate at the bottom of the bus.
- The weight of pipes, vanes, valves, and the other propulsion plumbing equipment is reduced by consolidating the whole subsystem in one location.

2. <u>The propellant tanks should have a cylindrical shape</u>. Cylindrical tanks maximize the use of the area on the bottom plate, and leave more volume to the other parts and pieces.
3. <u>The mono-propellant configuration was chosen as the Modsat standard</u>.

- Simplicity, cost, and reliability advantages over the bi-propellant configuration.
- No significant difference in required tank volume between mono and bi-propellant configurations.

4. <u>The pressure feed system should be a blow-down system</u>.

- Modsat does not require the capability for high-level, long-duration thrust. A low-thrust, small propulsion system is adequate.
- Since blow-down systems do not need regulators (and sometimes filters), they are simple and reliable (Sutton, 1992:326).

5. <u>Propellant: hydrazine</u>
6. <u>Storage system: positive expulsion</u>
7. <u>Commercial off-the-shelf thrusters will be used</u>.

Thus, all of the major system alternatives for this study were designed with a mono-propellant (24%HN and 76% N_2H_4 blend), blow-down pressure fed system with cylindrical tanks.

6.5.2.3 Propulsion Alternatives

After making the design decisions discussed above, the remaining propulsion alterable is the amount of propellant, which dictates the size of the cylindrical tanks.

Although the mission requirements are uncertain, the idea behind the propellant alterable was to provide different amounts of ΔV capability. This capability must be divided among the spacecraft ΔV budget, which accounts for tactical maneuver capability, mission lifetime, retirement, and orbital accuracy. For a given amount of propellant, this budget must be managed by the user of Modsat.

The amount of ΔV required to maintain the baseline orbital altitude or 350 km is 62.2 m/s per year. At this altitude, the amount of Δ V required to make a one degree plane change is 134.3 m/s (Sackheim and others, 1992: back cover). It was assumed that momentum management will require 35-50 m/s of ΔV per year. The chosen propulsion alternatives for this study are shown in Table 6-3.

Table 6-3: Propulsion system alternatives

Alternative	Altitude Keeping (m/s)	Momentum Dumping (m/s)	Inclination Change (m/s) / Degree	Total Δ V (m/s)
1	62.2	37.80	------ / 0	100
2	62.2	36.35	201.45 / 1.5	300
3	62.2	52.05	335.75 / 2.5	450

Although the placement of thrusters was fixed, for each alternative the propellant tanks are located as near as possible to the center, in order to minimize the effect of diminishing propellant.

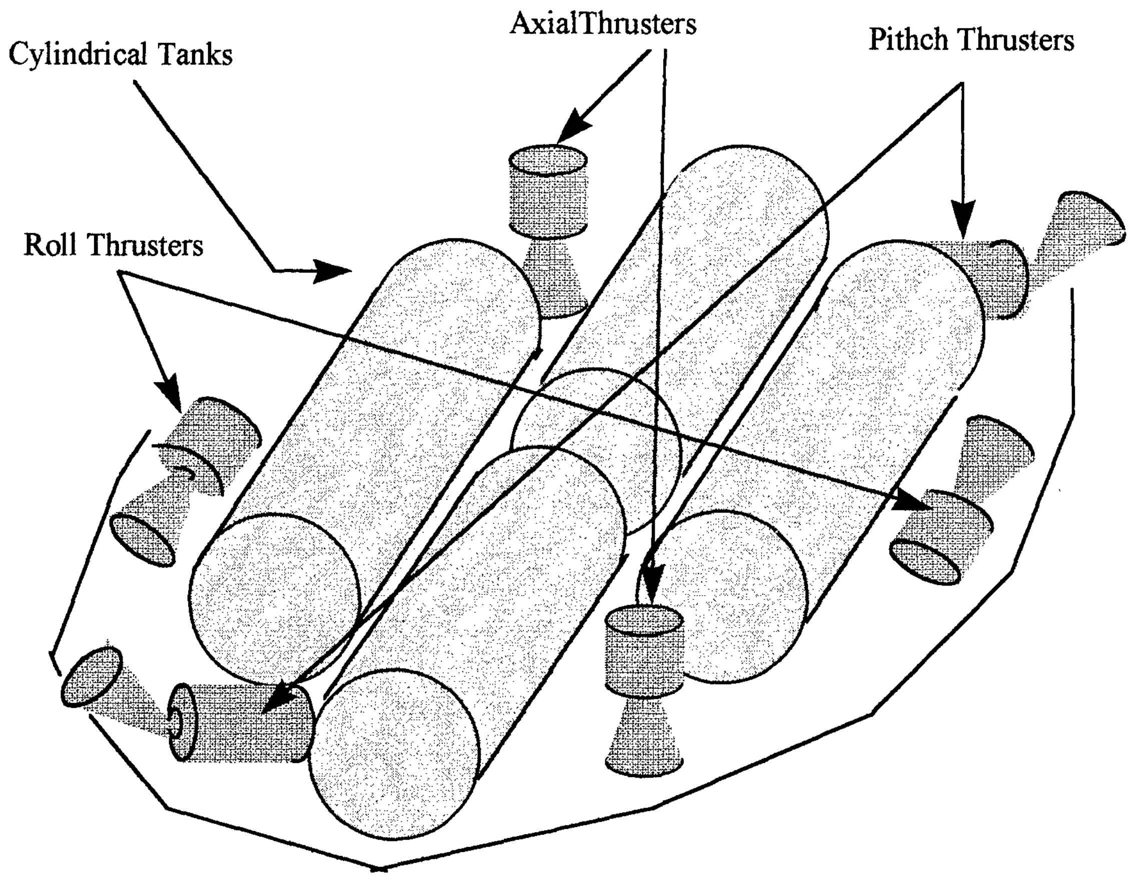


Figure 6-7: The propulsion system

6.5.3 Structures and Mechanisms

The foundation of a spacecraft is its structures and mechanisms subsystem. Structures provides the framework for mounting the mission module and other subsystems; it must also connect to the launch vehicle. "Structures must [also] endure environments from manufacture to the end of the mission" (Doukas and others, 1992:430). As with all satellite structure designs, the designer must review and adhere to the mission requirements. In this study the mission requirement was to develop a low cost, "launch on need" tactical satellite bus capable of supporting various mission modules while launching off a Pegasus class booster.

In support of a low-cost design, graphite and composite materials were not considered. Although graphite-epoxy composites provide "...extremely high stiffness-to-weight ratios..." , they are costly and "...often require a long, expensive development program to establish manufacturing processes" (Doukas and others, 1992:435-441). Because the use of composites increases the complexity of the structure and drives up construction costs, the use of other metallic alloys, such as aluminum, is more desirable. According to Doukas, et al, "Aluminum is relatively light-weight, strong, readily available, easy to machine, and low in raw material cost" (Doukas and others, 1992:435-441). Therefore, only aluminum alloys were considered for the main structure of Modsat.

"Launch-on-need" implies the ability to launch within a few days, as opposed to several months, as with current launch schedules. Since the satellite bus must be able to support multiple mission modules, modular design will to be the most viable answer to a quick response launch. To further satisfy the "launch-on-need" requirement, the Modsat satellite bus allows for quick mission module and launch vehicle integration, easing the removal, replacement, and addition of the mission module and internal subassembly components.

Since the Pegasus XL is Modsat's primary launch vehicle, the team first concentrated its efforts on ensuring the satellite bus and mission module would fit within the Pegasus payload bay.

The next consideration in selecting a satellite structure was to select a geometric shape that best utilizes the payload bay's volume and maximizes solar access (see Figure 6-8).

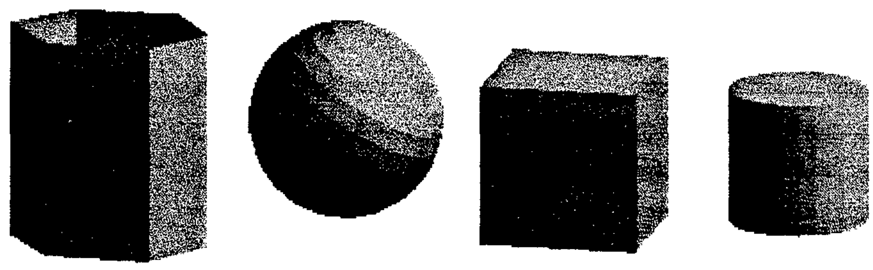


Figure 6-8: Structure Shapes

Except for the sphere, the geometric shapes are of similar construction, differing only in the number of sides, as depicted in Figure 6-9. One can see how increasing the number of sides utilizes more volume; however, there is a diminishing return to increasing the number of sides.

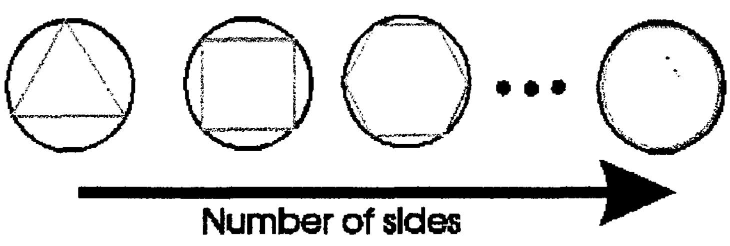


Figure 6-9: Geometric Shapes with Similar Properties

Although it appears that a cylindrical bus is the best geometric shape, the increased weight, limited access to subcomponents, and restricted solar wing placement in the stowed configuration make it a less desirable choice. Incidentally, Leritz and Palmer, in considering a geometric shape for their satellite design example, selected the cylinder and hexagon because they "distribute loads more uniformly" (Leritz and Palmer, 1995:490). From volume, surface area, and loading considerations, it seems likely that the best

Modsat structural design is a "polysat," an n-sided polygon. By varying the number sides, the best Modsat design can be found. Therefore, the "polysat" design became the main focus of structural modeling.

Other considerations in the design of a satellite bus are the accessibility of internal subcomponents and the ease of mating the mission module to it. Although it is possible to space out structural members farther apart for cylindrical and spherical bus designs, the number of members still exceeds that of the "polysat" designs. Also, internal component accessibility from the side is more impeded than with the "polysat" designs.

The next consideration of the team was to choose a satellite bus structure that best suits the goals for "low-cost" and "launch-on-need." Figure 6-10 below depicts just a few possible structural designs. In considering these designs the team evaluated how each could be molded into a "polysat" design based on the following objectives:

Modularity and Subcomponent Accessibility: How modular is the design and how accessible is it for removal and replacement of subassembly components?

Materials Usage: Does this design minimize the use of materials in its construction?

Structural rigidity: What is the inherent strength of the structure?

Manufacturing: The difficulty of constructing the structure

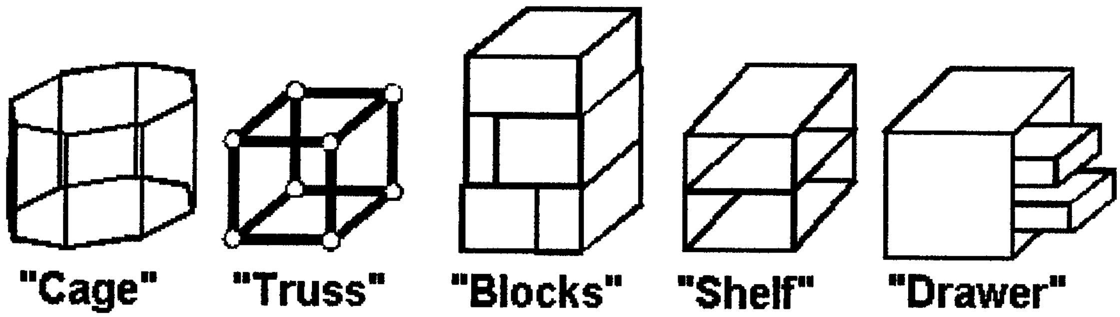


Figure 6-10: Satellite Bus Types

Cage: The structure is made up of a variable number of sides with each corner constructed with a load bearing beam. This design is the closet to the "polysat" design discussed earlier. Although this design received high marks for modularity and manufacturing, it scored below average in material usage and structural rigidity.

Truss: This structure mimics those used for future space constructions. This design scores above average in all categories except for modularity .

Blocks: This design incorporates the bolting together of subassembly components. The block design is the strongest of all , but at the expense of added weight and machining.

Shelf: Most satellite designs use the box frame with shelves to layer the placement of payload and subassembly components. This type of design received average scores for material usage and manufacturing, and below average scores for modularity and structural rigidity.

Drawer: This structure, a modified "Shelf" design, incorporates a pull-out and plug-in design. Although its modularity score improved, the drawers add extra weight and complexity to the design.

Although it was determined that Modsat should have a "polysat" structure, the best polysat configuration can only be found upon modeling the complete Modsat bus design. Second, the team reviewed both current and revolutionary designs to determine the best functional design. Both the "Cage" and the "Truss" scored well, but the "Truss" design will require additional harnesses and brackets to support subcomponent attachments. Therefore, the team decided to design Modsat with the "Cage" configuration, which best supports the "polysat" design.

6.5.3.1 Polysat Design

Since it was determined that the best shape for Modsat is a "polysat", it is necessary to find the optimal number of sides to build the Modsat bus structure.

Although the number of sides is a design variable that can be specified in the model, this number should be minimized. By doing so, more space is allowed between support beams, granting greater side access to the subcomponents. The other consideration is the solar panels. As the number of sides increases, so does the number of folds in the solar array structure (see Figure 6-11). This increases the complexity of the solar panel design and reduces the reliability of its structure. The main driver is to select the minimum number of sides to obtain the solar panel area necessary to meet power requirements.

Through extensive modeling the octagon was chosen as the baseline structure because it makes good use of the LV fairing volume while meeting power requirements.

6.5.3.1.1 Solar Panels

The requirement for large amounts of electrical power played an important part in the design of the Modsat structure. To meet the power supply goal of up to 500 watts, considerably large solar panels are necessary. Modeling for the electrical power subsystem (EPS) determined that up to seven square meters of solar array area would be required. Thus, the determination of the solar wing placement and deployment mechanisms proved to be a sizable challenge.

Since the structural design focused on maximizing internal volume, the team discounted the use of a folded-wing configuration for the stowed arrays, due to its inefficient use of space within the launch vehicle fairing. The team also considered a deployment scheme similar to that used by Milstar, where the arrays spread out accordion-style from a small canister. Although this approach saves a great deal of space, it is very technologically complex and costly. Therefore, the team decided to wrap the solar array assemblies around the bus as shown below in Figure 6-11.

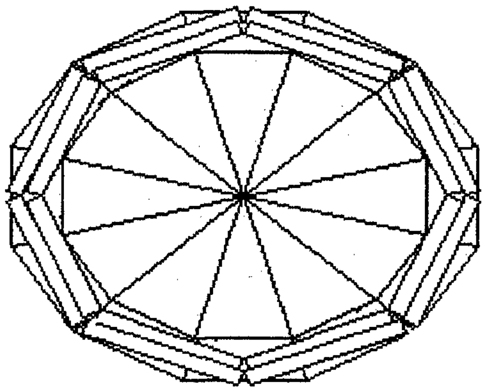


Figure 6-11: Top View of the Solar Panel Configuration During Launch

In this configuration, the solar array assemblies must be constructed and hinged so as to fold around the polygon perimeter of the bus shown in Figure 6-11. This technique makes efficient use of the space within the launch vehicle fairing by placing the solar panels along the perimeter of the launch vehicle's payload bay. Moreover, since the most severe launch loads are in the axial direction, the vertical placement of the arrays has structural advantages.

It was decided that in order to simplify the design and construction of the solar array assemblies, they would not be allowed to extend in height beyond the curve in the Pegasus fairing. Any protrusion beyond this curve would require the use of special hinges and mechanisms, which would drive up cost and decrease the structural reliability of the assemblies. Therefore, the maximum height of the stowed solar arrays is dictated by the distance from the LV-to-spacecraft interface plane to the curve in the fairing.

6.5.4 Thermal Control

6.5.4.1 Introduction

Thermal design begins with defining its purpose. As the name implies, thermal design is concerned with constructing a "*...thermal-control subsystem* ... to maintain all elements of the spacecraft system within their temperature limits for all mission phases" (McMordie, 1992:409). In every satellite design, the designer must consider the thermal impacts due to the sun, the earth, and internal heat generation. Considering each of these thermal sources is important to ensuring the subcomponents within the mission module or satellite bus operate within their prescribed temperature ranges shown in Table 6-4.

Table 6-4: Typical Spacecraft Component Design Temperatures

Component or subsystem	Operating Temperature (degrees in C)	Survival Temperature (degrees in C)
Digital electronics	0 to 50	-20 to 70
Analog electronics	0 to 40	-20 to 70
Batteries	10 to 20	0 to 35
Infrared detectors	-269 to -173	-269 to 35
Solid-state particle detectors	-35 to 0	-35 to 35
Momentum wheels, motors, etc.	0 to 50	-20 to 70
Solar panels	-100 to 125	-100 to 125

Source: Wingate, 1994:433

Just as the table suggests, the extent of thermal analysis depends on the thermal sensitivity of the satellite's subcomponents. As pointed out by McMordie, the power system has the greatest impact on the thermal design because of the electrical energy being dissipated throughout the satellite and the tight operating temperature limits of the batteries (McMordie, 1992:411).

6.5.4.2 Thermal Design and Modeling

Thermal control can be accomplished either actively or passively, or both. The team decided from the start to maximize the use of passive systems to minimize cost, weight, and the power required of active systems (McMordie, 1992:413). Since "...preliminary mission design indicates that unmanned, low-Earth orbit spacecraft can be controlled passively", active systems were not investigated in this study (McMordie, 1992:413). Therefore, passive systems such as thermal coatings, thermal insulation, and space radiators are ideal devices for meeting thermal constraints in a satellite design (McMordie, 1992:411).

6.5.4.3 Designing Thermal Interface Plate Protection

To support the bus/mission module design discussed in the structure's trade study, the team designed a thermal blanket constructed of woven insulation to be placed directly below the top interface plate, as shown in Figure 6-12. This blanket (under the mission module interface plate) will reduce the heat transfer between the bus (cage structure) and the mission module (on top of mission module interface plate) by restricting temperature changes between the two structures.

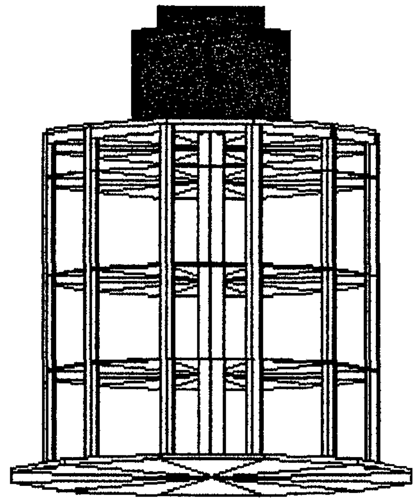


Figure 6-12: Thermal Protection of the Satellite Bus/Mission Module Interface

This preliminary analysis concentrated on the 100 °C differential of a typical bus structure as the baseline. Depending on the temperature required by the mission module, the minimum thickness of the woven insulation to preclude a 15 degree Celsius change between either the plate or the mission module can be calculated (see calculations in Appendix C). For Modsat, the team designed a 4 cm thick interface blanket.

6.5.5 Telemetry, Tracking, and Commanding/Command and Data Handling, and Communications

Three different tradeoffs affected the Telemetry, Tracking, Commanding (TT&C)/Command and Data Handling (C&DH), and Communications Subsystem. The tradeoffs occurred in the areas of communication systems, command and data handling system, and the type of components to be used in the satellite bus design.

6.5.5.1 Communications

Communications systems are very straight forward functions for a satellite. The system consists of equipment necessary to relay commands to the vehicle from the ground and to send health, status, and in some cases, payload (mission module) data from the satellite to the ground. The sponsor indicated that the bus communications system had to be compatible with the Air Force Satellite Control Network (AFSCN). The Space Ground Link System employed by the AFSCN imposed a limit on how much data could be transmitted to the ground. Since it was not uncommon for a satellite bus to employ two separate communications systems, the team decided that allowances had to be made to support a separate high data rate communications system. To provide flexibility for the user, the satellite could be included as part of the mission module design or a dedicated area on the satellite bus could be made available for a high data rate transceiver and antenna. A restriction was levied on the high data rate communications system. In addition to its primary function of downlinking mission module data, the system was required to interface with the satellite bus command and data handling system.

6.5.5.2 Command and Data Handling

The command and data handling system offered the team another area for tradeoffs. The conventional, or nominal, design of a command and data handling (C&DH) system requires that both the command and telemetry equipment, plus associated data lines, be hard-wired together. The command decoder, telemetry format unit, and controller are usually contained within a single unit called the command and telemetry unit. The C&DH functions operate under the direction of the controller. The controller maintains the satellite timing and determines command priority if more than one commanding system is in use. Figure 6-13 shows the functional relationships for a nominal C&DH subsystem.

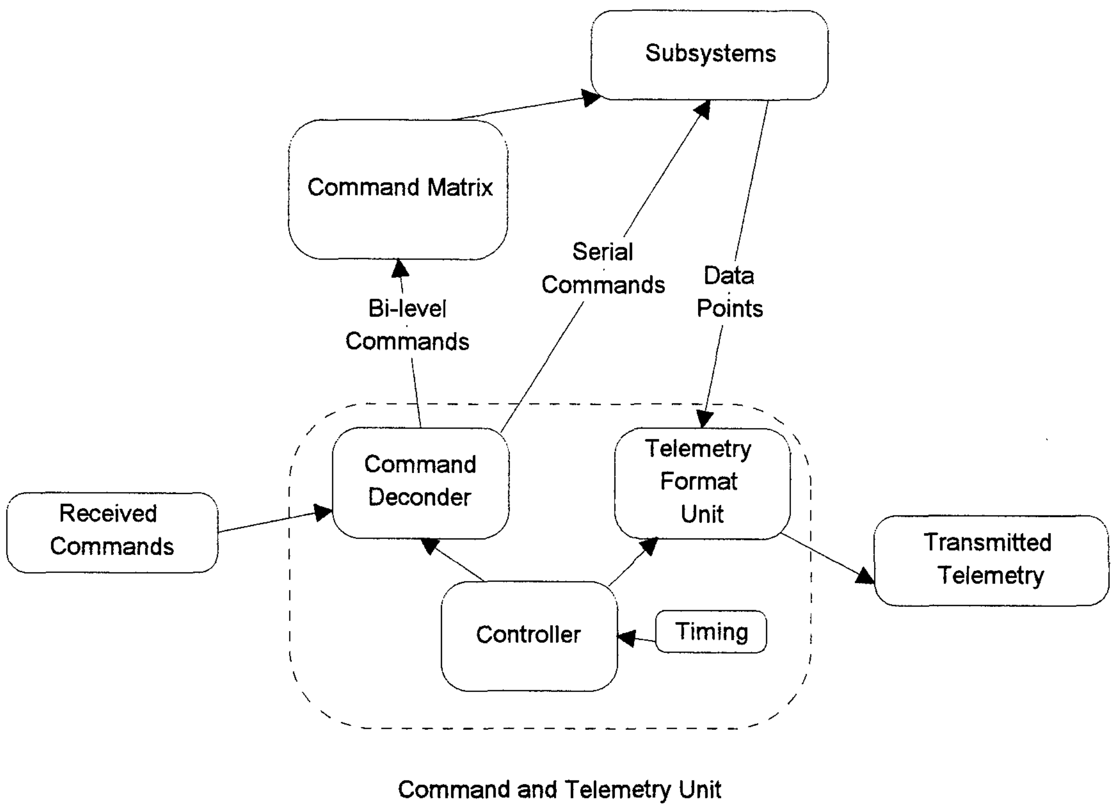


Figure 6-13: Nominal Style of Command and Data Handling

An alternate Command and Data Handling architecture involves the use of a Satellite Operating System (SOS) which enables an extremely broad range of modularity. Using software, the main processor can perform the process of decryption/encryption, command controller functions, and data compression algorithms. Subsystem interfaces route through the Interrupt Request (IRQ) processor which handles traffic deconfliction and prioritization. Instead of conventional hard-wired interfaces among subsystems, modular subsystems route their functional requests/needs through the central processor which prioritizes subsystem requirements in view of overall satellite operations. Subsequently, appropriate action is directed to the proper subsystem in a format understood by the modular subsystem. In essence, SOS provides an interface to allow vastly different subsystems to communicate to each other and work together through a baseline structure. Further advantage is gained by the ability to change SOS on the fly while the satellite is in orbit, through the uploading of new software to the central processor. A block diagram is shown in Figure 6-14:

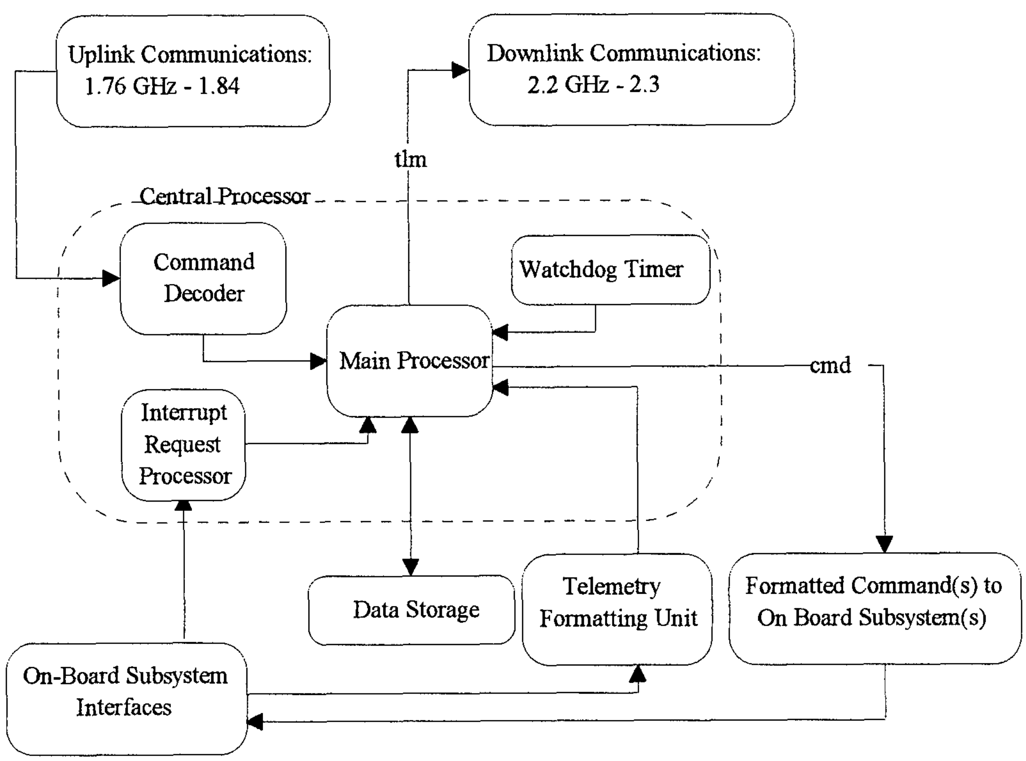


Figure 6-14: Proposed Style of Command and Data Handling

A subsystem design trade-off was performed between the nominal C&DH design and the Satellite Operating System. The Satellite Operating System was selected as the baseline for the satellite bus design. This system was selected over the conventional means because the SOS offers more advantageous characteristics. The SOS will be a lighter-weight design because software can replace some of the subsystem hardware. For example, software code can replace the command matrix board, the encryptor, and the decryptor hardware. SOS offers yet another advantage. SOS has a means for modifying the downlink of health and status data in a more user friendly-manner. Software code can permit the formatting of satellite telemetry data in such a manner that telemetry data can be accessed by ground users in much the same way that users access the internet.

Another reason for the selection of the SOS is that it is modular in nature. The modular characteristics of the system allow for the easy addition or removal of components to the C&DH subsystem. By having the IRQ checks, the system could operate with or without the additional components. If a component is not connected to the bus, the SOS would be aware of this status. If a component is added, "driver" software would be resident within the SOS program that checks to see if the added component is available for operation. Another advantage of the selection of SOS is that the C&DH code could be modified while the spacecraft is in flight. While some spacecraft permit Attitude Control Subsystem modification, C&DH subsystem modification is not an option on most current satellite designs.

6.5.5.3 Component Selection

Component selection was the last area where tradeoffs where performed for this subsystem. In order to select components for the design study, several companies were solicited for information. One of the goals of the study was to use commercial off the shelf components were possible. Components from other satellite programs, as well as the Jet Propulsion Laboratory (JPL) and the Naval Research Laboratory, provided the team with TT&C/C&DH component information.

The Command and Data Handling components chosen for the generic satellite bus design were primarily rated on their performance-to-mass ratios. Cost was not a deciding factor. Discussions with satellite designers and engineers such as Richard Warner of AeroAstro, Dave Everett of NASA/Goddard Space Flight Center, and Joel Hagan of Phillips Laboratory's MightySat Program, revealed that Command and Data Handling is one of the most critical subsystems on-board the satellite (Warner, 1996; Everett, 1996; Hagan, 1996). Without the proper functioning of this subsystem, the mission of the satellite would be lost. With this in mind, it was decided that only space-rated products should be used in the design of this subsystem.

Other factors were considered in the selection of components. A requirement for the transceiver and antenna is that they must be compatible with the Space Ground Link System. Cincinnati Electronics provided information on a small transceiver that provided the correct frequencies for communicating with the Air Force Satellite Control Network. The amount of memory provided by a data storage device is also an important factor, since the satellite needs the capacity to store both mission module data and state of health data. The Clementine Program provided a data storage device that provided 2 Gigabytes of data for a mass of only 3.4 kilograms. Other data storage devices considered were much more massive. Realizing that a Satellite Operating System architecture would require strong computing power, the team chose a radiation hardened, 32-bit, high speed Reduced Instruction Set Computer (RISC) architecture employing Very High Speed Integrated Circuit (VHSIC) technology. This computer was developed by Honeywell for the Ballistic Missile Defense Office. The associated controller and memory modules were developed under sponsorship of the Naval Research Laboratory. The computer and its associated controller were light weight and extremely powerful.

Specific components were not selected for the high data rate communications system. As part of the design study, it was decided that the mission module developers would provide a transceiver and antenna for the satellite bus. This arrangement provides more flexibility for the mission module. Interface data, power, mass, and size requirements would be provided to the mission module developers to assure compatibility with the Satellite Operating System. In the concept of operations, the use of the Satellite Operating System would allow easier integration of a high data rate communications system.

6.5.6 Electrical Power

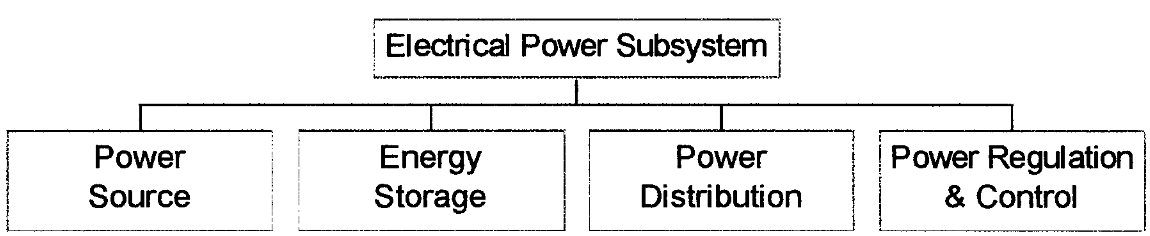
6.5.6.1 Objectives

An emphasis was placed on using readily available, relatively inexpensive components. EPS design also emphasized the use of proven components and architectures. In order to maximize the flexibility, an effort was made to accommodate modularity and ease of integration.

The primary objective for EPS design is to provide sufficient power for the possible range of mission modules. The EPS must accommodate the most power-hungry applications planned to be flown on Modsat. Thus, a large amount of power must be available for the mission modules. However, not all configurations will require all the available power. A sound systems engineering approach must consider the impact of excess power (thermal effects, inefficient use of weight, etc.).

6.5.6.2 Functional Allocation

The EPS consists of four major functional areas, as shown in Figure 6-15: power source; energy storage; power distribution; and power regulation and control. A simple block diagram of the EPS is shown in Figure 6-16.



Source: McDermott, 1992:391

Figure 6-15: Electrical Power Subsystem Functions

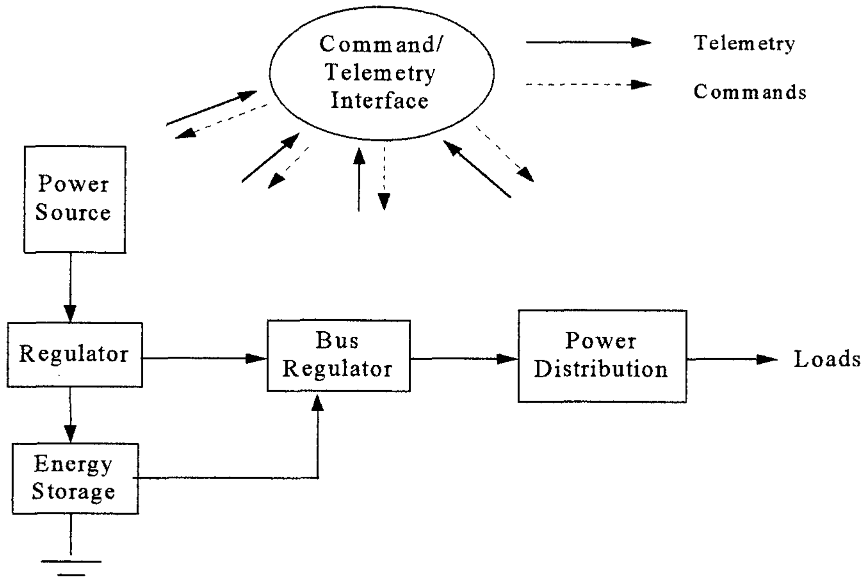


Figure 6-16: EPS Block Diagram

6.5.6.3 Power Source

The power source generates electrical power for the spacecraft, supporting the electrical loads over many orbits. The power generation system of choice for Modsat is photovoltaic solar cells. Solar photovoltaics are the typical power source for LEO spacecraft. They are proven and widely available. Solar power is adequate for the intended mission, and lighter (in terms of specific power) than the other alternatives. Although there are cheaper sources, i.e., solar thermal dynamic and nuclear reaction, these alternatives require more mass. Nuclear reactors, furthermore, require a fuel that has limited availability, while solar energy is unlimited. Reactors are appropriate for applications requiring a large amount of power, but not for the loads and environment expected for a small, tactical LEO spacecraft such as Modsat. The same can be said of the solar thermal dynamic alternative. The radioisotope alternative is very cost prohibitive. Given the strong emphasis on affordable access to LEO missions, the radioisotope alternative was ruled out.

Solar cells must be mounted on solar array assemblies. The arrays should be planar and oriented toward the sun (Reeves, 1992:317). The power output is proportional to the amount of sunlight incident on the panels. Thus, the spacecraft should track and point the arrays for maximum incidence.

Planar arrays may be body- or panel-mounted. The team decided to use panel-mounted solar arrays, on deployable booms, for Modsat. Following are reasons for this decision:

- Panel-mounted arrays are usually mounted on deployable booms that rotate the panels for maximum incident sunlight.
- Though body-mounted arrays reduce the requirement for tracking and pointing, they achieve less effective sun incidence angles and power conversion efficiencies than deployable panels.
- There may not be enough available area on the body of the spacecraft for body-mounted arrays to provide the required power.
- With deployable booms, it is easier to place the panels away from payload instruments and other temperature sensitive subsystems that could be damaged from the highly variable temperature environment of solar cells.

On the other hand, panel-mounted arrays on booms add appendages to the spacecraft that add to the mechanical complexity and take up precious space within the launch vehicle fairing.

The team recommends the modular solar array approach used by NASA's Small Explorer (SMEX) Program (Everett, 1996), as well as Phillips Laboratory's MightySat II spacecraft (Hagan, 1996), for the following reasons:

- It is desirable to minimize the amount of excess power on Modsat.
- It is more economical to limit the size of the solar array structure to the size necessary to produce the required power.
- Flexibility in the design and integration of each spacecraft
- With the modular approach, the engineers can build the solar array assemblies much later in the overall process of spacecraft construction (Everett, 1996).

In the modular approach, a solar array is constructed of smaller solar modules. These modules are essentially small solar panels (8" by 16" for the SMEX program), which can be purchased in mass quantities (at reduced prices) long before assembly. Once the required power for the spacecraft is determined, the appropriate number of modules can be pulled from storage and integrated into a customized solar array assembly. This assembly consists mainly of a structural grid for inserting the solar modules. It must be built so as to conform to the chosen stowed/deployed configurations for the spacecraft.

Although the modular solar array approach is recommended, the team decided to design each of the alternative Modsat concepts with maximum power for the sake of designing to the most difficult set of requirements.

During launch, the arrays will be wrapped around the bus in the stowed configuration. Thus, the amount of surface area provided by the bus, combined with the number of wraps, determines the power generation capability of the arrays, depending on the efficiency of the solar cells.

Either silicon (Si) or gallium arsenide (GaAs) cells could be used on Modsat. Silicon is a clear choice for consideration. It is a proven, mature technology that has been the industry standard for years, although it is more bulky and less survivable than other materials. Gallium arsenide is also a proven technology, and is becoming more and more common on spacecraft. Although it is expensive compared to silicon, alternatives for which mass and volume become critical may require the use of gallium arsenide.

In the EPS design portion of the Modsat model (see Volume III), one can choose either silicon or gallium arsenide as the solar cell material. However, throughout the design process, it became clear that silicon cells would not provide enough power per area to satisfy the requirement of the CDM (up to 500 watts), given the stowed configuration limitation on the size of the arrays. Thus, all of the Modsat alternatives were designed with gallium arsenide solar cells.

The solar arrays must be sized to provide the required power at the end of the spacecraft life. Thus, the arrays must be large enough to account for the degradation that occurs over time.

A final consideration for the solar arrays is the selection of solar array drive motors (SADMs). SADMs are necessary to mechanically rotate the solar array structures to achieve optimum angles of incidence with sunlight. Two SADMs are needed, one for each solar array. For modeling purposes, representative SADMs were chosen from the Jet Propulsion Laboratory (JPL) Flight Hardware Survey, with the following characteristics:

- Mass: 5 kg each.
- Dimensions: Each is a cylinder, 10 cm diameter by 27 cm length.

6.5.6.4 Energy Storage

Since Modsat will not be generating power with a constant source, such as radioisotopes or nuclear reaction, a system must be designed to store energy for eclipse operations. The storage system must also be able to handle peak loads beyond the capability of the solar arrays, which are designed to handle average loads.

The most common storage devices for LEO spacecraft are chemical batteries, known as secondary batteries since they discharge during eclipse and recharge in sunlight (McDermott, 1992:402). Other storage schemes are available, such as thermal storage, but they are far less common for LEO spacecraft than batteries. Some new approaches, such as flywheel storage, require much less weight than batteries. However, it was decided that new approaches with unproven technology would not be appropriate for the Modsat EPS. Space batteries are commonly used and widely available, and are therefore appropriate for Modsat energy storage.

The two primary types of batteries in use today are nickel cadmium (NiCd) and nickel hydrogen (NiH_2) batteries. Nickel cadmium is a proven, easily available and comparatively cheap technology, but it delivers a relatively low energy density. Nickel hydrogen delivers more energy per kilogram than nickel cadmium, but it is more expensive. However, the cost objective received a lower priority than that of mission utility. Although NiH_2 is a less mature technology for LEO applications (McDermott, 1992:403), in recent years it has been used successfully in many spacecraft. The new common pressure vessel (CPV) design nickel hydrogen technology was successfully qualified on the Clementine mission in 1994 (Clementine Report).

Since the CDM requested a peak power capability of 1000 watts, Modsat requires a great deal of battery capacity. The weight of the bus must be minimized; therefore, nickel hydrogen was preferred by the team. In fact, in an effort to drive down weight, the team decided to design all alternatives with two nickel hydrogen CPV batteries from Johnson Controls, the same battery flown on the Clementine mission. Characteristics of this battery are shown in Table 6-5. This design choice gives Modsat 30 amp-hours of battery capacity.

Table 6-5: Characteristics of the Johnson Controls/Clementine NiH2 CPV Battery

# of cells	Capacity (amp-hour)	Energy density (W hr/kg)	Dimensions (cm)	Weight (kg)
22	15	47.1	50.8 x 13.4 x 13.4	9.5

Source: Clementine Report

More flexible alternatives were examined, such as a modular scheme employing several low-capacity batteries. However, in keeping with the importance of minimizing weight and maximizing capability, modularity was sacrificed for a low weight, high power battery choice.

The batteries were sized to handle:

- Required power during eclipse
- Peak power demands (supplementing the solar arrays)

6.5.6.5 Power Distribution

The power distribution system consists of the cabling, fault protection, and power switching gear to turn power on and off to the spacecraft loads. A major focus in distribution system design is on keeping mass and power losses at a minimum while providing survivability, cost, reliability, and power quality.

Since Modsat is intended to be a generic, standardized bus, the required load of the mission equipment is unknown until the mission need arises. Moreover, flexibility in configuration is a key value of the decision maker. Therefore, there is no fixed load profile for Modsat; the distribution system must take this into consideration.

Mechanical relays are the clear choice for power switching, "because of their proven flight history, reliability, and low power dissipation" (McDermott, 1992:405). Solid-state relays may be the standard choice in the future, but presently they are an immature space technology.

Decentralized distribution is the best approach for Modsat. It allows for a great deal of flexibility, thereby complementing a modular architecture. Each power-using component, including the mission module, must provide its own converter/regulator. The load nodes will be fixed, but the load users may be changed. This scheme obviates the need for customizing the distribution network for each mission application.

The distribution system should include provisions for fault detection, isolation, and correction during testing (McDermott, 1992:406). Each fully integrated Modsat spacecraft must undergo extensive testing, especially since each many configurations will be unique. A full set of fuses, placed in series with the power bus, will be required for fault testing (McDermott, 1992:407).

The wiring can have up to 4% of the mass of the dry spacecraft (Reeves, 1992:319). Since this is not a trivial amount, it is important to keep the distribution cables as short as possible.

For modeling purposes, a power control unit (PCU) was chosen from JPL's Flight Hardware Survey to represent the distribution system. This unit is a 24 x 16 x 20 cm box.

6.5.6.6 Power Regulation and Control

Power regulation must be considered for three key elements: controlling the solar array, regulating bus voltage, and charging the battery (McDermott, 1992:407). Solar array power must be controlled at the array to prevent battery overcharging and excessive spacecraft heating (McDermott, 1992:407).

Two primary schemes are available: a peak-power tracker (PPT) and a direct-energy-transfer (DET) system (McDermott, 1992:407; Everett, 1996). A PPT is nondissipative; it extracts the exact amount of required power up to the array's peak output. A PPT operates in series with the solar array, and uses 4-7% of the total power (McDermott, 1992:407). The DET system is dissipative because it shunts all power not used by the loads. A shunt regulator operates in parallel to the array and shunts the array current away when the loads or battery charging do not require the power (McDermott, 1992:407).

A DET system should be used as opposed to a PPT. The DET has fewer parts, lower mass, and higher total efficiency (McDermott, 1992:407). Moreover, given the use of modular solar arrays, different Modsat missions will have different amounts of power being supplied. A simple shunt-regulated system, i.e., the DET system, would aid in the integration of such an architecture.

An unregulated system should be used to control bus voltage, since regulation involves the dissipation of energy. This is inefficient and could potentially create heat in undesirable places.

Batteries can be charged individually or in parallel. Parallel charging keeps the voltage of all batteries the same, but allows current and temperature to vary (McDermott, 1992:409). Individual charging "optimizes the battery use by charging all the batteries to their own unique limits" (McDermott, 1992:409).

Due to the emphasis on design flexibility, it seems advantageous to employ independent battery chargers. Their use aids in vehicle integration and maximizes the use of each battery (McDermott, 1992:409). A modular battery approach would be greatly facilitated by the use of independent charging, and greatly hampered by parallel charging (Everett, 1996). On the other hand, independent charging is more expensive and complicated than parallel charging (McDermott, 1992:409), and adds more weight. The current tradeoff of cost for mission utility led the team to recommend the independent charging approach.

For modeling purposes, a shunt regulator was chosen from JPL's Flight Hardware Survey to represent the regulation system. This unit is a 42 x 18 x 11 cm box.

6.6 Tradeoff Summary

6.6.1 System Level

- One satellite per launch vehicle
- 3-axis stabilization
- Modular component interfaces
- On-board propulsion
- Mission data storage
- Spacecraft to Payload Interface Guideline (SPIG) interface

6.6.2 Subsystem Level

ATTITUDE DETERMINATION AND CONTROL

- Inertial Measurement Unit
- Star sensor
- Earth sensor
- Reaction wheels

PROPULSION

- Monopropellant configuration
- Hydrazine blend propellant; 24% HN, 76% N_2H_4
- Blowdown pressure system
- Positive expulsion fuel storage system
- Bottom level location for the system
- cylindrical tanks

- thrusters

STRUCTURES

- Octagon "cage" structures with "polysat" design
- Modular mounting plates

THERMAL

- Interface thermal blanket between the mission module and satellite bus

TTC/CDH

- Satellite Operating System

  >> software handles interrupt requests to collect telemetry and handle housekeeping

- Internet communication format (TCP/IP)

  >> JAVA compatible architecture enables user in field to access data via browser capable system

  >> provides point-and-click environment to control satellite

  >> encryption on demand with software plug-in modules

EPS

- Modular solar arrays on deployable, articulated booms
- Solar arrays wrap around the bus in the stowed launch configuration
- Gallium arsenide solar cells
- NiH_2 common pressure vessel batteries
- Decentralized, unregulated distribution
- Shunt regulation; individual battery charging

7. Modeling

Models play an important role in solving complex and multiple variable problems; as such, it is a critical element of systems engineering. To use systems engineering in designing a standardized tactical satellite bus, the team was faced with many variables and the relationships amongst them. Satellite design by its very nature is quite complex, and trying to account for every detail in this preliminary study is infeasible. Modeling enabled the team to approach the satellite design at a high level, focusing only on the major elements of the spacecraft. The team also used the model to test and evaluate the performance of the alternative solutions.

Before considering modeling methods, it was important to revisit the problem definition, the objectives, and the measures of effectiveness (MOEs). This ensured the modeling effort was fully relevant to solving the problem, and within the framework of the objectives, the model would be able to evaluate the alternatives proposed to solve the problem.

Once the basis for the model was understood, the team outlined the scope and type of model necessary to meet the objectives. Starting from a high level, the team determined the model must be highly integrated and compatible; the best way to satisfy this requirement was to use one program for all of the modeling. Once the model produced a satellite design, it would be important to perform some level of environmental and integration testing on it. Those designs meeting the testing and integration modeling elements required data analysis to assist in the final evaluation of how the alternatives ranked with each other.

Although these requirements for an integrated model appear easy to fulfill, they were not. The research conducted on the Internet found satellite software modeling still geared toward specific subsystems. Discussions with aerospace companies confirmed our findings, however, they mentioned how some companies are now using computer labs to bring subsystem experts together for satellite design sessions. To satisfy all the modeling requirements, it was necessary to develop an in-house satellite design model, using Matlab, a mathematical and graphics software package.

To meet the "integrable, compatible, and adaptable" modeling requirement, Modsat (Modular Satellite) was constructed and built around a generic database structure format. By constructing all the subcomponents with the same generic database, compatibility among all the subcomponents was achieved. Once the subcomponent databases were constructed, they were combined into a single satellite database, thus ensuring the overall satellite design was totally integrable. Modsat satisfies the "adaptability" requirement by allowing the satellite designer to modify Modsat in three ways:

- Correct, delete and/or expand the satellite database.
- Make changes to the Modsat code substructure to incorporate more detailed analysis or changes in technology.
- Tie into other external programs

Using the modeling requirements listed above, Modsat development started with a high-level structure as shown in Figure 7-1.

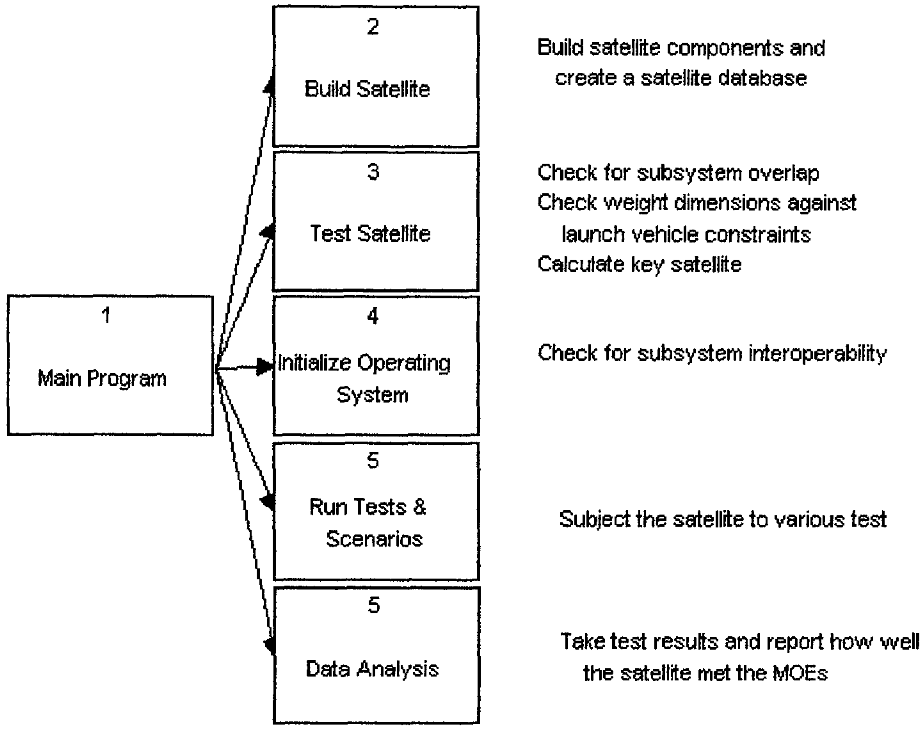


Figure 7-1: Logical Flow of Modsat Model

The Modsat model is geared toward sequential operation, starting from the top of the logic flow diagram and working down. Each step provides additional information about the satellite design, and at any given step the satellite design can be stopped to be modified or re-accomplished.

Building satellite: Build subcomponent databases and combine them into one integrated satellite database.

Test Satellite: Check the satellite database to ensure total satellite mass, center of mass, and sizing meet the launch vehicle constraints. If the satellite design fails any of these tests, the satellite design must be either scrapped or modified.

Initialize Operating System: Once the satellite design passes the "test" section, the satellite database is checked for subsystem interoperability such as power requirements. This section also calculates cost and reliability for the satellite design.

Run Scenarios: This section subjects the satellite bus to launch and orbit environmental testing to determine its overall performance.

Data Analysis: The satellite performance parameters are fed into data analysis either directly or indirectly, before being evaluated against a set of objectives. Each satellite design receives an overall utility score as well as an overall ranking with other designs. To complete the analysis and provide some variability in the results, sensitivity analysis can be performed by modifying and rescaling the top level objective weights.

Although the Modsat model used first order assumptions and calculations, it proved to be an excellent systems engineering tool, allowing the team to design, test, and evaluate satellite designs. For a full description of the model, see Volume III.

8. System Synthesis

8.1 Subsystem Baselines

Each subsystem level study addressed the selection of individual components and the narrowing of requirements for each satellite subsystem. These components then became fixed in all of the candidate designs, yielding a baseline design on which alternative configurations could be based by varying the characteristics and/or configurations of the subsystems not fully determined by the tradeoff studies. This streamlined approach to subsystem and eventual system design facilitated the focusing of effort and the effective utilization of extensive satellite subsystem expertise by the more experienced team members.

8.2 Alternative Design Descriptions

The following discussion details the alternative designs evaluated using the Modsat computer modeling, design, and analysis software. To begin scoping the alternatives, all were designed to be compatible with the Pegasus XL/without HAPS configuration. Of the seven designs, six are based on the 38" interface, while the seventh is based on the 23" interface. The team judged early in the design process that the 23" interface configuration probably would not provide enough volume for Modsat because it reduces volume for the mission module and the size of the solar arrays. However, one alternative with this interface was generated for the sake of a complete analysis.

The designs fall into one of two categories. In the first category, there are three levels for component placement. The third level has a Supplemental Mission Adaptive Shelf, or "SMASH", which is a large volume of space reserved for either a dedicated high data rate antenna and its transponder, or some other user-specified payload or mission-unique equipment. The second category has only two levels, reducing the overall bus height by not including a SMASH for additional mission equipment (such as a high data rate communications package). In the case of the second category designs, all mission-unique equipment would be placed entirely on the payload interface plate (very top of the bus).

All designs fit a particular tactical profile, with tactical capability being determined by the amount of fuel on-board. Those designs with more fuel have more ability to make in plane or out of plane maneuvers, satellite altitude corrections, or other maneuvers. In-plane changes can be measured by ΔV; thus, total ΔV capability is the measure for how much a satellite can change its velocity. ΔV capability is directly proportional to the amount of fuel carried by the spacecraft. All of the designs have varying amounts of ΔV capability according to three sizes of propellant tanks carried by the spacecraft. There are three profiles for ΔV: max (450 m/s), mid (300 m/s), and low (100 m/s). These tactical profiles correspond to varying demands which may or may not be placed on the satellite during its mission. For a given category the three tactical profiles are essentially the same except for the size of the propellant tanks. Since the bottom level of the bus is reserved for propulsion, different tank sizes will raise or lower the bottom mounting plate, thus increasing/decreasing the overall height of the satellite bus. The three tactical profiles and their capabilities are summarized in Table 8-1:

Table 8-1: Tactical profiles

MAXTAC	Maximum ΔV capacity for orbit maintenance Up to 2.5 degrees of inclination change during mission
MIDTAC	Median ΔV capacity for orbit maintenance Up to 1.5 degrees of inclination change during mission
LOWTAC	Minimum ΔV capacity for orbit maintenance (one year)

All designs have the same power supply capability (447 Watts average/1000 Watts peak), except for the 23" interface design, which has lower output (390 Watts avg, 950 Watts peak) due to less area for the solar arrays. The seven designs generated as alternative solutions are as follows:

- MAXTAC: Max tactical profile, with three component levels and SMASH
- MIDTAC: Mid tactical profile, with three component levels and SMASH
- LOWTAC: Low tactical profile, with three component levels and SMASH
- MAXTAC-N: Max tactical profile, with two component levels and no SMASH
- MIDTAC-N: Mid tactical profile, with two component levels and no SMASH
- LOWTAC-N: Low tactical profile, with two component levels and no SMASH
- MIDTAC-23: Mid tactical profile, with three component levels and SMASH; 23 inch interface

The major characteristics of each design, including the height of each bus, were determined through modeling, and the results are shown in Table 8-2 .

Table 8-2: Major Design Characteristics

Acronym	SMASH	Number of Levels	ΔV (m/s)	Interface (in)	Bus Height (cm)
MAXTAC	Yes	3	450	38"	71
MIDTAC	Yes	3	300	38"	68
LOWTAC	Yes	3	100	38"	63
MAXTAC-N	No	2	450	38"	69
MIDTAC-N	No	2	300	38"	66
LOWTAC-N	No	2	100	38"	61
MIDTAC-23	Yes	3	300	23"	95.7

8.3 Convergence of Individual Designs

Most of the designs are very similar, except for the category and tactical profile differences; once an initial positioning of components is accomplished, the other designs simply incorporate the SMASH/no SMASH option and the tactical profile (propulsion) options. The design most divergent from the rest, of course, is the design based upon the *Pegasus* launch vehicle 23-inch interface.

The process of design convergence begins with the physical constraints placed upon the satellite by the launch vehicle (in this case, the *Pegasus*). Further, the main physical dimension constraining the bus is the diameter of the interface (38 inches for the first six designs, and 23 inches for the seventh). Starting with the interface plate of the launch vehicle and proceeding upwards with successive component "shelves", placement of components may be initially estimated, spatially evaluated, and iteratively adjusted, until a reasonable, desirable arrangement of components solidifies into an integrated satellite design. The semi-cylindrical, shelf-like structure chosen for Modsat is conducive to this design convergence approach and facilitates the organization of components into "functional" groups.

8.4 Component/Characteristic Listing

The following list summarizes the set of characteristics and components that were chosen for Modsat. It should be clear that, to be consistent with the scope of this study, this is a high-level set of major components. The rationale behind the selection of these components is discussed in the Tradeoffs section of this report. Only those characteristics that were relevant to the integration of the overall design are included below.

- Launch Vehicle Fairing Static Envelope (inside of which the spacecraft may be placed)
  Note: The Pegasus User's Guide specifies a dynamic envelope of 46". The team decided to use a more conservative static envelope due to the risk inherent in the use of the wrap-around solar array assemblies. This approach is used by the SMEX program (Everett, 1996).
  - Diameter: 44" ≈ 112 cm
  - Height from spacecraft interface to curve in fairing: 111 cm
- Structures and Mechanisms
  Note: All structural members are made of 7075-T6 aluminum, with a density of 2,800 kg/m^3.
  - Shape: Octagon
  - Structural cylindrical columns (8)
    - Dimensions: hollow; 4 cm outer diameter; 1.5 cm inner diameter; height varies with each design
  - LV interface plate
    - Diameter: 111.76 cm
    - Thickness: 1 cm
  - Component mounting plates (2 for with SMASH, 1 for no SMASH; note that the LV interface plate forms the bottom component plate)
    - Diameter: 87.26 cm

- Thickness: 0.2 cm

- Center spine
  - Dimensions: 5 cm square; hollow; height varies with each design

- Propulsion

  - Propellant tanks (4)
    - Dimensions: hollow cylinder; 2 tanks are 42 cm long, 2 are 33 cm long; diameter varies with tactical profile

  - Thrusters (6)
    - Dimensions: modeled as a cylinder; 6.6 cm diameter; 10.2 cm height

  - Valves (6)
    - Dimensions: modeled as a cylinder; 6.6 cm diameter; 8.2 cm height

- ADCS

  - Reaction wheels (4)
    - Dimensions: cylinder; 25.5 cm diameter; 9 cm height
    - Mass: 5.1 kg each
    - Power requirement: 17 watts max

  - Reaction wheel electronics boxes (4)
    - Dimensions: box; 17.8 by 15.2 by 3.2 cm
    - Mass: 0.9 kg each

  - Earth sensor (head)
    - Dimensions: cylinder; 10.4 cm diameter; 16.3 cm length
    - Mass: 1.27 kg
    - Power requirement: 0.5 watts

  - Earth sensor (electronics)
    - Dimensions: box; 10.2 by 20.3 by 6.7 cm
    - Mass: 1.14 kg
    - Power requirement: 3.5 watts

  - Star sensor
    - Dimensions: cylinder; 13.5 cm diameter; 14.2 cm length
    - Mass: 2.5 kg
    - Power requirement: 10 watts

- Inertial Measurement Unit (IMU)
  - Dimensions: cylinder; 21.6 cm diameter; 13.3 cm height
  - Mass: 3.72 kg
  - Power requirement: 33 watts

- EPS

  - Solar array panels (16)
    Note: Since the bus has an octagon shape, there are eight hinged panels per wrap around the bus. All configurations used two wraps, therefore all have 16 panels.
    - Dimensions: box
      -- Thickness: 4 cm
      -- Width: inner wrap panels are 36.14 cm wide; outer wrap panels are 39.45 cm wide
      -- Height: varies with each design
    - Mass: varies with size of arrays

  - Batteries (2 NiH_2)
    - Dimensions: cylinder; 13.4 cm diameter; 50.8 cm length
    - Mass: 9.5 each
    - Capacity: 15 amp-hours each

  - Regulator
    - Dimensions: box; 42 by 18 by 11 cm
    - Mass: varies with power output

  - Power Control Unit (PCU)
    - Dimensions: box; 24 by 16 by 20
    - Mass: varies with power output

  - Solar Array Drive Motors (2 SADMs)
    - Dimensions: cylinder; 10 cm diameter; 27 cm length
    - Mass: 5 kg each

- TT&C/CDH

  - Transceiver
    - Dimensions: box; 21 by 15 by 13 cm
    - Mass: 3.41 kg
    - Power requirement: 22 watts

  - SGLS Antennas (2)
    - Dimensions: cylinder; 4 cm diameter; 10.2 cm height

- Mass: 0.25 kg each

- Central Processing Unit (CPU)
  - Dimensions: box; 5 by 22 by 15
  - Mass: 0.9 kg
  - Power requirement: 2.8 watts

- Data Storage Unit
  - Dimensions: box; 13 by 13 by 13
  - Mass: 3.4 kg
  - Power requirement: 1 watt

8.5 Baseline Design

The MAXTAC bus (maximum propulsion, three shelves with SMASH) was chosen as the baseline upon which to conduct the detailed placement of components. The following discussion applies to the design of the MAXTAC. In section 8.6, the other alternatives will be discussed. The complete set of databases for each design is included with the modeling software (see Volume III).

With the basic structure at hand, the team had to choose desired locations for the components. It was decided to place the components as shown in Table 8-3.

Table 8-3: Component Placement

Component Level	Components
1st	Propulsion; one SGLS antenna
2nd	Batteries; PCU; Regulator; Reaction wheels and electronics; Earth sensor and electronics; Star sensor
3rd	IMU; TTC & CPU components; SADMs

The rationale for locating the propulsion subsystem on the bottom level is documented in the Tradeoffs section of this report. It was decided to place a SGLS antenna on the bottom level as well, such that the antenna would have a field of view through a hole in the LV interface plate. It was felt that an antenna was necessary in this location in order to communicate with Modsat prior to its full structural deployment.

The team decided to place components on the second level in such a way as to keep the spacecraft center of mass as low as possible (due to LV considerations). Thus, the relatively massive batteries and reaction wheels were placed on the second level. In order to collocate as many subsystem components as possible, the regulator, CPU, reaction wheel electronics, earth sensor and electronics, and star sensor were placed on the second level as well.

However, it was necessary to place the SADMs on the third level. The axis of the SADMs must be aligned with the axis of the solar array assemblies, which must in turn be located high enough to provide a balanced inertia matrix for the total spacecraft. In addition, the SADMs must be placed on the outer edge of the component plate, in order to mate with the extended solar array booms. The remaining TT&C/CPU components were placed on the third level. Nearly one half of the third level was reserved for the SMASH. The team modeled the use of this space with a hypothetical high data rate (HDR) package, consisting of an antenna and a transceiver. The antenna was modeled as a large cylinder, while the transceiver was chosen to be identical to the TT&C transceiver.

The very top mounting surface of the bus (designated the "payload interface plate") serves as the substructure for the mission modules. Several different types of mission modules may be evaluated with the various bus designs by using the Modsat computer model. For ease of evaluation for this design study, the mission modules are oriented in the axial direction because the on-orbit attitude for the Modsat will have the payload interface plate pointing nadir. Future design efforts may introduce alternative orientations.

Finally, it was necessary to keep the spacecraft center of mass as close as possible to its central axis. Thus, mass symmetry was a major factor in the placement of components.

8.5.1 Structure

Based upon the ΔV requirements given for the tactical profile desired, the height of the first component plate was set, and the initial positioning of components could commence. The MAXTAC structure is shown in Figure 8-2.

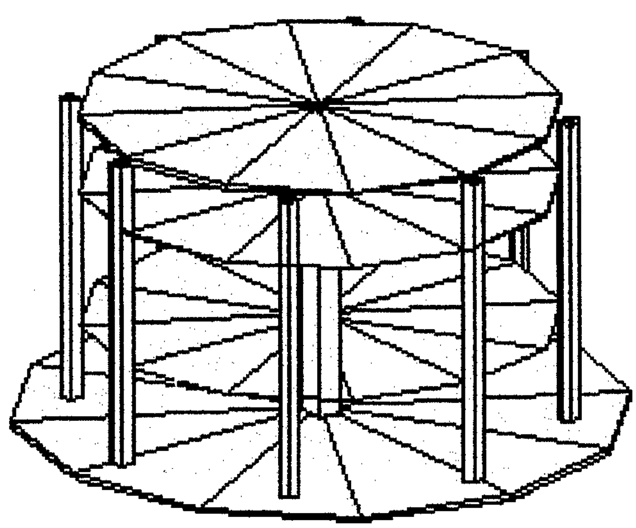


Figure 8-2: MAXTAC Structure

8.5.2 Propulsion

Figure 8-3 shows the propulsion subsystem integrated into the structure.

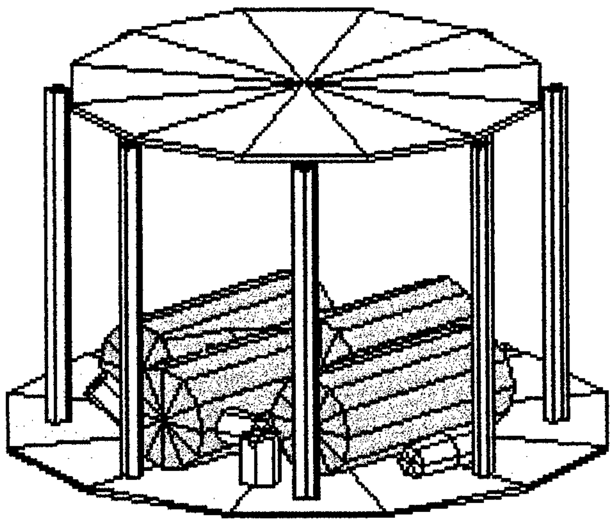


Figure 8-3: MAXTAC Propulsion Subsystem

8.5.3 ADCS

For the purpose of optimum three-axis control, the reaction wheels are canted at 45 degree angles. The sensor components are placed at the outer edge of the component plate, such that they have a field of view through holes in the outer bus wall. The sensors face directions in which there are no solar arrays blocking the view. In Figure 8-4, the ADCS components are shown integrated into the structure.

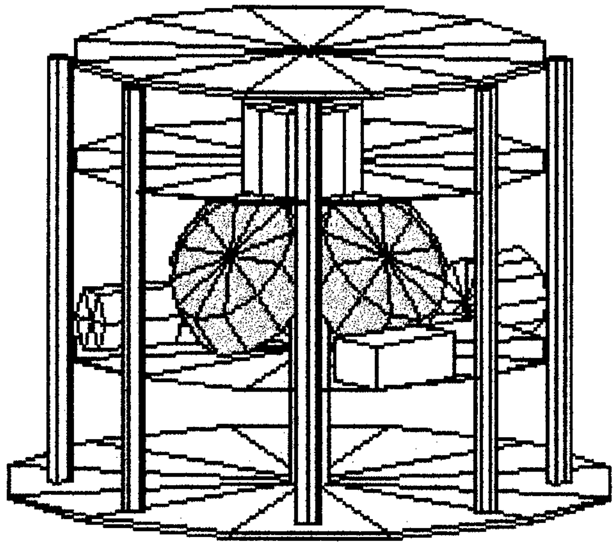


Figure 8-4: MAXTAC ADCS

8.5.4 EPS

Due to the placement of the reaction wheels, the batteries hang from the bottom of the 2nd component plate. The regulator and CPU are placed symmetrically where space allows. The stowed and deployed EPS configurations are shown in Figure 8-5.

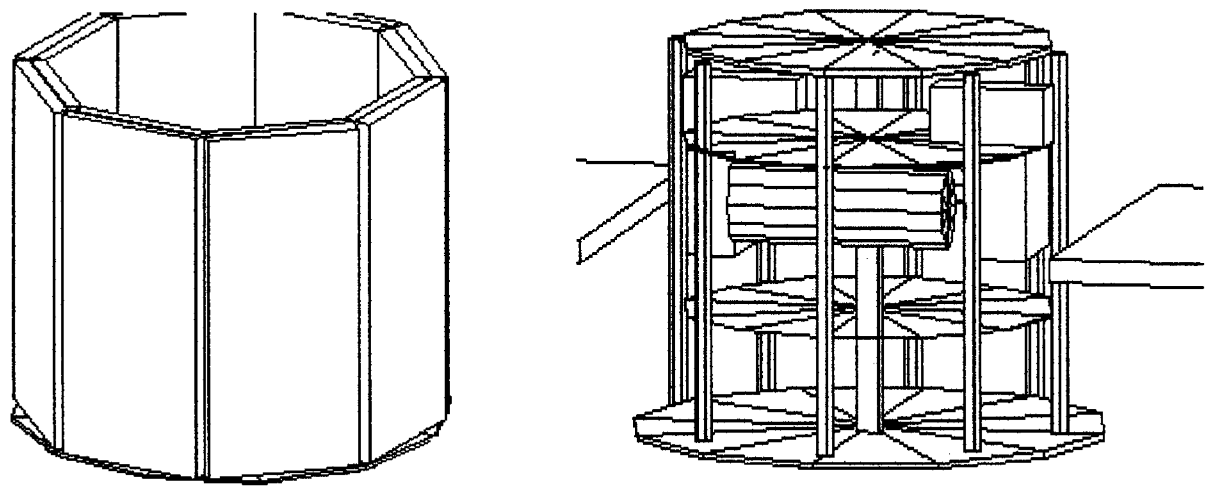


Figure 8-5: MAXTAC EPS

8.5.5 TTC

The SGLS antenna on the third component level is placed close to the outer edge of the bus. This antenna must be deployed on a boom with an appropriate mechanism. TT&C/CPU components are shown in Figure 8-6.

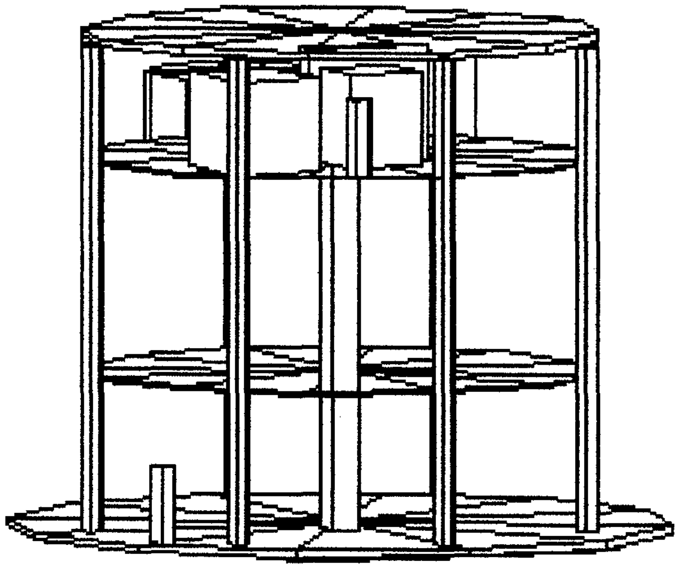


Figure 8-6: MAXTAC TT&C/CPU

8.5.6 MAXTAC SMASH

Figure 8-7 shows the top level of the bus, with the SMASH being utilized by a high data rate communications package.

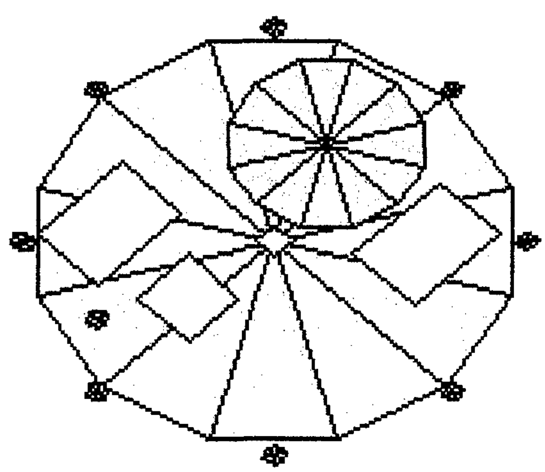


Figure 8-7: MAXTAC SMASH

8.5.7 MAXTAC Composite

The entire MAXTAC design is shown in Figure 8-8. The primary characteristics of MAXTAC are listed in Table 8-4.

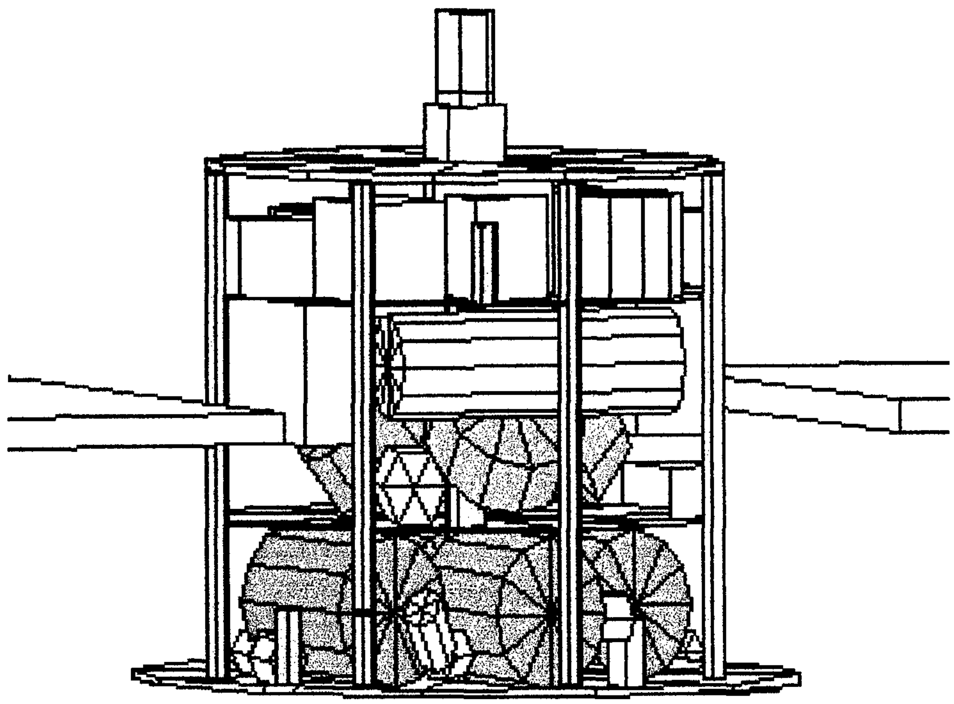


Figure 8-8: MAXTAC (deployed)

Table 8-4: Primary Characteristics of MAXTAC

Mass (kg)	262.5
Height of bus (cm)	71
Average Power (watts)	447
Peak Power (watts)	1000

8.6 Variations on the Baseline

8.6.1 MIDTAC

The MIDTAC alternative is exactly the same as the MAXTAC bus, except that the bottom level is shorter due to the smaller propellant tanks.

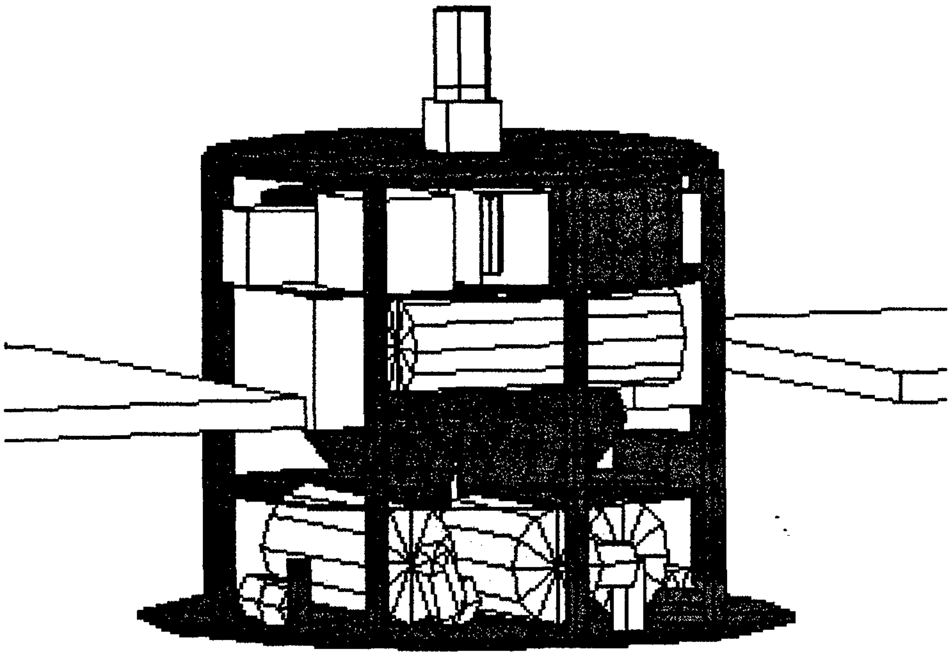


Figure 8-9: MIDTAC (deployed)

Table 8-5: Primary Characteristics of MIDTAC

Mass (kg)	242.9
Height of bus (cm)	68
Average Power (watts)	447
Peak Power (watts)	1000

8.6.2 LOWTAC

The LOWTAC bus is also the same as MAXTAC, but with an even shorter bottom level.

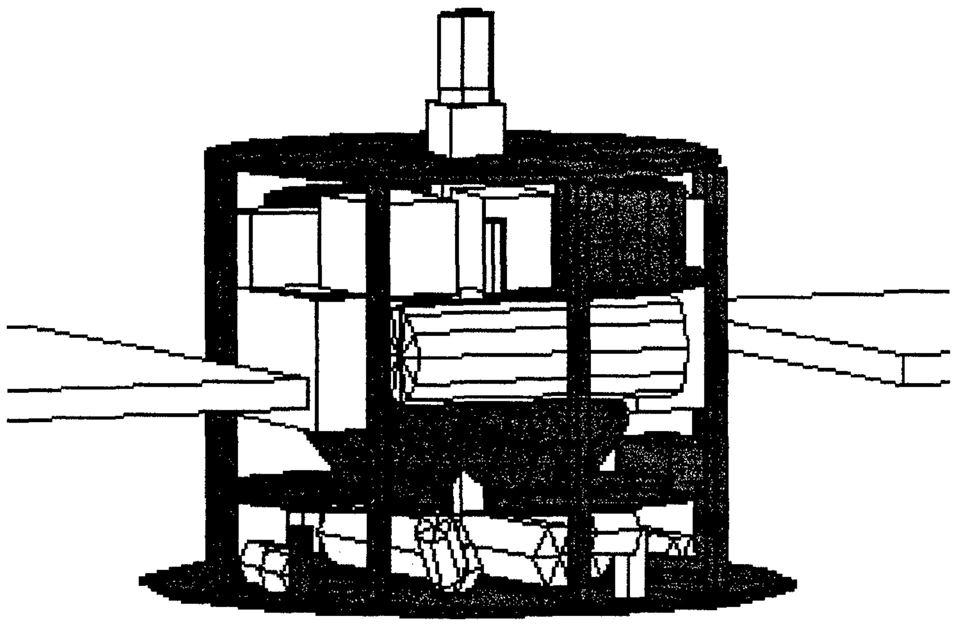


Figure 8-10: LOWTAC (deployed)

Table 8-6: Primary Characteristics of LOWTAC

Mass (kg)	215.7
Height of bus (cm)	63
Average Power (watts)	447
Peak Power (watts)	1000

8.6.3 MAXTAC-N

In the next three alternatives, the layout of the bottom level is the same as in the previous three alternatives. However, there is no SMASH. In fact, since the removal of the SMASH space leaves only a few components on the top level, these components were relocated onto the second level to reduce the height of the bus. But the height of the second level was increased in order to fit all of components within.

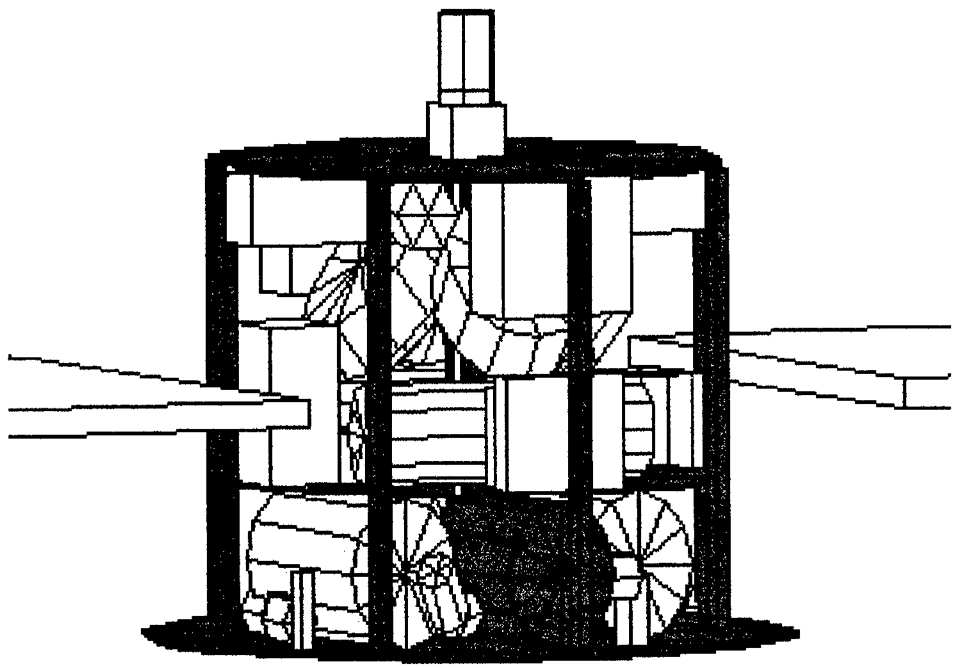


Figure 8-11: MAXTAC-N (deployed)

Table 8-7: Primary Characteristics of MAXTAC-N

Mass (kg)	256.1
Height of bus (cm)	69
Average Power (watts)	447
Peak Power (watts)	1000

8.6.4 MIDTAC-N

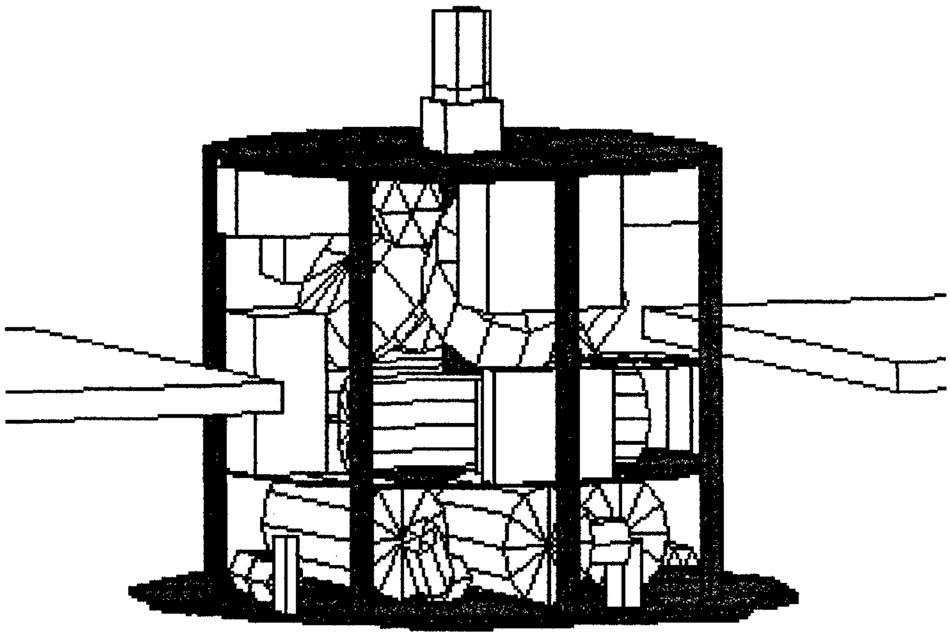


Figure 8-12: MIDTAC-N (deployed)

Table 8-8: Primary Characteristics of MIDTAC-N

Mass (kg)	236.5
Height of bus (cm)	66
Average Power (watts)	447
Peak Power (watts)	1000

8.6.5 LOWTAC-N

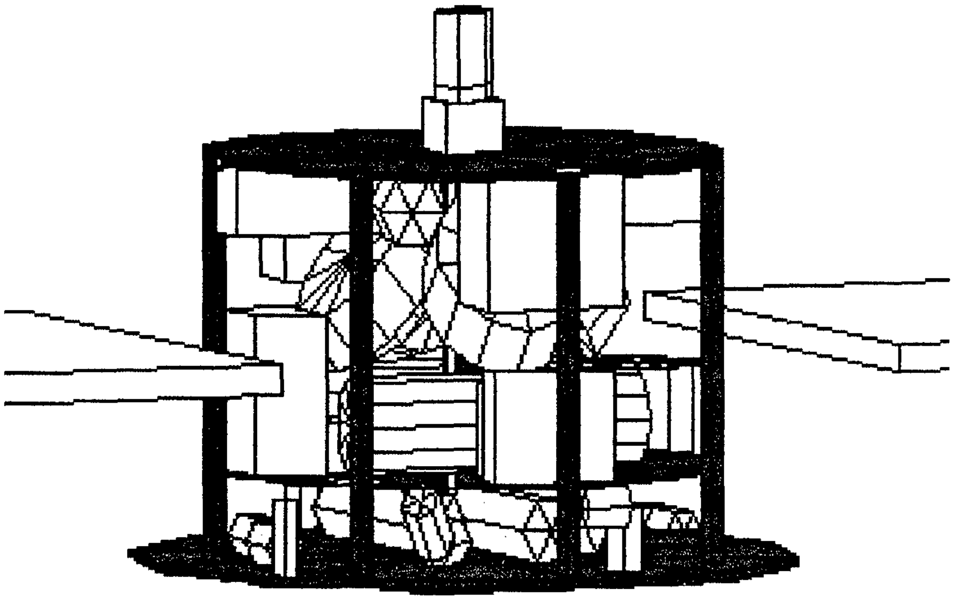


Figure 8-13: LOWTAC-N (deployed)

Table 8-9: Primary Characteristics of LOWTAC-N

Mass (kg)	209.3
Height of bus (cm)	61
Average Power (watts)	447
Peak Power (watts)	1000

8.6.6 MIDTAC-23

Finally, a 23" LV interface version of Modsat was created simply to broaden the solution space somewhat. The MIDTAC platform was chosen, although any of the tactical profiles would have been appropriate. The MIDTAC-23 bus is exactly the same as the MIDTAC bus, except that it is placed higher in the LV due to the interface.

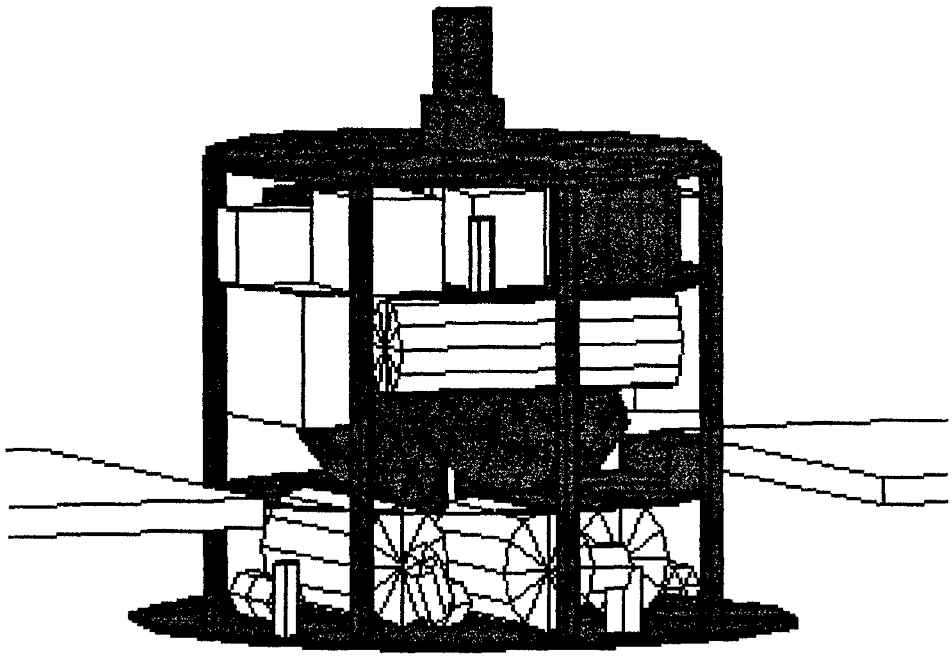


Figure 8-14: MIDTAC-23 (deployed)

Table 8-10: Primary Characteristics of MIDTAC-23

Mass (kg)	292.8
Height of bus (cm)	95.7
Average Power (watts)	390
Peak Power (watts)	950

8.7 Summary

To primary characteristics of each alternative are summarized in , where

1 = MAXTAC
2 = MIDTAC
3 = LOWTAC
4 = MAXTAC-N
5 = MIDTAC-N
6 = LOWTAC-N
7 = MIDTAC-23

Table 8-11: Primary Characteristics of All Designs

Characteristic	1	2	3	4	5	6	7
Mass (kg)	262.5	242.9	215.7	256.1	236.5	209.3	292.8
Height (cm)	71	68	63	69	66	61	95.7
Avg Power (W)	447	47	447	447	447	447	390
Peak Power (W)	1000	1000	1000	1000	1000	1000	950

9. System Analysis

This section discusses the evaluation of the alternative designs. The seven design alternatives are evaluated objective by objective. For objectives with natural measures of effectiveness, the performance value was obtained from the Modsat model. For objectives with attribute scale MOEs, the team rated the performance of each alternative against its attribute scale.

The scoring results of the seven alternatives were extremely close, differing only by 0.0467 in their final weighted score. This is not surprising, since the designs are not vastly different from one another.

The main results for each alternative are presented below in Table 9-1 and Table 9-2. The scores listed under each main objective in Table 9-1 are weighted scores; that is, they are the result of multiplying the utility scores for a given objective by that objective's weight. For a given alternative, the sum of the weighted scores for each main objective yields the overall utility score in the final column.

Table 9-1: Weighted Scores: Standard Weights

Weight	0.1382	0.3119	0.1527	0.1746	0.2226	
	Cost	Responsiveness	Risk	Availability	Utility	Total
MAXTAC	0.07329	0.2038	0.1048	0.1131	0.1123	0.6073
MIDTAC	0.09589	0.1835	0.1048	0.1058	0.1309	**0.6209**
LOWTAC	0.1165	0.1546	0.1048	0.06152	0.151	0.5884
MAXTAC-N	0.07378	0.1962	0.1054	0.1169	0.1134	0.6057
MIDTAC-N	0.09354	0.1763	0.1054	0.0947	0.1296	0.5995
LOWTAC-N	0.1098	0.1469	0.1054	0.06507	0.152	0.5792
MIDTAC-23	0.09937	0.18	0.1048	0.09085	0.09914	0.5742

The final scores for each alternative are presented in again in Table 9-2, where the alternatives have been ranked.

Table 9-2: Ranking of alternatives; standard weights

1	MIDTAC	0.6209
2	MAXTAC	0.6073
3	MAXTAC-N	0.6057
4	MIDTAC-N	0.5995
5	LOWTAC	0.5884
6	LOWTAC-N	0.5792
7	MIDTAC-23	0.5742

Considering the relative scale range of final utility values, the MIDTAC design scores significantly higher than the other alternatives. This comparison is clearly shown in Figure 9-1 below.

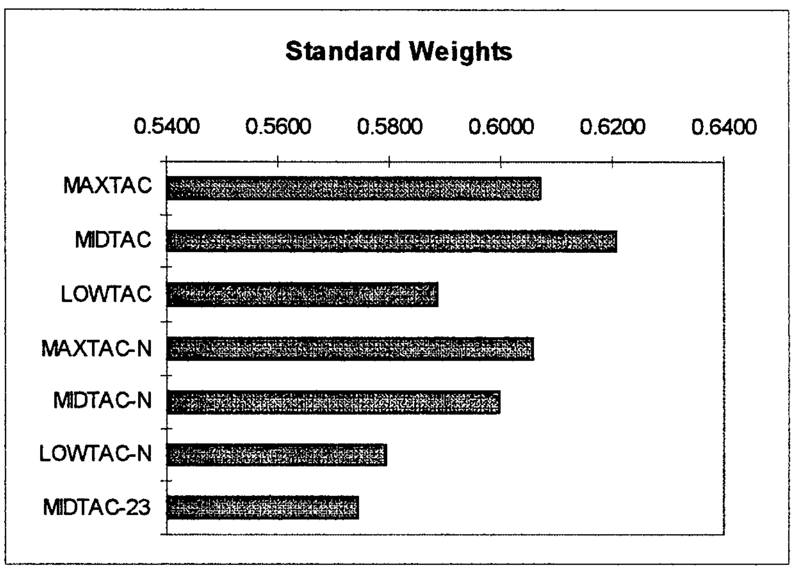


Figure 9-1: Performance at Standard Weights

Many factors combined to give the MIDTAC alternative the highest score. In the two highest-weighted objectives, tactical responsiveness and mission utility, the MIDTAC design fared the best. In the remaining three lower-weighted objectives MIDTAC obtained average scores. The MAXTAC alternative also fared well, receiving the second highest score. This is attributed to MAXTAC's high performance rating for the tactical responsiveness objective, which is the highest-weighted objective in this study. On the opposite end, the MIDTAC-23 alternative scored last, scoring average in four objectives and considerably low in the second highest weighted objective, mission utility. This is due to its usage of a great deal more of the launch vehicle fairing volume, as well as the reduced area available for solar arrays. Although MIDTAC-23 was last, LOWTAC-N was not that far behind, thus reminding the team that scoring well in higher weighted objectives is the key to a better overall score.

10. Decision Making

The results of the previous chapter revealed that the MIDTAC alternative appears to be the best solution. The analysis discussed in this chapter was performed in order to demonstrate the sensitivity of the results to changes in the objective weights, due to shifting environmental factors.

In this analysis, environmental scenarios were created in order to examine the effect of changes in the top-level objective weights. In each scenario, the weight of one of the objectives was increased to correspond with a certain environmental situation. The weights of the four remaining objectives were scaled down proportionally, so that all of the weights would still sum to one. With the new weights, the overall utility function was performed on each of the alternatives, and the results were recorded. In some cases, an alternative other than MIDTAC received the highest score. The scenarios are:

1. Cost is twice as important as its original priority.
2. Responsiveness is twice as important as its original priority.
3. Risk is twice as important as its original priority.
4. Availability is twice as important as its original priority.
5. Utility is twice as important as its original priority.
6. Cost has a full 50% of the sum of the priority weights for all the objectives (cost weight = 0.5).

The rankings of the alternatives, for all scenarios, are shown below in Table 10-1. The MIDTAC alternative is the best solution. It scored in the top three in every scenario, with three first place finishes, two second place finishes, and two third place finishes. The ranks for each alternative, summed over all the scenarios, are calculated in Table 10-2. These results further portray MIDTAC as the superior alternative. It is a well-rounded design that optimizes the tradeoffs between the objectives. Its performance in both the responsiveness and utility objectives, the two most critical attributes of the study, contributes to its high score. Note that MIDTAC scores well in both the responsiveness and utility scenarios, while the other alternatives fail to do so.

Table 10-1: Sensitivity Analysis; All Environments

Rank	Standard	Cost x 2	Responsiveness x 2	Risk x 2	Availability x 2	Utility x 2	50% Cost
1	MIDTAC	MIDTAC	MAXTAC	MIDTAC	MAXTAC-N	LOWTAC	LOWTAC
2	MAXTAC	LOWTAC	MAXTAC-N	MAXTAC	MIDTAC	MIDTAC	LOWTAC-N
3	MAXTAC-N	LOWTAC-N	MIDTAC	MAXTAC-N	MAXTAC	LOWTAC-N	MIDTAC
4	MIDTAC-N	MIDTAC-N	MIDTAC-N	MIDTAC-N	MIDTAC-N	MIDTAC-N	MIDTAC-23
5	LOWTAC	MIDTAC-23	MIDTAC-23	LOWTAC	MIDTAC-23	MAXTAC-N	MIDTAC-N
6	LOWTAC-N	MAXTAC	LOWTAC	LOWTAC-N	LOWTAC	MAXTAC	MAXTAC-N
7	MIDTAC-23	MAXTAC-N	LOWTAC-N	MIDTAC-23	LOWTAC-N	MIDTAC-23	MAXTAC

Table 10-2: Sum of Rankings for the Alternatives

Alternative	Calculation	Sum
MAXTAC	2+6+1+2+3+6+7	27
MIDTAC	**1+1+3+1+2+2+3**	**13**
LOWTAC	5+2+6+5+6+1+1	26
MAXTAC-N	3+7+2+3+1+5+6	27
MIDTAC-N	4+4+4+4+4+4+5	29
LOWTAC-N	6+3+7+6+7+3+2	34
MIDTAC-23	7+5+5+7+5+7+4	40

11. Implementation

This section is intended to aid the CDM in the implementation of the results of the Modsat study. As the purpose of the study was to perform high-level systems engineering, the continuation of the Modsat program will require much more detailed design effort. Moreover, the scope of this study was limited to the design of a spacecraft bus. Many factors and functions must eventually be considered and designed in support of Modsat operations. Recommendations from the team are included summarized as an overall "concept of operations," and are discussed below.

11.1 Continued Design Effort

The systems engineering process is iterative and converging in nature. The scope of each iteration of the design process depends on the current stage within the life-cycle of the program. As the life-cycle progresses, the design process become more detailed. Eventually, the effort converges on an accepted detailed design.

The design information included in this study is relevant for the first stage of the potential life-cycle of Modsat. In this stage, sometimes called "concept exploration," the systems engineer "identifies all reasonable system alternatives that may satisfy the mission need and makes recommendations...; the [CDM] then selects those alternatives or concepts which meet [the] objectives" (Systems Engineering Management Guide, 1989:2-4). If the Modsat program is to progress further, the CDM must build on the concepts and recommendations included in this study.

In the next iteration of the design process, engineers must revisit the selection of components for Modsat, with a view toward optimizing the MIDTAC design. Interfaces must be designed, at the subsystem and component level. In particular, command, telemetry, power, and other signal flows must be examined. The design of software must begin, within the context of the modular satellite operating system recommended by this study. Prototype hardware should be developed to demonstrate the functionality of some of the unique aspects of Modsat, such as its cage structure, or its wrap-around modular solar array assemblies.

At the system level, the mass properties (center of mass, inertia matrix, etc.) must be carefully examined for their effect on stability and control. Control logic should be developed to model the attitude control function of Modsat. Thermal characteristics should be modeled in more detail, and a thermal control system should be designed.

The continued systems engineering effort on Modsat must incorporate concurrent engineering, wherein current engineering efforts reflect consideration of manufacturing, testing, logistics, operational support, etc.

The items mentioned above are just a few of the many challenges awaiting the further design of Modsat.

11.2 Concept of Operations (CONOPS)

The design of a satellite comprises one of the many engineering efforts necessary for the operation of a complete space system architecture. The full architecture encompasses not only the design of the spacecraft and its mission-specific equipment, but also the ground segment (equipment and personnel), the launch segment (equipment and personnel), and the information/ communications architecture (user interface with the system).

11.2.1 Spacecraft Architecture

Implementation of the small tactical satellite design (Tacsat) should take into account the fact that the "baseline" design is generally "over powered" (i.e., the baseline design provides too much average and peak power levels required for operation of most payloads). With this consideration in mind, application of an alternative design architecture (other than the "one size fits all" or "baseline" architecture) becomes desirable for optimization of the bus to the wide variety of mission modules, the majority of which do not require an average power of more than 400 watts or a peak power of over 800 watts. The payloads which may require these high average and peak power loads are those of the active type (LASERs and SARs), whereas passive sensors rarely require more than 100 watts of power (peak -- during a sensing pass). The optimization payoff can be measured in these cases with more available volume on the bus (for other equipment) and less spacecraft bus mass (increasing available mass for the mission module, or allowing a different orbit configuration -- including a higher initial altitude).

Given the fact that the solar arrays for MIDTAC are already a modular design tailorable to specific needs, and the fact that batteries already come in varying sizes for differing capacity requirements, the modular option should be the architecture chosen for initial implementation of the Modsat. Although actual operational configurations (stored and ready for use) for the Modsat will probably mimic the "family" architecture by including "pre-integrated" buses already tailored for either the low-power or high-power mission modules (possibly already integrated with the some mission modules, as well), the modular design of the Modsat provides ultimate reconfigurability to suit changing needs.

11.2.2 Launch Segment

A small air-launched system provides maximum flexibility for the choice of initial orbit for small tactical spacecraft, because of the fact that any launch azimuth may be chosen. This capability also minimizes the time to reach a chosen target (i.e., the direct approach will be the fastest). The *Pegasus* air launched booster currently represents the only launcher in this category. The *Taurus* booster, while not air launched, is more powerful and was developed specifically for rapid deployment, integration, and launch from unimproved areas, making it also a good choice for the launch segment.

The force structure recommended for employment of small tactical satellite designs consists of elements from the 30th, 45th, and 50th Space Wings (30th, 45th, and 50th SW) working in concert. The 1st Tactical Space Launch Squadron (1st TSLS), based at Vandenberg AFB, CA (30th SW) would be responsible for westward (retrograde and Sun-synchronous) launches over the Pacific Ocean . Similarly the 2nd TSLS (45th SW, Patrick AFB, FL)would be responsible for flights east, over the Atlantic. The 19th Space Operations Squadron (19th SOPS), based at Falcon AFB, CO (50th SW) would have responsibility for Modsat launch and early orbit support, Modsat command and control, Tactical Network (TacNet) maintenance and support, and manning and operation of the Consolidated Tactical Space Control Center (CTSSC -- see below) for in-theatre operations support. Under the direction of a Launch Director, each of the Tactical Space Launch Squadrons would operate B-52 or other *(Pegasus)* carrier aircraft. In addition, a spacecraft-to-mission module integration (SMI) crew, a spacecraft-to-launcher integration (SLI) crew, a loading crew, a flight and launch crew, an analysis crew and a launch (command and control) crew (under the direction of an Operations Director at the 19th SOPS and possibly deployed to a mobile or in-theatre site), would accomplish the integration, loading, flight (to launch location over either ocean), launch, and early orbit support for the Modsat mission.

Tactical Space Launch (East Range/Atlantic)

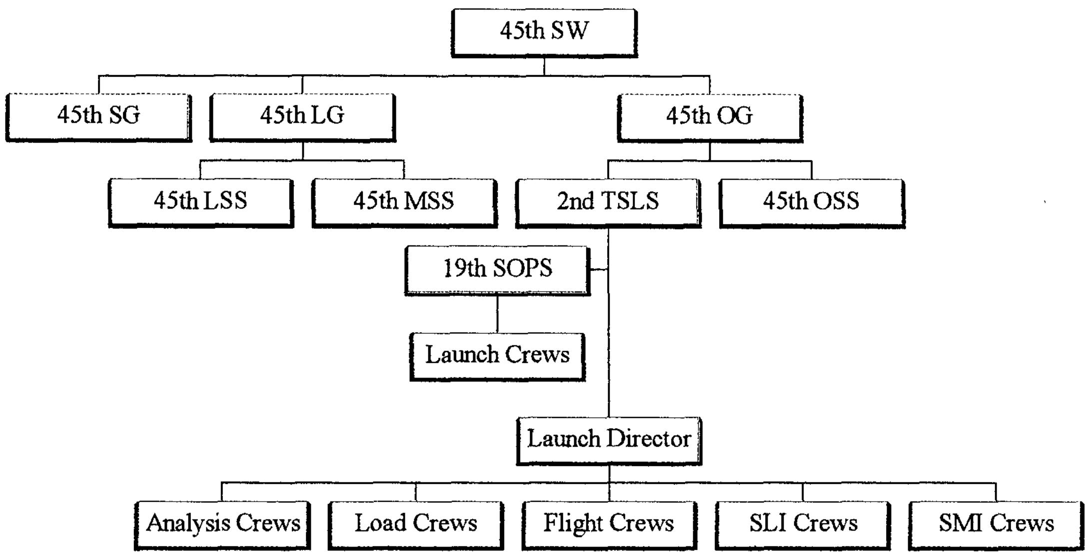


Figure 11-1: Proposed Organizational Structure for Tactical Spacelift (Eastern Range/Atlantic)

The environment for the design study also assumes that a ready supply of launchers, satellite buses, and mission modules is on hand for both rapid response situations and space asset replenishment. Along with the supplies of hardware, other assumptions include ample logistics equipment (transports, "clean" environments, maintenance equipment, special tools, etc.), maintenance personnel, and storage facilities.

Specific personnel for each crew include enlisted-grade crew members and technicians/specialists led by officer-grade flight commanders and deputies (doubling as bus, mission module, and launch vehicle experts for their respective crews), and civilian engineering and analysis personnel (in-depth engineering-intensive positions should be made civilian contractor or GS billets to retain "corporate knowledge" on the systems).

Tactical Space Launch (West Range/Pacific)

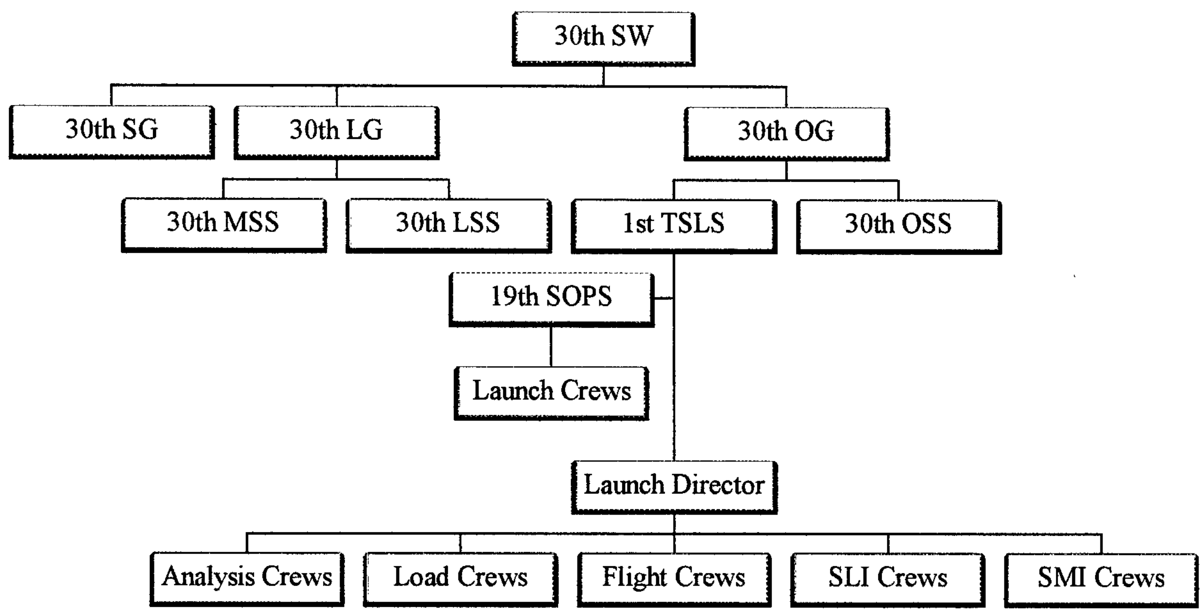


Figure 11-2: Proposed Organizational Structure for Tactical Spacelift (Western Range/Pacific)

11.2.3 Ground Segment and Information/Communications Architecture

The main ground segment for the small tactical satellite should be an X, Ka, or Ku band receiving station, perhaps similar to the Air Force's "Eagle Vision" mobile, in-theatre ground station, which receives mission data directly from both LANDSAT and SPOT satellites, processes the imagery, and overlays the high-resolution visible-region imagery of SPOT with the multispectral imagery of LANDSAT. Multispectral imagery products from Eagle Vision have received rave reviews from operational and theater commanders (Veseley, 1996). Initial operations testing and/or proof of concept testing (for the new tactical satellites) could be set up to utilize current Eagle Vision equipment.

This central receiving, processing, and distribution station, known as the Consolidated Tactical Space Control Center (CTSCC) would be manned and operated by crews from the 19th SOPS (50th SW) and would also incorporate an S band (SGLS/AFSCN compatible) commanding capability for spacecraft specific commanding. CTSSC terminal stations (including a permanent CTSSC at Falcon AFB, CO) would be deployed at several positions on the Earth to ensure full support for any and every theatre of operations. The CTSSC would be the centerpiece for a tactical space information network into which tactical users would input requests and receive information products. User requests/data updates would be transmitted via wireless ethernet protocols (adapted from currently existing internet protocols) carried through either MILSTAR MDR, Teledesic, Iridium, or a similar high data rate, global communications system.

The process would be as follows:

1) User signs on, and requests are encrypted and transmitted from a laptop or similar small computer in the field or from the cockpit.

2) Requests are received, authenticated, and prioritized (set by the Theatre or Operational Commander) by CTSCC server.

3) CTSCC server queries (via intelligent agents) the types and availabilities of appropriate TACSATs (based on the type(s) of information requested by the user) as well as orbit predictions for those required assets which are available.

4) The server calculates time over target, time to receive mission data, time to process, filter, and package, and time to transmit requested information to the user.

5) If the requested information is available, the CTSSC server replies with the product; if not available, the CTSSC server replies with an estimated time of arrival (ETA) for the product, based on its calculations in 4).

These tasks and the artificially intelligent systems and software required to accomplish them are possible with current computer and software technology and may be made functionally "modular" (in both hardware and software design) to ensure future upgrade capability. To make this system work well for an easily estimable "many" users, the communications systems employed and/or utilized will have to support the necessary bandwidth to successfully support the tactical user in a timely fashion; otherwise, the CTSCC's tactical support capability will be degraded. Centralizing the processing power of this tactical space information system in the CTSSC (as opposed to performing the data-to-information processing on the satellites) allows more processing power to be utilized, allows easy upgrades to software, allows simpler troubleshooting of software, retains the satellites' simplicity -- both from the hardware and the software standpoints, retains central control authority for information dissemination, promotes optimal tasking of resources (again, due to centralized control and prioritization), simplifies communications to and from the satellites (multiple channels are unnecessary), and allows future upgrades to the overall architecture to perhaps "evolve" into a system which incorporates more "on board" processing and direct downlinking to users. Having the information processed by the ground segment is the simpler, more powerful choice for current technology.

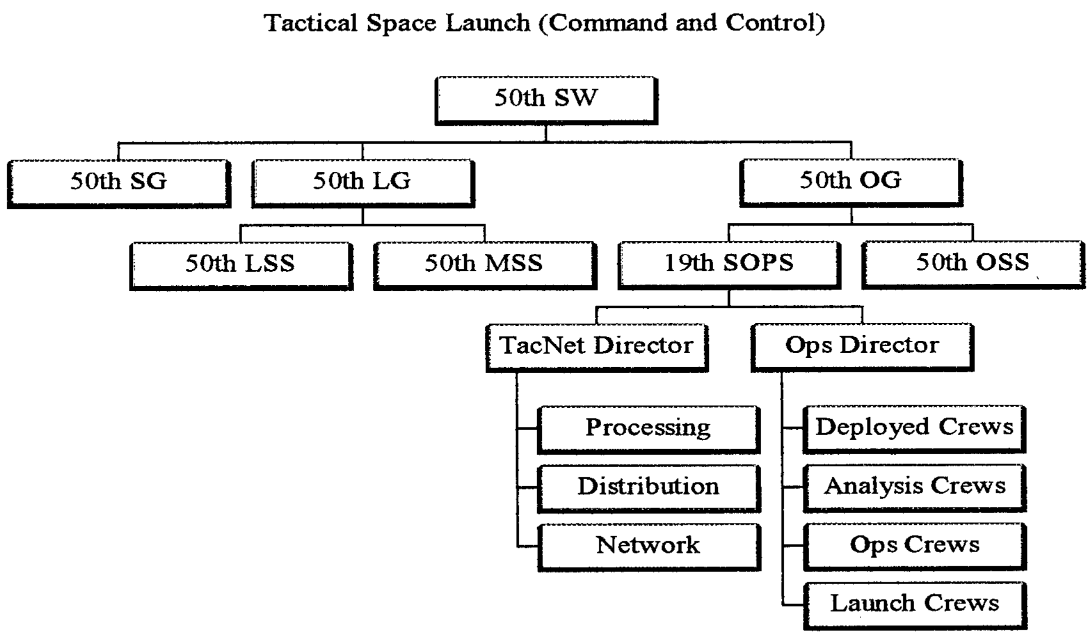


Figure 11-3: Proposed Organizational Structure for Tactical Spacelift (Command and Control)

An alternative to the (assumed baseline) ground-based CTSSC would be a dedicated command and control aircraft, or the CTSCC could be palletized and flown aboard a C-17 or C-5, further enhancing its mobility.

12. Future Technologies and Continuing Investigations

The preliminary design for a small tactical spacecraft bus provides a solid foundation for further investigations as well as continued expansion of the Modsat computer design model. Design efforts generated a tremendous amount of information in addition to producing the Modsat computer-based design tool and a design for a standard, tactically applicable satellite bus. Much of this information was not included within the formal system process either due to the planning horizon for the proposed design (five years) or because of the "nonessential" nature of much of the information (to the design). These topics nonetheless sparked many discussions during meetings and comprise an interesting set of ideas and technologies on which to base a possible future design study (or design studies). The design process also generated several concepts for future scientific research, operational analysis, and system design.

12.1 Modsat Computer Model Enhancements

The Modsat computer model, which was developed to aid in the design and evaluation of small tactical satellite designs, provides a foundation for further enhancements, additions, refinements, and detail. Modsat's coding, though large in scope, is rather simple in style, and it is thoroughly documented internally. Future additions or modifications to the code may include:

- more integration of orbit analysis features
- additional mission module configurations and/or types
- additional cost models

- more detailed component or subsystem modeling
- larger libraries of selectable components
- future technology modeling
- more in-depth launch vehicle modeling and/or design
- more detailed interface modeling

Detailed discussion of the individual sections and functions (background, scope, functionality, limitation(s), and future feature suggestions) of the Modsat computer model can be found in Volume III in the Modeling section.

12.2 Constellation Design

Multiple satellites, carrying active sensors, may in the future be employed in concert as an "array" of sensors. The operational challenge of such a constellation can be solved either by maintaining relative satellite positions and attitudes to within close tolerances (within a few millimeters), or by maintaining extremely accurate position and orientation knowledge (to within a few millimeters). The second solution is the more feasible of the two, because satellites equipped with multiple GPS receivers can produce these knowledge products to within the accuracy necessary (NASA, 1995: sec.7, p.3), and with this knowledge ground-based computer processing can correct for discrepancies in received signals (between spacecraft). A constellation of this type would have the potential of creating an extremely large synthesized aperture; however, the processing power required for a constellation of more than two or three satellites would no doubt be tremendous and possibly prohibitive with current capabilities.

Further analysis may be performed regarding the types of orbits employed for tactical satellites. The primary orbit chosen as the baseline for this study was a Sun-synchronous orbit at an altitude of 350 kilometers. This orbit type has definite advantages for remote sensing and Earth resources missions; however, other orbits exist which may provide greater utility for specific missions. The highly eccentric Molniya orbits provide long dwell time, but at a geosynchronous altitude during that dwell time; this orbit type may be useful for tactically-applied communications satellites designed for short mission durations. "Walker" orbit parameters (Wertz, 1992: 189-192) provide a useful method for construction of constellations with specific coverage goals in mind. One example is that of multiple satellites being "staggered" in one or more specific orbital planes such that, as one satellite "sets " on a target, the next satellite in line comes into view. Constellation design and optimization is a complex art. Space planners and designers may wish to examine the advantages and disadvantages of the coverages, dwell times, revisit times, and environmental stresses associated with various orbit types.

12.3 Logistics and Operations

There are logistical challenges involved with the transport, storage, and maintenance of a (large) supply of not only tactical satellite buses, but also mission modules (of multiple types), small launch vehicles, and support equipment (and support aircraft in the case of air-launched spacelift). Another possible (albeit novel) logistical and/or operational strategy study would be to investigate possible ways of recovering the satellites -- either by retrieval or reentry -- and gauge their feasibility and utility in a tactical environment constrained by costs and the increasing desire to reuse hardware.

Further analysis may be performed on the actual missions and/or applications for the tactical satellite, not only on the sensing and support (of terrestrial forces) aspect of space based platforms, but also on the possible force application roles of such platforms.

The air-launched ICBM test program of the 1970s and the current "ALT-Air" program sponsored by the Ballistic Missile Defense Organization (BMDO) have demonstrated the feasibility of a carrier aircraft-based, palletized launch scheme. In this scheme, a launch vehicle is transported inside the carrier aircraft, deployed off of an aft-ejected, parachute-assisted pallet with a stabilized drogue chute, and ignited. This launch scheme could be applied to a new launch system specifically designed for this purpose. Investigations could include adaptation of an existing system to this scheme (like the ICBM tests of twenty years ago). This transport environment has the potential of providing a much more benign environment (vibrational and thermal) for the launcher and spacecraft payload, as well as possibly supporting a "clean" environment (e.g., a "clean tent" such as used by the *Pegasus*) inside the aircraft itself during transport -- due to the fact that the interior of the aircraft is environmentally controllable. Additionally, regular loading crews would need no special training for handling the rocket pallet, since the pallet would be the same as any other pallet fitted for that particular aircraft.

12.4 Mission Modules

This design study focuses specific design determination efforts upon only the satellite bus; however, due to the particular design paradigm, the primary design standards imposed upon the mission module designer may be enumerated. Though the standard small tactical spacecraft bus will provide support for the mission modules, mission module designers will be required to design "to the bus", as opposed to the bus being built for the specific payload equipment. Analogous to the interface requirements to which underwing stores on a fighter must be designed, the mission modules must conform to certain electrical, mechanical, data protocol, telemetric, and software interfaces incorporated into the bus design. Many of the specifications for these interfaces will be provided by the SPIG (see Tradeoffs).

Table 12-1 summarizes the generally specified requirements for the design of the various mission modules.

Table 12-1: Mission Module Design Requirements

Design Consideration	Mission Module Design Requirement
Mission Scope	single-sensor type; narrow mission specification
Mission Mass	under 120 kg
Mission Power	average under 320 W peak under 820 W
Mission Volume	under 0.6 m
High Data Rate Downlink	must be integral if greater than SGLS rate is desired
Data Storage Capacity	must be integral if greater than 2Gbytes is desired (unless storage is modular)
Mechanical Interface	conform to SPIG
Electrical Interface	28 V regulated bus standard; conform to SPIG
Telemetry/Software Interface	compatibility with bus standard formatting; specialized mission software extensions (to SOS) must be integral
Thermal Environment	isolation from bus; specialized mission equipment integral
Design Focus	tactical; minimize testing time; minimize warmup time

The mission modules modeled in the Modsat computer model are necessarily "generic" in nature to provide both flexibility in design evaluation and a foundation on which more specific types may be modeled in both the current version and in future versions.

Future operational analysis may focus on optimizing the functionality of different types of mission modules and putting together the most tactically useful, easily storable, quickly integratable, and technically feasible combination of mission modules for tactical space missions.

A specific mission module for performance of a "LASER designator from space" role was not specifically addressed by the system design study, due to the number and variability of the many factors involved in the mission analysis for such a mission module, as well as the "experimental" nature of any such mission module if constructed with current technology. A mission module of this type may be roughly modeled, however, with the LASER/LIDAR mission module tools incorporated in the Modsat computer model. A future trade study on the design of such a mission module would require analysis of 1) illumination efficiency, power, and wavelength(s); 2) target reflectivity/signature in the given wavelength(s); 3) detector positioning (azimuth, elevation, and altitude), sensitivity in the given wavelength(s), field of view, signal to noise ratio, and velocity; and 4) possible adversarial countermeasures and spoofing. Finally, sizing of the mission module's volume, mass, and power requirements may or may not make this application feasible for a small satellite application. Of course, a primary goal of this type of research would be the determination of "payoffs" in costs, manpower, equipment, capability, and responsiveness that this type of system may or may not achieve.

A similarly experimental application (and as worthy or more worthy of further investigation) being developed for LASERs is that of extremely high data rate communications systems. Many of the tradeoff factors and design considerations involved in the design of a LASER designation system are applicable to the design of a LASER-based communications system (i.e., power, sensitivity, positioning, signal to noise ratio, field of view, and , of course, atmospheric attenuation).

12.4.1 Small Satellite Technologies "On the Horizon"

Many novel and exciting (as well as very technically challenging) technologies promise to change the face of satellite design in the not so distant future. All of these technologies are expected to be developed within the next ten years.

Flywheel Technology -- Flywheels provide power and momentum storage through the utilization of kinetic energy storage. These structures represent potentially lighter weight and higher capacity than chemical-based batteries, with the added functionality of naturally stabilizing a spacecraft in (as do traditional momentum wheels). This "functional density" (in which one component performs more than one function) is a popular theme for small satellite design, and is already evident in most designs for spacecraft CPUs, as multifunctional microprocessors are becoming the norm for small satellites (Hively, 1996).

Lithium-Ion Batteries -- These batteries provide vastly higher capacity than traditional and current technology chemical batteries (see Vol II, Tradeoffs, Electrical Power Subsystem).

Inflatable Structures -- This technology will allow smaller spacecraft buses to support much larger, more capable active sensors, such as those required for synthetic aperture RADAR. Current efforts are underway at NASA to produce electronically steerable, high-resolution RADARs for launch on small vehicles, but inflatable structure technology would significantly reduce payload mass and required payload fairing volume (scaling down required volume from "cubic feet" to volumes on the order of "cubic inches"), thereby freeing up space on the booster for other experiments (on the spacecraft) or other vehicles (within the fairing) (NASA, 1995).

Global Positioning System (GPS) Applications -- GPS promises to provide much better accuracy and more timely and autonomous orbital position prediction and tracking than current methods of ground tracking. Utilization of GPS will free up much of the overtaxed Air Force Satellite Control Network (AFSCN) from mundane "tracking" supports for the new vehicles equipped with GPS receivers. Experiments in the future will also include single vehicles equipped with multiple receivers, testing GPS capability to determine spacecraft attitude. If this application proves functional, it will relieve much of the attitude control system requirements for attitude knowledge sensors, thereby further reducing spacecraft mass.

"Toroidal" Propellant Tank -- This propellant tank design was borne out of system synthesis efforts as a theoretically more efficient propellant tank packaging scheme, optimizing available volume within a spacecraft.

Fourier Transform Hyperspectral Imaging -- This imaging package under investigation at Phillips Lab represents a new paradigm in multispectral imaging -- spectral resolution approaching that of gas chromatography and/or spectrometers (i.e., evolution toward a "continuous spectral imaging system" paradigm) through the usage of Fourier Transform systems for spectral separation. Improved spectral resolution, lighter instrument weight, and more efficient transmission (than the current "best" method of dispersion gratings) is achieved through Fourier Transform separation (Hagan, 1996;Otten and others, 1995).

Pulsed plasma thrusters -- These and other low-thrust, high specific impulse, non chemical propulsion systems will provide lower thrust, but more total delta-velocity capability (per unit mass) over the life of the satellite than chemical thrusters (Hagan, 1996). Experiments utilizing PPTs are planned for the Phillips Lab's MightySat program.

Ka-Band Transmitter Experiment -- Another experimental payload project for the Phillips Lab MightySat program, this phased array communications package will provide testing and validation of technologies expected to reduce the mass, moving parts, and spacecraft attitude adjustments required to track a signal from a communications ground station. It will also study high data rate modulation techniques.

Small liquid-fueled booster -- Solid rocket motors (SRMs), while inexpensive and easily adaptable to small launch systems, are heavy, toxic, fragile, less flexible, and less reliable (in general) than liquid rocket-based launch systems. Phillips Lab and other research organizations (as well as some sectors in industry) are developing prototypes of a simplified "blow down" pressurized liquid-fueled rocket for use as a small launch system (Warner, 1996; Worden, 1996). This system incorporates propellant, oxidizer, and an inert pressurant (helium) to inject the propellant and oxidizer into the combustion chamber. This system can be built with few or no moving parts, and, by virtue of being "throttleable" the system's performance can be fine-tuned and/or trimmed during flight (as opposed to a SRM, which may be vectorable, but not dynamically thrust-variable while in-flight), thereby increasing initial orbit accuracy. These types of simple, small rockets could eventually replace SRMs in most applications (e.g., as an air-launched system).

13. Conclusions

Although the individual products and ideas generated throughout this study may be individually worthy of merit, the three important results of this study fully characterize the synergism of the effort. The utilization of an adaptable (i.e., specific to the task at hand and the circumstances of the environment) System Design Process made all of the effort possible and productive. The generation of a feasible, value-added, "clean-sheet" MIDTAC design for a small tactical satellite bus provides a basis for further development of "tactical space" or "TacSpace" concepts. The construction of a generic and modular (i.e., robust, modifiable, expandable) Modsat Computer Design Model, though the greatest challenge of the effort, provides a useful, valuable design platform for use by both future researchers and students.

13.1 Modsat Model

The construction effort involved with the development of a fully integrated computer design and analysis tool for small satellites comprised a systematic design process in itself. As with the individual subsystem component choices, individual subsystem modeling sections formed along baseline component characteristics determined by the subsystem trade studies. The value system determined by the team formed the basis of the analysis section of the model. The Modsat modeling software package provides for analysis of physical characteristics, mission performance, and overall costs.

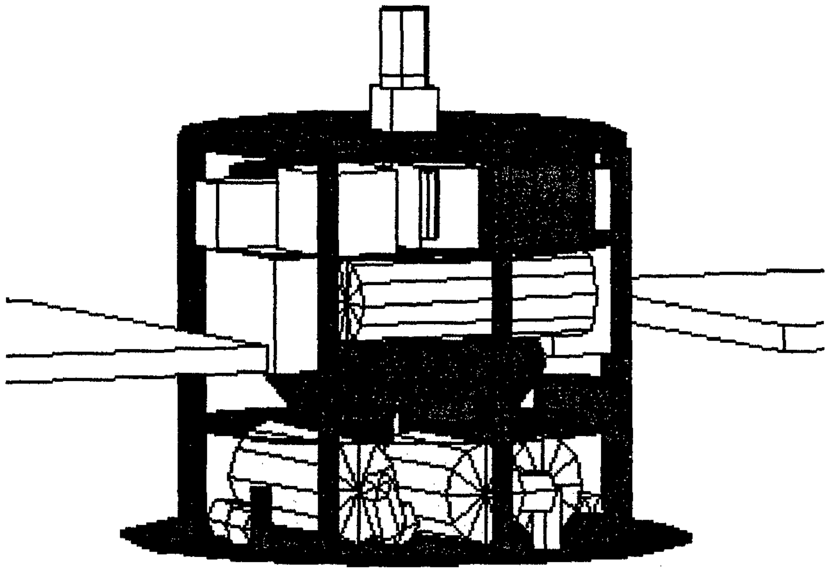
The Modsat model provides a foundation for further analysis efforts. The underlying functions of the software may be modified or expanded according to a particular user's requirements and objectives. A vast array of different sizes, shapes, materials, and other characteristics may be modeled, based upon user input. The initial version of the model provides estimations which may be updated (through modification of the underlying code) with evolutions in space technology or changes in design philosophy. Aside from these more esoteric considerations, the software, of course, may be used for the analysis of small satellite designs other than those analyzed in this preliminary study. Using differently modeled components and differently modeled value systems, the Modsat modeling tool has the capability to produce a wide range of possible designs.

13.2 Design Concept

The MIDTAC spacecraft bus provides a generic design that may be further developed for specific applications or more completely engineered for actual production. The MIDTAC bus, as a complete system, has been designed to the extent that feasible bus enhancements may be easily explored and developed. As recommended by this system study, the expected implementation of the MIDTAC bus incorporates a modular power generation system to provide tailoring of power levels required by various mission modules. The actual design may also incorporate other modular and/or tailorable components or subsystems.

The next step in the development of this bus design should be an engineering study of the interfaces required to integrate all of the individual satellite subsystems and components. While the MIDTAC design includes physical characteristics and a "working concept", this design is only the first step in development process; it comprises the results of an initial "concept exploration" phase for a new space system. Concurrent engineering studies concerning manufacturing, integration, and testing for the MIDTAC design must also be accomplished. Finally, mission modules must be designed for flight testing and operation on the MIDTAC bus.

Figure 13-1: MIDTAC Bus and Vital Statistics



<u>Physical Characteristics/Capabilities:</u>
Mass: 243 kg
Available Power (average/peak): 319/827 W (tailorable to mission)
Available Mission Mass: 75 kg (350 km, sun-synch)/200 kg (350 km, 28.5 deg)
Available Mission Volume: 0.7855 m^3
Pointing Accuracy/Attitude Knowledge: 0.1 deg/0.05 deg (nominal)
Data Storage: Modular 2Gbyte SSDR (tailorable to mission)
Delta-velocity Capacity: 300m/s

<u>Subsystem Features:</u>
Mission Modules: SPIG interfaces, EO, MSI, LASER/LIDAR, SAR
ADCS: three-axis stabilized, reaction wheels (4), star sensor, Earth sensor, IMU
Propulsion: monopropellant thrusters (6), cylindrical tanks (4)
Structural: octagonal "cage" structure, "3-level" shelf system, SMASH option
EPS: NiH_2 batteries, modular GaAs solar arrays, decentralized distribution
TT&C: SGLS compatible, 1024 kbps nominal downlink, Satellite Operating System (SOS), TCP/IP protocol

The MIDTAC design ultimately provides a solution to one portion of a much greater, underlying problem facing modern space efforts (military, scientific, and commercial). More responsive, less expensive, and more efficient access to space is an issue which requires new and innovative approaches to not only spacecraft design (the focus of this study), but also spacelift, command and control, and information processing and distribution. In addition to providing a tactically applicable satellite bus, the MIDTAC design will provide the Air Force with ready-to-fly space research platforms. The space environment will be more accessible to technology demonstrations, developmental payloads, and other space experiments by having these standard buses (with standard mission interfaces) readily available. The MIDTAC design essentially provides to the Air Force a standard vehicle -- putting the "horse" before the "cart" -- on which it may more readily and effectively conduct space operations and technology development. MIDTAC will allow the Air Force to more quickly accumulate valuable "spaceflight time" and experience. Only with increased "hands on" experience in spaceflight and space operations will the Air Force fully evolve into its role as a "Spacepower". MIDTAC provides the means to that end.

13.3 System Process and Beyond

The start of the system design began necessarily with a generalized, high-level treatment of the problem posed by the CDM: the generation of a "clean sheet" tactical satellite design for use as a "multirole" satellite, capable of supporting a wide variety of payload (mission module) types. This design should be easily and inexpensively produced for the Air Force, smoothly integrated by a primarily "blue suit" special weapons crew, and quickly launched for either quick-response or asset-replenishment missions. Initial considerations involved the various high-level (nonspecific components) design and implementation approaches to this problem; further efforts evolved to focus efforts on creating the "baseline" or "point design" with which the original design approaches could be considered in combination. This combination of design (the MIDTAC) and approach (i.e., modularized components) produced an optimal solution.

The (satellite) functional division of effort, in addition to the assignment of system process responsibilities among team members, was key to success. The convergence of the overall design and the development of the Modsat modeling software required expert knowledge of individual subsystems. Consideration of generic remote sensing mission modules provided baseline requirements for the bus design. The system, reliability, and subsystem trade studies narrowed the solution space for the bus design to a point where baseline designs could be generated. Effective coordination of the overall system design effort required central coordination for each of the system process phases: problem definition, value system design, system and subsystem design tradeoffs, system synthesis, modeling and optimization, decision making, and implementation. This approach not only allowed the systematic construction of a robust design and analysis tool, but also the synthesis of seven fully integrated, fully characterized designs which could then be thoroughly analyzed and evaluated.

The basic validity and robustness of the process may be fully and objectively realized when considering the role of the CDM. Other than an initial, broad design philosophy, the CDM provided little input. Faced with this "ill-posed" problem, the team created a process which allowed a "natural" evolution of objectives, focusing of efforts, convergence of designs, and, most importantly, solidification of goals. These goals included the construction of the Modsat model and the development of a "baseline" design. Modification of either the assumptions of the problem definition or the value system (based upon the preferences of the CDM) may yield vastly different results. This allows the adaptability of the process to other space system design projects.

This process is unique, innovative, and goes beyond traditional system and satellite design approaches. Although Hall's Process and the SMAD process were used as references and process "baselines", these approaches ultimately proved unsuitable, due to the unusual design paradigm assumed by the team. The "mission module" is a concept wherein the payload equipment must be integrated to the satellite bus and conform to bus-constrained design parameters. This is a design paradigm unpopular within the aerospace industry, and runs contrary to the traditional approaches to spacecraft design prescribed by the SMAD process. Considering the stagnation of space efforts and the lack of relief from the high cost and low availability of space access, the pursuit of an alternative spacecraft design paradigm was reasonable, if not necessary. The team required, and ultimately developed, a process that was less dependent upon mission-derived design specifications and allowed more design freedom to consider the wide range of spacecraft design possibilities.

Finally, the process is, unlike the SMAD process, synergistic in approach, and it epitomizes the concept of "functional density," wherein the process served many purposes simultaneously. The research involved in the subsystem trade studies produced the component data for use in the modeling tool, narrowed the scope of subsystem design choices, and refined the design objectives and value system. The iterative process of the synthesis of individual designs solidified design alternatives and also refined the model. The ultimate results of the development and application of this process resonate beyond the generation of the MIDTAC design and the development of the Modsat model. The results of this study show that, as a solution to the "space access" problem, an alternative to the current design paradigm (i.e., "generic" versus "specific") is not only feasible but desirable. This study, its process, and its products set the standards of creativity, innovativeness, adaptability, and robustness for future spacecraft design efforts. The Modsat design team has set the bold example that others will follow.

Bibliography

1. The Aerospace Corporation. Consolidated Mission Architecture. The Aerospace Corporation, El Segundo, California, Oct-93.
2. Aerospace Division Olin Defense System Group. Hydrazine Handbook. Redmond, WA: Rocket Research Company, 1995.
3. Aerospace Systems Division, 1996: http://www.ball.com/corporate/haerobas.html.
4. American Institute of Aeronautics and Astronautics, Inc., Phillips Laboratory Homepage, 1996: http://www.plk.af.mil.
5. Arizona State Univ. Statement of Work for the MightySat Phase II Program (Revision 4). Phillips Laboratory, Kirtland AFB, New Mexico, 31-Mar-96.
6. Asker, James R. "Ball Drops Pegasus Plan, Selects Lockheed Launcher," Aviation Week & Space Technology, : 58 (December 6, 1993).
7. Asker, James R. " Quick Response Key to Next U.S. Launcher," Aviation Week & Space Technology, : 44-46 (September 28, 1992).
8. Asker, James R. " Speedy Fix Eyed For Pegasus XL," Aviation Week & Space Technology, : 21-22 (July 3, 1995).
9. Athey, Thomas H. Systematic Systems Approach: An integrated Method for Solving System Problems. New Jersey: Prentice-Hall, 1982.
10. Bate, Roger R., Mueller, Donald D., and White, Jerry E. Fundamentals of Astrodynamics. New York: Dover Publications, 1971.
11. Bearden, David, and others. "Cost Modeling," Reducing Space Mission Cost. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1996.

12. Beer, Ferdinand P. and Johnston, E. Russell. Aircraft Structures. USA: McGraw-Hill Book Company, 1982.

13. Berget, Richard T., and Turner, Tim. "Command and Data Handling," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

14. Bible, Steven R. "The World Wide Web Amateur Satellite Ground Station," Proceedings of the AMSAT-NA 13th Space Symposium. Newington, CT: American Radio Relay League, 1995.

15. Boas, Mary L. Satellite Orbit Analysis Program (SOAP) Manual. The Aerospace Corporation, El Segundo, California, Apr-96.

16. Boden, Daryl G. "Introduction to Astrodynamics," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

17. Bonometti,R.J., and others. "DARPA Initiatives in Small-satellite Technology", SPIE Proceedings: Small Satellite Technology and Applications, Vol.1495: (1991).

18. Broadsky, R. F. "Defining and Sizing Space Payloads," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

19. Bruno, Pattan. Satellite Systems Principles and Technologies. NY: Van Nostrand Reinhold, 1990.

20. Chaggaris, Elias. Kearfott Capabilities in Space. Wayne, NJ: Kearfott Guidance and Navigation Corporation, 1996.

21. Chaggaris, Elias. Space Qualified Kearfott Inertial Reference Unit. Wayne, NJ: Kearfott Guidance and Navigation Corporation, 1996.

22. Chiang, J. J. and Kim, D. J. Mechanics of Materials. Boston, Massachusetts: PWS Engineering, 1984.

23. Chow, Brian G. An Evolutionary Approach to Space Launch Commercialization. Santa Monica, CA: RAND, 1993.

24. Clemen, Robert T. Making Hard Decisions: An Introduction To Decision Analysis. Belmont, CA: Duxbury Press, 1996.

25. The Clementine Team. Clementine II Concept Design Review. Naval Research Laboratory, CA. 14-Mar-96.

26. Davies, Richard S. "Communication Architecture," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

27. Dawdy, A.B. and Holker, D. L. An Approach to Rapid Payload Integration: The Spacecraft-To-Payload Interface Guideline (SPIG) Version 1. The Aerospace Corporation, El Segundo, California, Apr-96.

28. Day, Gregory R. and Lewis, Kelly D. "Strength Analysis". Spacecraft Structures and Mechanisms. Ed. Sarafin, Thomas P..Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

29. Department of the Air Force. FLTSATCOM Orbital Operations Handbook. The U.S. Air Force, 1977.

30. Dezelan, R. W. Cost Savings Recommendations for Warfighters Integrated Next Generation Space Systems (WINGSS). The Aerospace Corporation, El Segundo, California, Mar-96.

31. Dezelan, R. W. Distributed Satellite Formation Alternatives. The Aerospace Corporation, El Segundo, California, Mar-96.

32. Dornheim, Michael A. " Lockheed Must Prove LLV Reliability by June," Aviation Week & Space Technology, : 18-20 (August 21 1995).

33. Dornheim, Michael A. " Lockheed Completes LLV Mockup Stacking," Aviation Week & Space Technology, : 69 (August 1, 1994).

34. Dornheim, Michael A. " Range Video Captures LLV-1 Destruction," Aviation Week & Space Technology, : 53 (September 4, 1995).

35. Dornheim, Michael A. " Vectoring, IMU Destroyed LLV-1," Aviation Week & Space Technology, : 96-97 (December 18/25, 1995).

36. Doukas, Peter G., McCandless, James R., William R.Sarafin, and Thomas P. "Structures and Mechanisms," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

37. Doukas, Peter G., McCandless, James R.Britton, William R.Sarafin, and Thomas P. "Spacecraft Configuration and Structural Design". Fundamentals of Space Systems. Ed. Pisacane, Vincent L. and Moore, Robert C. New York: Oxford University Press, 1994.

38. Doyle, Christine. Space Inertial Reference Unit. Goleta, CA: Litton Guidance and Control Systems, 1996.

39. Dueber, Ross E. and McKnight, Darren S. Chemical Principles Applied To Spacecraft Operations. Malabar, Florida: Krieger Publishing Comp., 1993.

40. Eshbach, Ovid W. Mathematical Methods in Physical Sciences. NY: John Wiley & Sons, 1983.

41. Eterno, John S. "Attitude Determination and Control," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

42. Everett, James. Systems Engineer for Small Explorer program, NASA Goddard Space Flight Center Telephone interview. 3 October 1996.

43. Fleeter, Rick, and Warner, Richard. "Design of Low-Cost Spacecraft," Space Mission Analysis and Design. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1992.

44. Fleeter, Rick. "Reducing Spacecraft Cost," Reducing Space Mission Cost. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1996.

45. Ford, John. "Communications," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

46. Gere, James M. and Timoshenko, Stephen P. Mechanics of Materials. USA: McGraw-Hill Book Company, 1981.

47. Gorney, D.J., Blake J.B., Koons, H.C., Schultz, M., and Vampola, A.L. "The Space Environment and Survivability," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

48. Hagan, Joel. "Space Technology". Electronic mail. 16:40:14 EST, 11 October1996.

49. Hall III, Arthur D. "Three Dimensional Morphology of System Engineering", IEEE Transactions on System Science and Cybernetics, SSC-5, No.2: 156-160 (Apr-69).

50. Harvey, E.L. "Low-cost Spacecraft Buses for Remote Sensing Applications", SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

51. Hecht, Herbert. "Reliability During Space Mission Concept Exploration," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

52. Hecht, Herbert. "Reliability Considerations," Reducing Space Mission Cost. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1996.

53. Hecht, H. and Hecht, M. Reliability Prediction for Spacecraft. Rome Air Development Center, NY. December 1985 (RADC-TR-85-229)

54. Higbee, Terry A. "TECHSTARS: Small, Smart Space Systems," Small-Satellite Technology and Applications. 103-114. Washington: SPIE-The International Society for Optical Engineering, (4-5 April) 1991.

55. High Performance Gyro Assembly. Guidance and Control for Space Applications. Teterboro, NJ: Allied Signal Aerospace, 1996.

56. High Torque Momentum Reaction Wheel Assembly. Guidance and Control for Space Applications. Teterboro, NJ: Allied Signal Aerospace, 1996.

57. Hively, Will. "Reinventing the Wheel", Discover, . Aug 96, pp. 58-68.

58. Ithaco Space Systems, 1996: http://www.newspace.com/Industry/Ithaco/home.html.

59. Jackson, Jon D. LCOL, USAF (SMC/TE) Telephone interview. 24 October 1996.

60. Jeffrey, W. "Small Satellites (MSTI-3) for Remote Sensing: Pushing the Limits of Sensor and Bus Technology", SPIE Proceedings: Platforms and Space Systems, Vol. 2317. (1995).

61. Jet Propulsion Laboratory. Flight Hardware Survey (Revision # 1). Pasadena, CA: Jet Propulsion Laboratory, 1993.

62. Kilston, S., and others. "Multi-mission Sensor for the RESERVES Small-satellite Program", SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

63. Kitts, C.A. "Initial Developments in the Stanford Squirt Program", SPIE Proceedings: Platforms and Space Systems, Vol. 2317. 1995.

64. Kitts, C.A. "Integrated Control of Satellite Payloads", SPIE Proceedings: Platforms and Space Systems, Vol. 2317. 1995.

65. Kelements, H.D. Air Force Satellite Control Network Space and Ground Interface, Aerospace report No.TOR-0059(6110-01)-3. Contract No.F04701-88-C-0089. Satellite Control Network Directorate Satellite Data Division, March 1992.

66. Kolcum, Edward H. " NASA, Pentagon Chart Ambitious Unmanned Launch Vehicle Program," Aviation Week & Space Technology, : 131-133 (March 16, 1992).

67. Kostishack, D.F. "Small-satellite Sensors for Multispectral Space Surveillance", SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

68. Kramer, Stuart. "Application of Concept Mapping To Systems Engineering," IEEE International Conference on Systems, Man, and Cybernetics Conference Proceedings. 652-654. New York: IEEE Press, 1990.

69. Kramer, Stuart. Vector Dynamics. Dayton, OH: Private Printing, 1996.

70. Landsat. "Landsat: What other spacecraft aspire to be... LANDSAT spacecraft and network". http://l7oe01.atsc.allied.com/mainpage.html.

71. Larson,Wiley J. and Wertz, James R. "The Space Mission Analysis and Design Process," Space Mission Analysis and Design. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1992.

72. Larson,Wiley J. and Wertz, James R. Reducing Space Mission Costs. Torrance, California: Microcosm Press, 1996.

73. Lenorovitz, Jeffrey M. " January Pegasus Launch Key to Mission Rate Build-Up," Aviation Week & Space Technology, : 69 (December 7, 1992).

74. Leritz, John E. and Palmer, Donald L. "Mechanics of Materials". Spacecraft Structures and Mechanisms. Ed. Sarafin, Thomas P..Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

75. Lockheed Martin. Lockheed Martin Launch Vehicle (LMLV) Capabilities Summary, 1996.

76. London III, John R. Leo On the Cheap. Air University press, Maxwell AF Base, Alabama, Oct. 1994.

77. Low Cost Gyro Assembly. Guidance and Control for Space Applications. Teterboro, NJ: Allied Signal Aerospace, 1996.

78. Low Cost Momentum Wheel Assembly. Guidance and Control for Space Applications. Teterboro, NJ: Allied Signal Aerospace, 1996.

79. Maquire, Dick. Handbook of Engineering Fundamentals. USA: John Wiley & Sons, 1975.

80. McCall, G., and others. New World Vistas: Air and Space Power for the 21st Century, Summary Volume. Air Force Scientific Advisory Board, Albuquerque, 1995.

81. McDermott, Joseph K. "Power," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

82. McMordie, Robert K. "Thermal," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

83. McMordie, Robert K. "Configuring a Spacecraft," Spacecraft Structures and Mechanisms. Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

84. Mensa, Dean L. High Resolution RADAR Imaging. Norwood, MA: Artech House, 1981.

85. Meyers, Marc A. and Chawla, Krishan K. "Conceptual Design of Structures," Spacecraft Structures and Mechanisms. Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

86. MightySat Program Office. Statement of Work Appendix A Technical Requirements Document (TRD) for the MightySat Phase II Program (Revision 3). Phillips Laboratory, Kirtland AFB, New Mexico. 19-Apr-96.

87. Miniature Inertial Measurement Unit. Honeywell Satellite Systems. Glendale, AZ: Honeywell Space Systems Inc., 1996.

88. "Model Error Led to Pegasus Failure," Aviation Week & Space Technology, : 28 (August 1 1994).

89. Morgan, Walter L.and Gordon, Gary D. Principles of Communication Satellites. New York: John Wiley and Sons, Inc.1993.

90. Mosard, Gil R. "A Generalized Framework and Methodology for Systems Analysis," IEEE Transaction on Engineering Management, EM-29: 81-87 (Aug 1982).

91. Mosier, M.R., and others. "Pegasus Air-launched Space Booster Payload Interfaces and Processing Procedures for Small Optical Payloads," SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

92. Muolo, Michael J. Space Handbook: An Analyst's Guide. Air University press, Maxwell AFB, AL: Air University Press, Dec 1993.

93. Muolo, Michael J. Space Handbook: A Warfighter's Guide to Space (Vol.1). Maxwell AFB, AL: Air University Press, Dec 1993.

94. Myers, Raymond H. and Montgomery, Douglas C. Response Surface Methodology. New York: Jon Wiley & Sons, Inc., 1995.

95. The NASA Home Page, 1996: http:www.nasa.gov.

96. The NASA, ASUSat Homepage, 1996: enws229.eas.asu.edu/asusat/asusat.shtml

97. The NASA/JPL Imaging Radar Home Page. "The NASA/JPL Imaging Radar Home Page". http://southport.jpl.nasa.gov/desc/imagingradarv3.html.

98. The NASA/JPL SAR. "Miscellaneous issues on Synthetic Aperture RADAR (SAR)". http://southport.jpl.nasa.gov/.

99. NASA Launch Task Group. Future of the US Space Launch Capability: A Task Group Report. NASA, Washington, DC, Nov-92.

100. NASA Scientific and Technical Information Office. Technical Memorandum 4679: Space-borne Synthetic Aperture RADAR: Current Status and Future Directions. NASA, Washington, DC, 1995.

101. The NASA Small Explorer Program, 1996: www.laafb.af.mil/SMC.

102.Negron, David and Chomas, Arthur. “Mission Operations,” Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

103.Neter, John and others. Applied Linear Statistical Models. USA: Richard D. Edwin, Inc.,1990.

104.Nguyen, P., and others. Unmanned Space Vehicle Cost Model, 7th ed. Space And Missile Systems Center, Los Angeles, 1994.

105.Nicastri, Edward (Col. USAF Retiered). Former Director of DARPA's ATSSB program. Telephone interview. 5 June 1996.

106.Nicastri, Edward (Col. USAF Retiered). Former Director of DARPA's ATSSB program. Telephone interview. 30 July 1996.

107.Nicastri, Edward (Col. USAF Retiered). “Tactical Maneuvering, Payload Sizing, and Reliability issues”. Electronic mail. 18:43:22 EST, 22 October1996.

108.Nozetto, S. and others. “Clementine Engineering and Technology Workshop,” Proceedings of The Clementine Program’s Engineering and Technology Workshop. Washington DC: Department Of Defense Ballistic Missile Defense Organization, 26 July 1994.

109.O'Neil, J., and others. “Eagle-class Small Satellite for LEO Applications”, SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

110.Orbital Sciences Corporation, 1996: http://www.orbital.com.

111.Orbital Science Corporation. Commercial Pegasus Payload User's Guide. Office of Launch Vehicle, Business Development, Pegasus Launch Vehicle Systems, Dulles, Virginia. 1-Oct-93.

112.Orbital Science Corporation. Commercial Taurus launch Vehicle Payload User's Guide. Office of Launch Vehicle, Business Development, Taurus Launch Vehicle Systems, Dulles, Virginia.18-Apr-96.

113." Pegasus Carrier Aircraft Completes Flight Tests," Aviation Week & Space Technology, : 57-58 (December 6, 1993).

114.Pegasus Marketing Video. Prod. Orbital Science Corporation. Dulles, Virginia. 1995.

115.Peery, David J. and Azar, J. J. Mechanical Metallurgy. Englewood Cliffs, NJ: Prentice-Hall, Inc., 1984.

116.Pohl, Ed. Class handout, SENG 520.School of Engineering, Air Force Institute of Technology, WPAFB OH. 6-Oct-95.

117.Presley, S.P., Agardy, F.J., and Berrier, D.J. An Appoach To Rapid Payload Integration: The Spacecraft-To-Payload Interface Guideline (SPIG). The Aerospace Corporation, El Segundo, CA. 8-Apr-96.

118.Rademacher, Joel. "NASA Small Explorer (SMEX) Home page: NASA Small Explorer Program," 1996: sunland.gsfc.nasa.gov/smex/smexhomepage.html.

119.Rademacher, Joel. "Phillips Laboratory," 1996: enws229.eas.asu.edu/joel/joel.shtml

120.Ramakumar, R. Engineering Reliability - Fundamentals & Applications. Englewood Cliffs, CA: Prentice Hall, 1993.

121.Rash, James L. Third International Symposium on Space Missions and Ground Data Systems Part I/II. Greenbelt, Maryland: NASA, Nov 15-18, 1994.

122.Reaction Wheel and Momentum Wheel Assemblies. Space Systems - Momentum Control. Glendale, AZ: Honeywell Space Systems Inc., 1996.

123.Rees, W.G. Physical Principles of Remote Sensing. Cambridge: Cambridge Univ. Press, 1990.

124.Reeves, Emery "Spacecraft Design and Sizing," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

125.Reeves, Emery "Spacecraft Manufacture and Test," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

126.Reichhardt, Tony. " Launch Woes Ground NASA Science Spacecraft," Naure, 45.717 (August 31, 1995).

127.Reinert, Richard P. and Wertz James R. "Mission Characterization," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R.. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

128.Reynolds, William C. and Perkins, Henry C. Tactical Support Satellite/Standard Payload Interface (TSS/SPI) Study: Standard Interface Definitions and Cost Comparisons. The Aerospace Corporation, El Segundo, California, Sep-94.

129.Sackheim, Robert L. "Space Propulsion System," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

130.Sarafin, Thomas P., Heymans, Robert J.Wendt, Robert G. Jr., and Sabin, Robert V. "Thermal Effects," Spacecraft Structures and Mechanisms. Ed. Sarafin, Thomas P..Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

131.Sarafin, Thomas P. AIAA Aerospace Design Engineers Guide. American Institute of Aeronautics and Astronautics, Inc., Washington DC, 1993.

132.Scott, Pace. US Access to Space Launch Vehicle Choices for 1990-2010. RAND Santa Monica, CA, Mar-90.

133.Sellers, Jerry, and Milton, Ed. "Technology for Reduced Cost Missions," Reducing Space Mission Cost. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1996.

134.Servo Space Systems, 1996: http://www.servo.com/spasys.html.

135.Shuster, Malcolm D. Fundamentals of Space Systems. New York : Oxford University Press, 1994.

136.Skullney, W. E. "Spacecraft Thermal Control". Fundamentals of Space Systems. Ed. Pisacane, Vincent L. and Moore, Robert C. New York: Oxford University Press, 1994.

137.Smith, Bruce A. " Orbital L-1011 Ready for First Launch," Aviation Week & Space Technology, : 30 (July 4, 1994).

138.Smith, Bruce A. " OSC Seeks Cause of Pegasus XL Failure," Aviation Week & Space Technology, : (1994).

139.Smith, Bruce A. " Pegasus XL Success Opens Door to Fast-Paced Launch Schedule," Aviation Week & Space Technology, : 26-27 (March 18, 1996).

140.Space & Missile Systems Center, Orbital Sciences Corporation's (OSC's) Home Page, 1996: www.orbital.com

141.Space Systems Division. Pegasus Design Safety Requirements Document. Orbital Science Corporation, Fairfax, Virginia, 27-May-92.

142.Stanford SSDL Home Page. "Stanford SSDL Home Page. Space System Design at Stanford University. http://aa.stanford.edu/~ssdl/.

143.Stodden, David Y. and Galasso, Gina D. Engineering Theromodynamics. USA: McGraw-Hill Book Company, 1977.

144.Sutton, George P. Rocket Propulsion Elements. New York: Jonh Wiley and Sons, 1992.

145.Taurus Marketing Video. Prod. Orbital Science Corporation. Dulles, Virginia. 1996.

146.Telemetry Components and Systems, Command, Control and Communications Catalog. Newtown, PA. AYDIN VECTOR, 1996.

147.The Aerospace Corporation. Communications Satellites 1958-1992. El Segundo, California: The Aerospace Corporation, 1991.

148.U.S. Air Force Home Page, 1996: http://www.af.mil.

149.U.S. Congress, Office of Tecnology Assesment. Affordabble Spacecraft Design and Launch Alternatives. Washington, DC: US printing Office. (OTA-BP-ISC-60)

150.Vesely, David. "Transcript : DOD News Briefing : Major General David Vesely, 14th Air Force Commander. Miscellaneous in-theater space operations concepts". http://www.laafb.af.mil/SMC/CZ/homepage/news/dodnew13.htm.

151.Velocci, Anthony L. " Orbital Says Human Error Caused Pegasus XL Loss," Aviation Week & Space Technology, : 60 (September 4, 1995).

152.Walker, Kevin J. Systems Analysis of GPS Electrical Power System Redesign. MS thesis, AFIT/ENY/AFIT/GSO/ENY/95D-03. School of Engineering, Air Force Institute of Technology, WPAFB OH, Dec-95.

153.Warner, Richard. Founder and Chief Technical Officer of AeroAstro Corp. Personal interview. 18 July 1996.

154. Wertz, James R. "Mission Evaluation," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R.. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

155. Wertz, James R. "Space Mission Geometry," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

156. Wertz, James R. "Orbit and Constellation Design," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

157. Wiesel, William E. Spaceflight Dynamics. New York: McGraw-Hill, 1989.

158. Williams, Michael S. "Requirements Definition," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

159. Wingate, Clarence. SatAC Materials Database (matter.dat). Phillips Laboratory, Kirtland AFB, New Mexico, Mar-96.

160. Wong, Robert "Cost Modeling," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

161. Worden, Col. Simon P.,COL, USAF (50SW/CC)."Space: Grabbing the Solar System (and Dominating the Planet)," AF 2025 Submission. Falcon AFB, CO: «Publisher», 1996.

Vita

Capt Gerald (Gerry) F. Ashby [redacted]. After graduating from Riverside High School in June 1983, he entered the enlisted ranks of the U.S. Air Force. A year later he entered the U.S.A.F Academy Preparatory School in Colorado Springs, Colorado. Upon graduating with honors, he entered the U.S. Air Force Academy and joined the class of 1989. Four years later on 31 May 1989, graduating in the top 15% of his class, he received his degree in Engineering Sciences of Aerospace Structures and his regular commission in U.S. Air Force.

Upon graduation, he left for pilot training at Williams AFB, Arizona and after unsuccessfully completing the program, he entered Undergraduate Space Training at Lowry AFB in August 1990. His next assignment was to Falcon AFB as a Launch Analyst for the Navy's UHF F/O communication satellite program. After leaving Falcon AFB in December 1993, he transferred to Detachment 1, AFSPC at Los Angeles AFB, California, where he worked as the main Air Force Space Command spokesman within the Milsatcom Joint Program Office. Then in May 1995, he entered the School of Engineering, Air Force Institute of Technology.

[redacted]

Captain Darren J. Buck [redacted]. He graduated from Holy Cross High School in Dover, Delaware May 1986 and began the Freshman Year Studies at the University of Notre Dame in August 1986. He graduated from Notre Dame with a Bachelor of Science degree in Mathematics in May 1990 . He received his commission on 19 May 1990 after having completed the Notre Dame Air Force Reserve Officer Training Corps (NDAFROTC).

After completing Undergraduate Space Training at Lowry AFB, CO in August 1991, his first assignment was as a Satellite Operations Crew Commander at the 3rd Satellite Control Squadron (3 SCS -- later the 3rd Space Operations Squadron (3 SOPS) at Falcon AFB, CO. His two subsequent assignments, also at Falcon, were as a Simulation Software Development/QAE Officer and Satellite Systems Instructor for the MILSTAR Satellite Program at the 50th Crew Training Squadron (50 CTS), and as Group Operations Training Officer for MILSTAR at the 50th Operations Support Squadron (50 OSS). In May 1995, he entered the Graduate Space Operations Program in the School of Engineering at the Air Force Institute of Technology.

[redacted]

Lieutenant Robert Carneal IV [redacted]

He graduated from two high schools in 1988 located in Virginia: Thomas Jefferson School for Science and Technology, and Fairfax High School. In the Fall of 1988 he entered the Prescott, AZ campus of Embry-Riddle Aeronautical University. He graduated with honors with a bachelor's science degree in electrical engineering, with a math minor, in December 1992. Upon graduation he was commissioned second lieutenant in the USAF.

His first assignment was the 5th Space Launch Squadron at Cape Canaveral Air Station, where he was involved in launching Titan IV rockets. In May 1995 he entered the Air Force Institute of Technology at Wright-Patterson as a masters candidate in space operations. His follow on assignment is with the Space Warfare Center at Falcon AFB.

[redacted]

Lieutenant Tansel Cokuysal [redacted] He graduated from Maltepe Military High School in 1987 and entered undergraduate studies at the Turkish Air Force Academy in Istanbul, Turkey. He graduated with a Military of Science degree in Aerospace Engineering August 1991. Upon graduation, he was commissioned a Second Lieutenant in Turkish Air Force.

His first assignment was at 2nd Main Jet Base, Izmir Turkey for Basic Jet Pilot Training. His second assignment was to 3rd Main Jet Base, Konya Turkey for F-5A/B Combat Readiness Training. His third assignment was to 4th Main Jet Base, Ankara Turkey for F-16 Basic and Combat Readiness Training. His subsequent assignment was to 9th Main Jet Base, Balikesir Turkey as a Wing-man in 191st (Cobra) Squadron. In May 1995, he entered the School of Engineering, Air Force Institute of Technology.

[redacted]

Lieutenant Ahmet Tuna Donmez [redacted]

He graduated from Maltepe Military High School, Izmir in 1987 and entered undergraduate studies at the Turkish Air Force Academy in Istanbul, Turkey. He graduated with a Military of Science degree in Aerospace Engineering August 1991. Upon graduation, he was commissioned as a Second Lieutenant in Turkish Air Force.

His first assignment was at 2nd Main Jet Base, Izmir Turkey for Basic Jet Pilot Training. His second assignment was to 3rd Main Jet Base, Konya Turkey for F-5A/B Combat Readiness Training. His third assignment was to 1st Main Jet Base, Eskisehir Turkey for F-4E and RF-4E Basic and Combat Readiness Training. His subsequent assignment was to 1st Main Jet Base, Eskisehir Turkey as a Wing-man in 113th Tactical Reconnaissance Squadron. In May 1995, he entered the School of Engineering, Air Force Institute of Technology to have his Maters of Science degree in System Engineering.

[redacted]

Capt James A. From [redacted]. He graduated from Frank W. Cox High School in 1981 and entered undergraduate studies at the University of Virginia in Charlottesville, Virginia in 1982. He graduated with a Bachelor of Science degree in Aerospace Engineering in May 1987. He received his commission on 16 May 1987 and was a Distinguished Graduate from Reserve Officer Training Corps.

His first assignment was at Lowry AFB for Undergraduate Space Training. His second assignment was to Peterson AFB as a Defense Satellite Communications System phase III (DSCS III) Commander and Planner/Analyst Instructor. His subsequent assignment was to Buckley ANGB as a Satellite Operations Crew Commander and later Chief of Standardization and Evaluation for the Defense Support Program. While at Buckley ANGB, he earned a Masters of Arts degree in Space Systems Management from Webster University. In May 1995, he entered the School of Engineering, Air Force Institute of Technology. His follow on assignment is to United States Space Command Headquarters at Peterson AFB.

[redacted]

Capt Todd C. Krueger [redacted]. He graduated from Sprague High School in 1987 and entered undergraduate studies at the University of Southern California in Los Angeles, California. He graduated with a Bachelor of Science degree in Aerospace Engineering in May 1991. He received his commission on 9 May 1991, having completed the Air Force Reserve Officer Training Corps (AFROTC) program.

His first assignment was at Falcon AFB as a satellite analyst/instructor. In May 1995, he entered the School of Engineering, Air Force Institute of Technology.

[redacted]

First Lieutenant Brian I. Robinson [redacted]

He graduated from Whitney M. Young High School in 1987, and earned a Bachelor of Science degree in Aerospace Engineering with honors, and an Air Force commission (Distinguished Graduate) from Tuskegee University in May 1993.

His only assignment prior to AFIT was as an engineer for the Titan System Program Office (now the Launch Programs System Program Office), Los Angeles Air Force Base, CA, from September 1993 to May 1995. His duties included managing the ongoing development, production, acquisition, and delivery of the Titan IV's Solid Rocket Motors (SRM), and was present at Cape Canaveral AS for five Titan IV launches, serving as the Titan SPO's system expert on the Titan SRMs.

Lt Robinson is married to Lt Coleen Y. Robinson, USAF.

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Current satellite design philosophies concentrate on optimizing and tailoring a particular satellite bus to a specific payload or mission. Today's satellites take a long time to build, checkout, and launch. An alternate approach shifts the design paradigm to one that focuses on access to space, enabling tactical deployment on demand and the capability to put current payload technology into orbit, versus several years by today's standards, by which time the technology is already obsolete. This design study applied systems engineering methods to create a satellite bus architecture that can accommodate a range of remote sensing mission modules. System-level and subsystem-level tradeoffs provided standard components and satellite structures, and an iterative design approach provided candidate designs constructed with those components. A cost and reliability trade study provided initial estimates for satellite performance. Modeling and analysis based upon the Sponsor's objectives converged the designs to an optimum solution. Major products of this study include not only a preliminary satellite design to meet the sponsor's needs, but also a software modeling and analysis tool for satellite design, integration, and test. Finally, the report provides an initial implementation scheme and concept for operations for the tactical support of this satellite system.

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