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BENEFITS OF A SPACE-BASED GROUP SYSTEM ARCHITECTURE

by

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June 2015

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The volatility in today's economics has resulted in government attempts to reduce cost while maintaining performance. One of the elements examined by the Defense Advanced Research Projects Agency was the idea of fractionating a current monolithic satellite system into several smaller, space-based group (SBG) satellites. This architecture would allow for multiple, smaller, and less expensive satellites to work together to accomplish the several missions.

This study focused on research and analysis of the system FireSat. The analysis removed the ground communications suite from the sensor platform. A Microsoft Excel spreadsheet was used to develop the resulting cost relation for the sensor-only satellite. Using assumptions provided by that analysis, three additional systems, currently in operation, were examined for cost savings if placed into the SBG. The Tracking and Data Relay Satellite was used as a basis for cost of a communications satellite. The cost analysis resulted in an estimated $52 million FY15 to the space segments alone. Additional research is required to determine cost savings within the full architecture and develop a risk-cost analysis to determine whether cost could be further reduced due to higher reliability, lower replacement cost risk, and longer lifetimes.

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BENEFITS OF A SPACE-BASED GROUP SYSTEM ARCHITECTURE

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LIST OF ACRONYMS AND ABBREVIATIONS

COCOM	Combatant Commander
CONUS	Continental United States
DARPA	Defense Advanced Research Projects Agency
DOD	Department of Defense
F6	Future, Fast, Flexible, Fractionated Free-flying Spacecraft United by Information Exchange System
FGST	Fermi Gamma-ray Space Telescope
FY	Fiscal Year
GALEX	Galaxy Evolution Explorer
GEO	Geostationary Earth Orbit
ISS	International Space Station
LAN	Local Area Network
LEO	Low Earth Orbit
MANASSAS	Mission-Aware Network Architecture for Small-Satellite Adaptive Systems
MUOS	Mobile User Objective Systems
MWNE	Mir Wireless Network Experiment
NASA	National Aeronautics and Space Administration
NAVAIR	Naval Aviation
NOAA	National Oceanic and Atmospheric Administration
RAAN	Right Ascension of the Ascending Node
R&D	Research and Development
SATCOM	Satellite Communications
SBG	Space-based Group
SMAD	Space Mission Analysis and Design
SSCM	Small Satellite Cost Management
STK	System Tool Kit
STS	Space Transportation System
SWIFT	Swift Gamma Ray Burst Explorer

TDRS	Tracking and Data Relay Satellite
TRL	Technology Readiness Level
TT&C	Telemetry, Tracking, and Control
UFO	UHF Follow-On

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I. INTRODUCTION

Since the launch of Sputnik 1 in 1957, the nations of the world have researched and designed spacecraft to perform a variety of services while in orbit. These missions range from weather monitoring to worldwide communications. Technology has advanced rapidly in the past half-century, and this has led to increased requirements placed upon orbital systems. These requirements have resulted in a transition from a singular mission based system into a platform capable of facets of different missions, all executed concurrently. These coupled requirements have given rise to an increase in complexities, increasing the need to reduce failures, resulting in increased costs. The continued focus on a requirement-centric at minimum-cost paradigm has endured due to the U.S. military, the "aerospace industry's star client" according to Brown and Eremenko (2006b, p. 3)

The focus of military application of space systems has maintained a requirement based acquisition architecture to fulfill Combatant Commander (COCOM) and national strategic desires in the field. The continuous battle has been the cost of these systems, and specifically the best way to reduce the lifetime costs while maintaining the desired system performance. The draw between utilizing several smaller satellite constellations over reduced large, singularly operated satellites continues to be examined by acquisition agencies.

The Defense Advanced Research Projects Agency (DARPA) was issued a contract to develop a project it termed Future, Fast, Flexible, Fractionated Free-flying Spacecraft United by Information Exchange System (F6). This system was developed around the basis of fractionated satellites to form a space-based group (SBG) system capable replacing the current model of monolithic spacecraft. This development will change the concepts of design and development of military satellite systems. The ability to relocate, remove, and replace a single system in a cluster is one of the many advantages that will improve overall military operations.

This thesis compares the architecture of a SBG to that of a system with a similar capability that would be representative of a COCOM requirement. It focuses on improvements in real-time intelligence and communications; overall cost of a system and the incorporation of this architecture into the current military satellite construct.

A. BACKGROUND

The U.S. military has exploited the freedom of space use for many decades. The systems placed in orbit have evolved from those providing basic geolocation to global communication to early warning and threat assessment. Each system has been singularly developed, focused on meeting national strategic or COCOM requirements. These requests for capability requirements are routed through several channels and may often result in an ultimate need to develop a new program. The new acquisition program typically utilizes current proven technology and a reasonable estimate of maturity of future technology. Current acquisition timelines have forced focused system development toward a specific payload to meet the requirements, provided by a primary sensor. As technology has advanced, the cost of implementation of a single sensor has increased as well. This increased cost has driven additions to the lifetime requirements, as a desire to ensure reliability and maintainability of a system in order to fully capture the cost return.

The DARPA System F6 was established in 2006 as a program to develop an architecture in which the functions of a monolithic system could be divided into a cluster of wirelessly-interconnected smaller satellites, sharing resources to achieve the same functions ("System F6," n.d., para 1). This architecture would display the capability, cost analysis, and viability of the SBG for military utilization.

Recently, the System F6 program has been canceled due to a number of factors, to include a lack of an overall integrator for the experiment. The program had awarded several small contracts to companies for work distribution; however the lead integrator role was not identified (Ferster, 2013). This recent cancelation illustrates a key performance requirement of all systems under development, the requirement to integrate all components for common use.

B. PURPOSE AND RESEARCH QUESTION

The increased uncertainty in today's economy has led to a minimum-cost acquisition process adopted by the Department of Defense (DOD). This process, combined with the military requirements-centric paradigm, have led architects to conclude that the answer is greater capability and increased system lifetime (Saleh, 2006). These increases have led to multiple requirements being placed upon a system, increasing costs of monolithic systems.

This thesis will examine the benefits of replacing a monolithic satellite system with clustered SBG systems, meeting requirements while reducing subsystems and complex integration. The paper will utilize the FireSat imaging satellite design provided by *Space Mission Analysis and Design* (SMAD) (Apgar, Bearden, Bell, Berget, Blake, Boden, et al., 1999), and model a SBG design, separating the sensor payload from the communications suite. The analysis will provide an overall comparison of space segment cost of the systems, as well as the benefits and drawbacks of the systems. A final analysis will expand the comparison to include other current on orbit systems that could operate in conjunction with the SBG. This comparison will attempt to answer the questions:

- Can the SBG architecture reduce cost of space systems?
- Could a SBG allow mission flexibility without substantial cost increases?

C. BENEFITS OF STUDY

Cost reduction is a highly desired objective in today's DOD acquisition process. Providing equal capabilities, while reducing cost and improving adaptability, will be a focus for the long term future of space system utilization. This thesis will examine the utilization of SBG architecture within the DOD Space Systems acquisition program, determining the benefits of this architecture, and the cost comparison to that of current monolithic architectures.

D. SCOPE AND METHODOLOGY

This thesis will focus on a military application of SBG system architecture. The example system, FireSat, architecture utilizes an imaging platform for forest fire identification and monitoring. The cost analysis focuses on system development, of the system, with the primary assumption that the technology utilized was researched and developed outside the architecture of the system. An additional assumption is that deployment of the systems will be accomplished by current proven means. All cost modeling and estimates were done utilizing Fiscal Year (FY) 2015 dollars, based on the inflation factors chart, table 20-1, provided in SMAD (Apgar et al., 1999),

II. SPACE-BASED GROUP COMPARISON

A. INTRODUCTION

The military has become heavily reliant on satellite systems for daily strategic, operational, and tactical decisions. The capabilities in place have become requirements, and the systems providing these capabilities cannot maintain the growing demand being placed on them. Additionally, the rate at which technology matures in today's economy, at times, results in an obsolescent system prior to its initial operational employment. Current acquisition models typically result in monolithic architectures to develop and deploy new systems to meet the growing demand, and anticipate future demand. The acquisition architecture utilized by the DOD requires several milestone periods, which could take years for transitions between milestones. Figure 1 illustrates the acquisition management the DOD utilizes, and illustrates how the technological development of a system is often frozen after a milestone "B" decision. Future developments have to proceed through the same process. This acquisition construct does not allow for rapid change to a system's technologic architecture due to obsolescent parts.

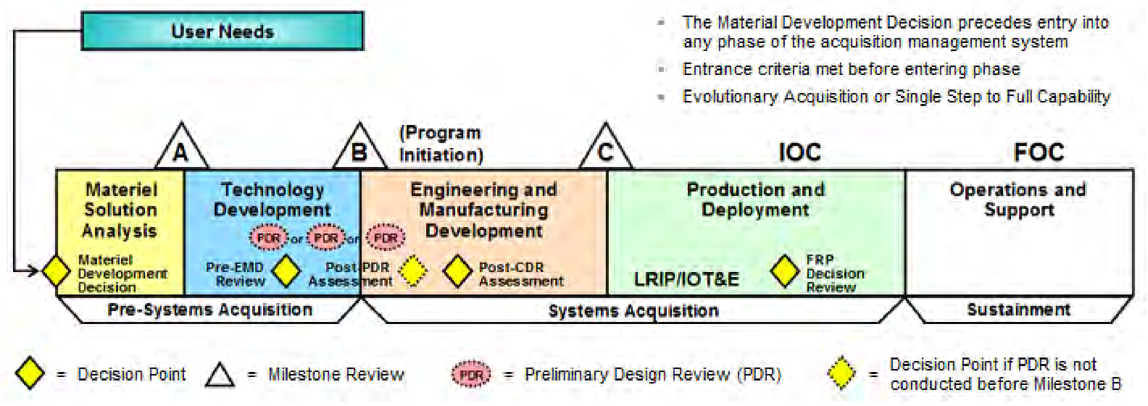

Figure 1. Basic System Acquisition Framework (from Defense Acquisition Portal, 2013)

The milestone reviews indicated also allow for program leads to make a determination to move toward deployment and operation. These reviews require both technological maturity of the system as well as cost performance and estimates to allow continuation of the program.

B. MONOLITHIC ARCHITECTURE

The structure of monolithic satellite system architectures follow typical acquisition models utilized by private and public businesses alike. This section outlines this structure from concept to operations, describing some of the characteristics and limitations recognized during a satellite system development.

1. Concept Inception

A capability requirement is produced by national security strategies, COCOMs, or other entities responsible for safe and successful mission completion. These requests for capabilities are what drive the initial concept of the satellite system. The system concept is refined through multiple requirement characterizations, with a final performance requirement being defined prior to milestone “A”. Once the final performance requirement is set, an associated payload sensor that can accomplish the mission set would be researched and chosen. The associated payload is scrutinized to determine the required subsystems needed to fulfil the payload requirements, and an estimated lifetime cost is created. The cost of these systems have often increased steadily over time, leading architects and managers to come to an inaccurate conclusion that greater capability is required to offset the cost growth (Brown & Eremenko, 2006b). This results in additional payload requirements and complexities when moving forward into the design and development of the system.

2. Design and Development

When the primary payload has been defined, the design and development of the monolithic system begins. The design phase of a space borne system is a major focus of program management, in that the desire is to provide the best subcomponents for the system, while still maintaining a low cost and on time delivery. Achieving a reliable, cost effective system becomes the focus of managers, while still remaining under the pressures of program and technical uncertainty. These goals often lead to a complex spacecraft with multi-mission set requirements, driving costs higher and higher. The responses to cost increases have been to maximize the capability versus cost quotient by increasing the spacecraft scale and increasing the system lifetime (Saleh 2008). This response is does not always allow for appreciated cost savings.

a. Multi-mission Design

The ability to fuse a system to provide a range of payloads to satisfy several user requirements is an approach that a program management can utilize. This multi-mission capability allows for a more secure program during development. However, there are several limitations to a multi-mission design, and these limitations can increase as the program matures. First, a mission requirement change in a single portion of the system during development may require a larger system design change. This design change, in turn, results in increased cost and delayed scheduling, and can jeopardize the program. A second risk to a multi-mission program is the requirements of each payload or subsystem, and negative interactions between them (Owen & Eremenko, 2006a).

> Orndorff and Zink (2010) illustrated this complexity in an example of a payload that requires precision pointing and low jitter, but produces a large amount of valuable data. Getting the data to the ground requires high power, which drives the solar arrays to grow, which in turn disturbs the pointing and jitter control capability of the space vehicle. (p. 275)

For a multi-mission system, independent development of smaller satellites may provide better cost and risk reduction to a program, as compared to the current paradigm used in typical architectures.

b. Technological Limitations

A satellite system requires tested technology for employment into the space environment. The DOD and National Aeronautics and Space Administration (NASA) utilize Technology Readiness Levels (TRLs) to systematically measure system and subsystem maturity for utilization. The TRLs range from one to nine, from least to most mature systems respectively. The example of NASA's TRLs is illustrated in Table 1.

These TRLs have huge impacts on the design and development life of a space borne system. During design, the payload and subsystems are set based on current mature technology, or technology that is expected to reach maturity prior to employment of the system. These restrictions require utilization of proven, less advanced technology for each sensor and subsystem in a monolithic satellite. In addition, there comes a point during the acquisition phase in which a technology freeze must be implemented. This forces the development of the current system with no additional changes or upgrades to the technology placed within the satellite. This requirement ensures the program stays on schedule and reduces programmatic cost that would result from integration and testing of newly developed technology. Fixing the design restricts the overall future capability of the system in development, freezing the system maturity to a point in time prior to the system employment. In order for a capability to be achieved utilizing advanced technology, a new system must be developed. The result is a program which continuously requires redevelopments to employ newer capabilities to replace the current system, or acquiescing on a less advanced technological system to be utilized for the original lifetime of the system. The later response is often chosen due to cost constraints and project uncertainties.

Table 1. NASA Technology Readiness Level Chart (from Mankins, 1995)

Technology Readiness Level	Description
1. Basic principles observed and reported	This is the lowest "level" of technology maturation. At this level, scientific research begins to be translated into applied research and development.
2. Technology concept and/or application formulated	Once basic physical principles are observed, then at the next level of maturation, practical applications of those characteristics can be 'invented' or identified. At this level, the application is still speculative: there is not experimental proof or detailed analysis to support the conjecture.
3. Analytical and experimental critical function and/or characteristic proof of concept	At this step in the maturation process, active research and development (R&D) is initiated. This must include both analytical studies to set the technology into an appropriate context and laboratory-based studies to physically validate that the analytical predictions are correct. These studies and experiments should constitute "proof-of-concept" validation of the applications/concepts formulated at TRL 2.
4. Component and/or breadboard validation in laboratory environment	Following successful "proof-of-concept" work, basic technological elements must be integrated to establish that the "pieces" will work together to achieve concept-enabling levels of performance for a component and/or breadboard. This validation must be devised to support the concept that was formulated earlier, and should also be consistent with the requirements of potential system applications. The validation is "low-fidelity" compared to the eventual system: it could be composed of ad hoc discrete components in a laboratory.
5. Component and/or breadboard validation in relevant environment	At this level, the fidelity of the component and/or breadboard being tested has to increase significantly. The basic technological elements must be integrated with reasonably realistic supporting elements so that the total applications (component-level, sub-system level or system-level) can be tested in a "simulated" or somewhat realistic environment.
6. System/subsystem model or prototype demonstration in a relevant environment (ground or space)	A major step in the level of fidelity of the technology demonstration follows the completion of TRL 5. At TRL 6, a representative model or prototype system or system - which would go well beyond ad hoc, "patch-cord" or discrete component level breadboarding—would be tested in a relevant environment. At this level, if the only "relevant environment" is the environment of space, and then the model/prototype must be demonstrated in space.

Technology Readiness Level	**Description**
7. System prototype demonstration in a space environment	TRL 7 is a significant step beyond TRL 6, requiring an actual system prototype demonstration in a space environment. The prototype should be near or at the scale of the planned operational system and the demonstration must take place in space.
8. Actual system completed and "flight qualified" through test and demonstration (ground or space)	In almost all cases, this level is the end of true "system development" for most technology elements. This might include integration of new technology into an existing system.
9. Actual system "flight proven" through successful mission operations	In almost all cases, the end of last "bug fixing" aspects of true "system development." This might include integration of new technology into an existing system. This TRL does *not* include planned product improvement of ongoing or reusable systems.

c. Reliability Development

Once a system has been designed and development is underway, a focused desire is to incorporate reliability and fragility for reduced mission degradation or failures as a result of any system damage. Fragility, as defined by Brown and Eremenko (2006a) "is the tendency of complex systems to exhibit unmodeled failure modes, usually due to an unanticipated component interaction leading to a catastrophic, albeit improbable, sequence of events" (p. 2). Reliability is the tendency of the system to be trusted to perform as designed, and to provide the data as required by the customer. The developmental requirements to attain reliability and flexibility necessitate continued examination of essential parts of the system during development. Every subsystem required to provide support to the payload must be scrutinized, ensuring that margins or additional redundant components are incorporated in order to provide a reliable system (Brown & Eremenko, 2006a).

The additions to a system to ensure reliability are not enough to protect against failure of employment and operation of the spacecraft. Several factors can occur that would result in a loss in the capability that the system is due to provide, such as launch vehicle failure, damage during launch, in-space collision, etc. These additional failures require contingencies in order to ensure capabilities are not lost. Commercial companies require additional insurance to cover these costs, such as 10 percent to 20 percent of the payload replacement cost for launch failure, and an annual two percent to five percent on-orbit insurance ("Commercial space," 2002). For the DOD, the missions often require an expensive replacement in the event of a failure during employment, even though they are technically self-insured (Brown & Eremenko, 2006b).

3. Test and Employment

Once a system has completed milestone B, it proceeds to test and evaluation, while continued development of the system is completed. The parallel test and development allows for contractors to improve systems prior to full implementation and integration.

a. Payload Integration

Great design and detail are required for payload integration of a monolithic satellite system. A singular satellite must ensure common communication between the payload, the power subsystems, the data storage devices, and the communication suite. This focused integration within a single system results in a single contractor, responsible for the development of all the systems, ensuring proper integration. Combine this with a multiple mission system, and the coupled interfaces increase the development risk, which in turn requires combinatorial complexity during testing (Orndorff & Zink, 2010). Integration is a discipline that requires complex interaction between the monolithic system itself, as well as the ground segments which receive from and transmit to the system.

b. Demand Requirements

During payload integration and system development, the required performance demand for a system must be determined. This determination must be made for a future date, due to increases in development, manufacturing, and launch windows (Brown & Eremenko, 2006a). The system must be able to meet these demands, which often involve the quality, availability, and repeatability of the products provided. Added into this demand is an estimated increase in customers once operational. An ideal example of added customers is that of military satellite communications (SATCOM). Developed initially for naval vessels at sea, the requirement has spread to all military units; at sea, ashore, desert, or jungle. The result has been the development of the Mobile User Objective System (MUOS) to replace the current UHF follow-on SATCOM system. The MUOS will increase SATCOM accessibility in theater (Bandil, Pandya, & Shields, 2010).

A monolithic system cost can increase exponentially when attempting to meet all of the requirements that would expanded its capability. Frequently, trade-offs must occur to reach cost goals while sacrificing requirements. For example, program developers will trade the amount of data a payload can provide with the bandwidth used to transmit that data down to a ground station (Orndorff & Zink, 2010). This type of trade-off results in a system that does not fully meet the intent of the request for capability; and is an inherent problem with a single, monolithic system.

c. On-Orbit Operations

Once a monolithic system is placed into orbit, the product provided is highly scrutinized by the customer that receives it. Once fully operational, the quality of data provided will also affect the demand placed on the system, and the combined result of the system to meet the demand and the product will feed into the cost allotted for maintaining and operating the system. This cost could result in continued operation past the initial date, or could cause the DOD to remove funding and kill the project before the intended lifetime has been reached. The volatility of military funding and national economics require a quick and accurate assessment of a system's vitality to the overall mission it has been developed to support. The operation and maintenance of a space system, conversely to that of other DOD acquisition programs, covers only 30 percent of the total lifetime cost of the program, as displayed in Figure 2. If a monolithic satellite has been placed that will not meet the requirements set forth during development, the system will be discarded and a new development process will begin, resulting in an exuberant cost increase to the program.

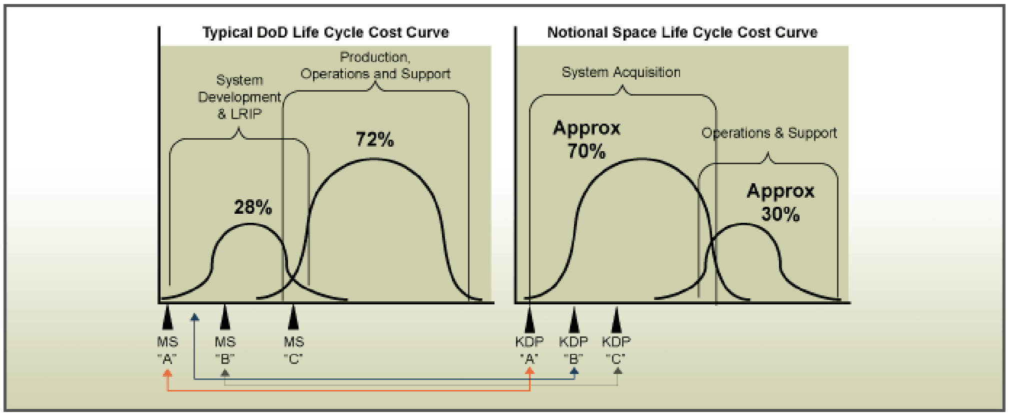

Figure 2. DOD versus Space Life Cycle Cost Curve (from Department of Defense Space Acquisition, 2014)

When a monolithic system provides quality information, the forced demand on that system will increase. This will often lead to delayed products and inaccurate assessments of poor performance by the customers. A monolithic system can only provide to the expectations of demand that were designed into the system years prior to employment. With the advancement of technology, these capabilities are often times enormously low as compared to the mid-life demands placed on the system. The only response the DOD has to correct for increase demand is to supplement the current systems with additional support prior to scheduled replacement with more advanced systems. An example of this addition was adding UHF Follow-On (UFO) satellite F11, as a gap filler between the UFO communications system and MUOS, which is a next generation narrowband communication system (Ghyzel, 2010).

C. UTILIZATION OF SPACE BASED GROUPS

The SBG architecture grants several cost advantages to the current monolithic architecture utilized today. The flexibility to contract out pieces of a system during design and development can provide cost savings to the DOD, due to competitive pricing on individual items, vice a single satellite. Once deployed, SBG systems will reduce operation cost, increase the reliability and maintainability of the system, and provide quick and relatively cheaper replacement options for customers and contractors alike. The savings is focused on how many satellites can perform dissimilar functions within the system (Brown & Eremenko, 2006a). Working from this concept, system design and development can proceed, with the payloads of each system proceeding parallel to the others.

1. Development

The development of a SBG system can save schedule cost, contractor cost, and allow for future expansion of the system. The ability to award contracts for individual modules, developed somewhat independently of each other, allows for competition and cost reduction. Each contractor can develop, in parallel, a smaller system that will meet the requirements provided, and the subsystems required to support that payload. Additionally, since the systems are independent, they do not need to be functionally identical (Orndorff & Zink, 2010). The additional integration with the communication hub satellite is the only common portion needing development. This requirement is the foundation of what binds the acquisition schedule of the system. While delays in the communications satellite could result in schedule slippage, the fractionated architecture of other components could allow for a steady schedule despite minor delays, allowing required milestones to be met on time.

As stated previously, the uncertainty during development can often lead to unanticipated expenses and schedule slippage. When the SBG architecture is used for development, additional options are provided that allow for subsequent modifications in the event of unanticipated changes. These options provide real value that can be computed into the architecture itself (De Weck et al., 2004).

2. Overall Integration

Integration is the key to any system under development in today's military. Integration between systems in a platform, as well as integration between platforms has become a continued focus within military leadership.

> Vice Adm. Dunaway, the Naval Aviation (NAVAIR) Commander (2013), has stated that NAVAIR is committed to develop the processes, skills, and tools to successfully execute mission-level systems-of-systems engineering, test and evaluation, and logistics—ensuring all of these important elements are as tightly integrated as the capabilities we deliver to the Fleet. (p. 4)

With a SBG structure, the development and fabrication of each module could be divided among multiple contractors, but this requires a standardized inter-module interface (Brown, 2004). The DARPA F6 project was cancelled by Brad Tousley for several reasons, including no overall integrator for the experiment (Ferster, 2013). The integration of any system is the key element to the program success, and for SBG, a central communication hub satellite that provides the routing and control services is required to meet this integration necessity (Orndorff & Zink, 2010).

The SBG architecture focuses on utilizing the communication hub satellite with additional smaller satellites that provide the required capability each was designed for to the desired customers. The entire cluster of satellites would utilize a wireless Internet Protocol local area network (LAN) to relay information throughout the group. Experiments utilizing wireless networks have been used to confirm the capability onboard the Mir and International Space Station (Orndorff & Zink, 2010). The success of these experiments, in both ensuring data exchange, even at low transmitter power, shows that utilization of wireless networks in space is feasible (Lofton, & Conley, 1997). The wireless network will allow for each system to provide the required data to the hub satellite. This data exchange needs to be robust while allowing for a resistance to interferences that are expected to be experienced (Sahel, 2006). Each satellite's data must also be capable of data transmission through the communication satellite to a ground station, or stored in a data storage device onboard the hub satellite.

Data storage and transmission is the major integration problem in a SBG design. The hub satellite must be capable of receiving, and possibly converting, payload data required by the customers. This data must also be capable of being compressed, and transmitted concurrently with other data from other mission satellites. The integration should be granted to the hub satellite contract, ensuring that the central network satellite can perform the mission of integrating all the data into a usable and transmittable form.

3. On-Orbit Maintainability

A separate issue that can arise for any system in orbit is the requirement to fix or replace the system due to mechanical malfunction or damage due to launch or space-borne hazards. The standard procedure for a malfunctioning system is to either utilize it under degraded conditions or decommission it and employ a replacement. A small fraction of satellite that have malfunctioned have been repaired, however that avenue is expensive, limited to low earth orbit systems, and unfeasible in today's environment without a manned space vehicle capable of providing that type of access for mechanical work.

A SBG system, by virtue of each component of the system being fractionated, allow for flexibility in the event of different uncertainties, such as technical failures and component obsolescence (De Weck, De Nuefville, & Chaize 2004). A malfunctioning component can be replaced at cost of the single component plus launch. This can also be completed for system components that have become obsolete, or for a technology that has been matured that would dramatically improve the performance of the SBG system. The entire system would not need to be replaced for improvements; additional components need only join the network for the enhancements to be realized.

Additionally, a SBG system could be designed with an on-orbit servicing component, a satellite capable of performing minor repairs required during the lifetime of the system. These servicing satellites would be employed in large SBG clusters where quick replacement would not be feasible, such as in geosynchronous orbit; or a situation where failure of a major component could not be tolerated, such as the communications satellite of a high demand system. Orndorff and Zink state “fully autonomous servicing has been proven on orbit with the DARPA Orbital Express mission. Orbital Express successfully demonstrated autonomous rendezvous, cooperative and uncooperative docking, fuel transfer, and replacement of individual packaged components” (2010, p. 277). The design would need to be defined as an initial capability during development, but could be built to reduce overall lifetime cost.

D. CHAPTER SUMMARY

This previous chapter has developed an architecture comparison of the current monolithic systems and that of a SBG fractionated system. The current paradigm that the military utilizes to fulfil requirements is flawed and has resulted in requiring greater capabilities on a system in order to answer cost growth, which has had the effect of increasing those costs (Brown & Eremenko 2006b). The desire is to provide a replacement for these traditional, large monolithic satellites with a SBG of satellites, which would fly in a wirelessly linked network (Orndorff & Zink, 2010). This architecture would allow for parallel development of components, transition to integration of a system-of-systems, and a capability to repair or replace a component faster than current technologies allow.

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III. RESEARCH ANALYSIS

A. INTRODUCTION

This chapter will utilize the FireSat example architecture found in SMAD and a fractionated breakdown of the sensor will be analyzed for cost comparison. FireSat's architecture was defined by the requirement for an imaging platform to conduct forest fire detection and monitoring. This chapter will present these requirements and the process in which a suitable payload system was developed. The chapter will also examine the comparison between the communication requirements of FireSat and those required of a SBG, with the expectation of growth of the requirements of the constellation.

B. MISSION REQUIREMENTS

The first emphasis of any system is to define the overarching program objectives. The objectives are the strategic mission goals that the customer provides to the government, contractor, or other developmental units. With the environment of today, primary objectives and secondary objectives are often times developed to allow for multi-purpose or multi-functional systems. The FireSat program was given the following objectives, found in figure 1-4 in SMAD (from Apgar et al., 1999, p. 13). The primary objective was "to detect, identify, and monitor forest fires throughout the United States, including Alaska and Hawaii, in near real time" (from Apgar et al., 1999, p. 13) Additional secondary objectives were:

- To demonstrate to the public that positive action is underway to contain forest fires.
- To collect statistical data on the outbreak and growth of forest fires.
- To monitor forest fires for other countries.
- To collect other forest management data.

These objectives allowed for focused allotment of resources by the developmental teams, while still allowing free design to accomplish the desired goal. The objectives of a program are translated into high level mission requirements, which drive the specific architecture development for a system. The requirements provide specific, measurable performance specifications that the system is required to meet. Table 2 provides the top-level requirements that are derived for FireSat.

The FireSat requirements drive several aspects of the system, including orbit, sensor, and selection, payload operation and support, and launch. The following sections break apart these sections, as provided by SMAD, to fractionated type architecture. A focus of the research is based on the idea that each SBG satellite focuses on a single mission element, therefore simplifying engineering and integration efforts (Orndorff & Zink, 2010).

Table 2. FireSat Top-Level Mission Requirements (from Apgar et al., 1999, p. 16)

Requirements	
Functional	
Performance	4 temperature levels 30 m resolution 500 m location accuracy
Coverage	Daily coverage of 750 million acres within Continental United States (CONUS)
Responsiveness	Send registered mission data within 30 min to up to 50 users
Secondary Mission	4 temperature levels for pest management
Operational	
Duration	Mission operational at least 10 years
Availability	98 percent excluding weather 3-day maximum outage
Survivability	Natural environment only
Data Distribution	Up to 500 fire-monitoring offices +2,000 rangers worldwide (maximum of 100 simultaneous users)
Data Content, Form, & Format	Location and extent of fire on any of 12 map bases, average temperature for each 30 m^2 grid

Requirements	
Constraints	
Cost	Less than $20M/year plus research and development
Schedule	Initial operating capability within 5 years, final operating capability within 6 years
Regulations	NASA mission
Political	Responsive to public demand for action
Environment	Natural
Interfaces	Communication relay and interoperable through NOAA ground stations
Developmental	Launch on Space Transportation System (STS) or expendable No unique operations people at data distribution nodes

C. FRACTIONAL DIVISION

A fractional system provides the ability to diversify the system between different components and functions. The following paragraphs detail the breakdown of FireSat, as a monolithic system, to a SBG of a single optical sensor system.

1. Orbital Modeling

The initial mission element required to develop an effective space system is to determine where in space it will be designed to operate. The orbital modeling drives the payload and subsystem capabilities to achieve the mission. Selection of an orbit is often a compromise of the ability to perform to perfect specifications, and the least costs options of a specific payload desired. During analysis of alternatives, a specific payload capability may restrict the orbital parameters in which the system can operate. Additionally, subsystem capabilities or launch limitations will dictate some of the orbital parameters. This analysis must be done early and often during acquisition of a system, as slight details could impact the system’s capability to meet the requirements due to orbital placement. The orbit design required focuses on meeting the objectives of the FireSat program, specifically revisit time of a specific area for surveillance or communications.

The objective of FireSat was to be able to detect, identify, and monitor forest fires in real time; as well as provide some additional tasks, to include public reassurance of current forest fire containment. This drives a requirement of daily coverage within CONUS, with no more than three days of outage for availability, with an objective of including Hawaii, Alaska, and other regions. The ideal orbital position to provide real-time, continuous coverage would be a geostationary earth orbit (GEO). This would allow for 100 percent coverage of the continental United States, but would not allow a single satellite to provide coverage of Alaska and Hawaii. The cost increase for a payload and subsystem capable of providing the data also increases, as well as requirements in reaching GEO. Recent accounts of government acquisition programs utilizing conventional GEO systems have had negative results on cost, schedule, and performance (Orndorff & Zink, 2010).

Utilizing the requirements provided, the ideal orbit to satisfy the program needs was a circular orbit near 700 km with an inclination that provides the access times to CONUS. The design point utilizes an inclination of 55 deg at this altitude, and to utilize two systems with a Right Ascension of the Ascending Node (RAAN) offset of 120 deg (Apgar et al., 1999). The cost analysis completed only focused on a single system at RAAN of 0 deg, using the assumption the additional system offset provides the capability to fill the multi-hour gaps observed. The program System Tool Kit (STK) was utilized to develop an orbital model and to compute access periods for the imaging sensor and communication points. The desired communication ground stations utilized were the National Oceanic and Atmospheric Administration (NOAA) stations located in Alaska and Maryland. The inputs for orbital information, area of interest, ground stations, sensors, and communication systems are provided in Table 2. Figure 3 provides a 2D view of the area of interest, the FireSat ground trace for a single orbit, and the ground station locations that were be utilized. Access times to the area of interest and ground stations are illustrated in Figure 4 and Figure 5, respectively. The assumption was made that a FireSat SBG has an identical orbits and imaging sensor. As displayed in Figure 5, a single sensor would have daily access to either ground station.

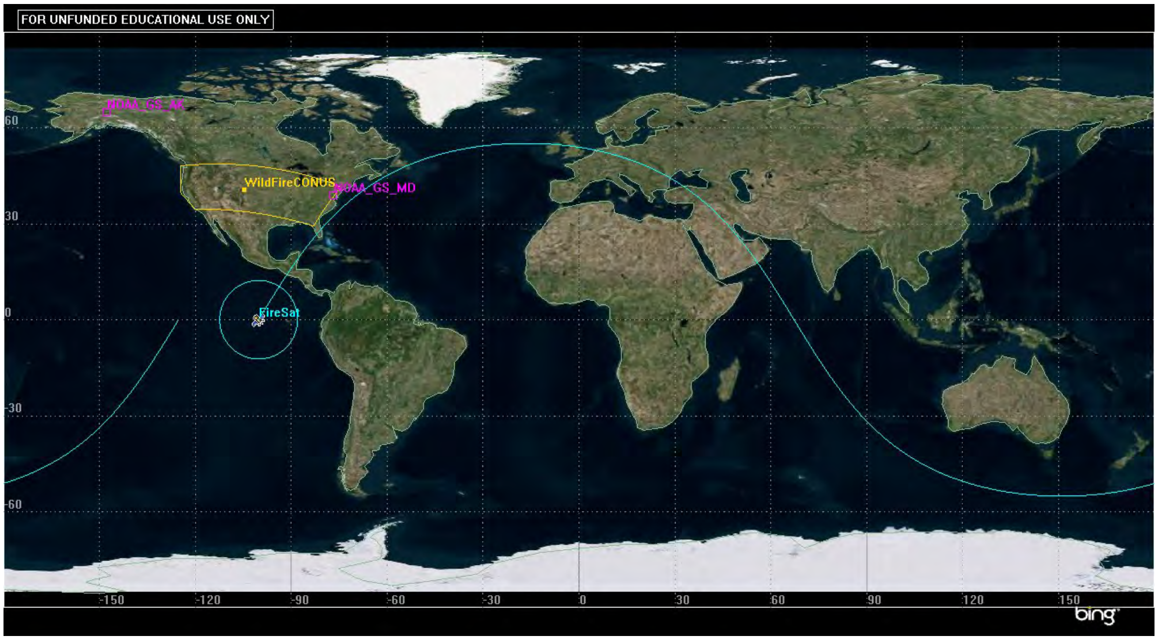


Figure 3. FireSat 2D Model from STK v. 10

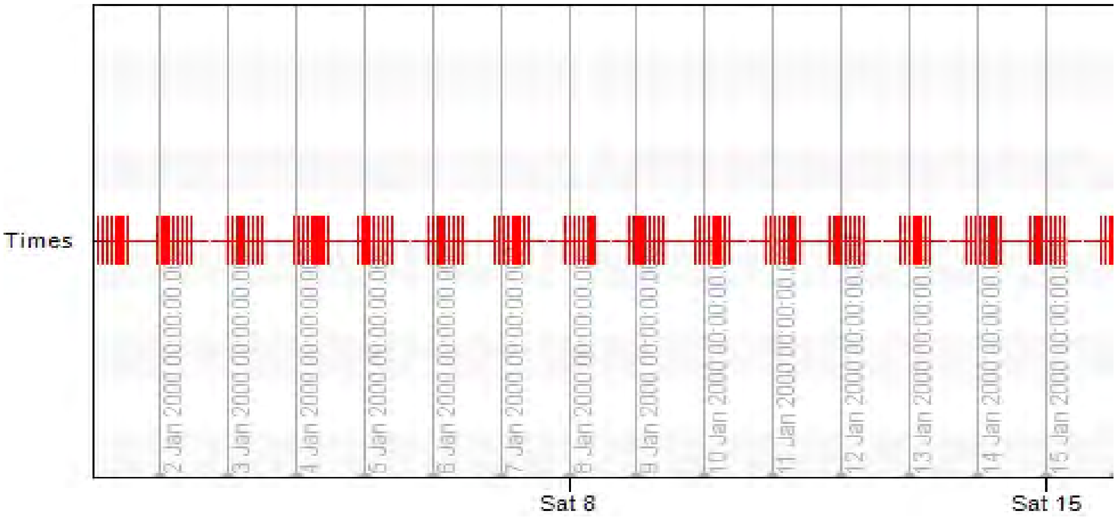


Figure 4. FireSat Observation of Wildfire Area from STK v. 10

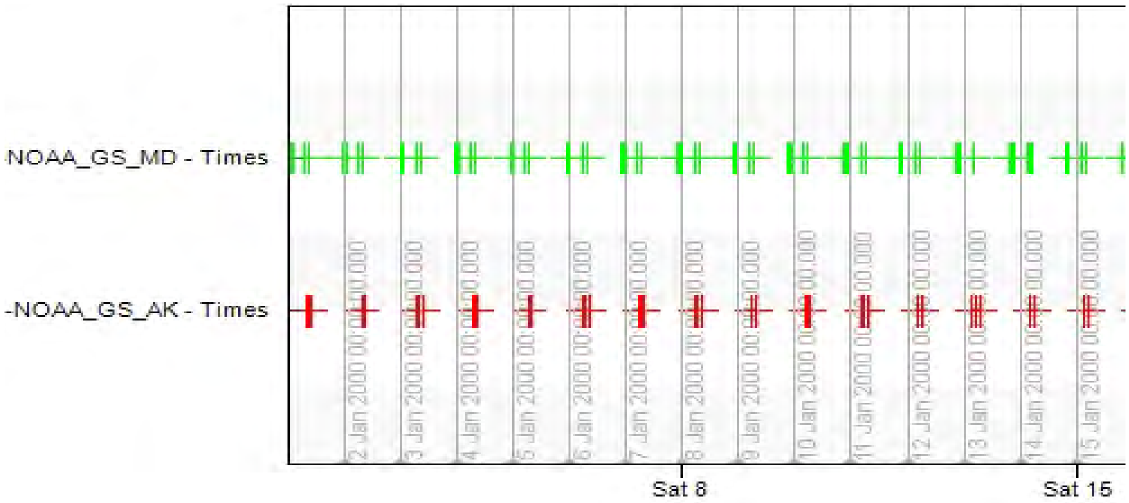


Figure 5. FireSat Communication Access to NOAA Ground Stations from STK v. 10

2. Sensor Selection

An observation and alert platform, such as FireSat, requires an optical sensor capable of not only monitoring large area fires, but detection of small fires to allow for early response for prevention of large area damage. This optic requires high fidelity to perform this detection, and an ability to perform day or night detection. This initial requirement leads to utilization of a passive Infrared optic.

To develop a preliminary understanding of the optic, it is required to step through design process, as provided in SMAD pages 293 through 297. Table 3 outlines the calculated results based on the requirements of 30 m resolution and selection of a 700 km, circular orbit.

Table 3. Sensor Design Parameters (from Apgar et al., 1999, pp. 293–297)

Parameter	Value	
Orbital Altitude	700	km
Orbit Period	98.8	min
Ground Track Velocity	6.76	km/s
Node Shift	24.8	deg
Earth angular radius	64.3	deg
Max. Horizontal Distance	3069	km
Max. Incidence Angle	70	deg
Nadir Angle	57.9	deg
Min. Elevation Angle	20	deg
Earth Central Angle	12.1	deg
Slant Range	1578	km
Swath Width	24.2	deg
Max. Ground Sampling Distance along track	68	m
Instantaneous field of view	0.00245	deg
Max cross-track pixel resolution at ECAmax	199.6	m
Cross-track ground pixel resolution at nadir	90	m
Along-track pixel resolution and nadir	90	m
Number of cross-track pixels	4.70E+04	
Number of swaths recorded along-track in 1 second	225.6	
Number of pixels recorded in 1 second	1.06E+07	
Number of bits used for each pixel	8	bits
Data rate	85	Mbps
Number of pixels for whiskbroom Inst.	256	
Pixel integration period	24.1	µs
Pixel read-out frequency	42	kHz

This information is the basic starting block for the design of the system. This information feeds the requirements of the sensor and communications suite. SMAD also provides a typical payload characteristics table that can allow for desired payload estimates, and is found on page 275 (Apgar et al., 1999). For the given FireSat requirements, a Multi-spectral Mid-IR optic was used to compare to the requirements of the sensors, with adjustments to the sensor characteristics based on aperture ratio of 0.26. This resulted in the following payload size:

- size: 0.4 m x 0.3 m diameter
- mass: 28 kg
- average power at 28 V: 32 watts

The mass and power results were based on an increased factor of 2, due to the substantial reduction in the payload size (Apgar et al., 1999, p. 297).

3. Operation and Support

As discussed in an earlier chapter, the operation and support cost of a satellite system is approximately 30 percent of the total lifetime cost of the system. The following sections express the how a SBG system can reduce risk and cost for initial operation and expresses the benefits a sensor only platform provided to everyday operations. Reparability of a SBG is also explored; however current cost and risk comparison are not within the scope of this research.

a. Initial Operation Capability

Often times the initial operational capability depends upon the current technological maturity of the subsystems being incorporated into the satellite and ground support stations, as discussed in an earlier chapter. The FireSat example shows two specific areas that require modifications in order to obtain the best result with the lowest cost. These areas are the control and data management, as well as the IR interface, to include noise reduction. Once a cost trade-off decision is chosen, the change to any of the parameters will result in a large impact to cost and schedule. Additional changes, to include updates to technology or any other innovation that could reduce operational risk to the program would also impact cost and schedule, with possible increases performance.

The requirement to select a single sensor to maintain lower cost is a key element in the drawbacks of a monolithic system. Although selection of multiple sensors will result in higher cost with regard to payload design, the ability to diversify the payloads between satellites, therefore limiting the size and weight of each platform, may result in the deployment costs for multiple small satellites comparing favorably to the deployment of a single system.

Additional benefits of a SBG are the separate design and production pipelines that can be used. Each individual element can be built independent of the others, allowing the communications satellite subsystem to be decoupled from system payload requirements. This allows for rapid development and deployment of the system. The development structure can be viewed as a system of systems with a larger control of the systems in development, allowing for limited problems of synchronization implementation. If a sensor platform falls behind schedule, at least a partially capable system can be deployed, awaiting the completion of the remaining parts. These remaining parts can join the deployed system at a later time (Orndorff & Zink, 2010).

b. Operation and Reliability

The operation of a system is dependent on each component of the system, and their reliability in both singular operation and successful integration with the spacecraft and ground stations, as required. The payload data collection processing by the onboard system is a large limitation to a monolithic system. An image sensor, as provided by the FireSat example, can allow for high data rates and large signal processing meeting the design capability of the sensor (Apgar et al., 1999). Often, the ability of the satellite's communication or processing systems to meet the sensor's capability limits the amount of data provided by the system. This requires natural tradeoffs to meet the system requirements, and processing to adjust the information provided from the sensor to the operator.

Payload control and data management onboard a satellite platform require analysis of architectural options in order to determine the best course of action with the lowest impact to cost. SMAD provides an example of these architectural comparisons for FireSat in table 16-22 on page 679 (Apgar et al., 1999). The table displays different processor and interface options and the resultant pros and cons of each of the four options. The requirements of a satellite to perform the processing of sensor data increases the complexity of the design, adding cost, complexity, and weight to accommodate the number of general processors; or reducing the reliability of the system to provide the requisite threshold needs in order to save cost by reduction in processing capabilities. The final determination, as made in SMAD, is to utilize a full Field of View sensor with two processors (Apgar et al., 1999).

Utilizing an SBG design, the gathered data can be off loaded from the sensor platform to the communication hub, or a separate processing satellite. The utilization of a separate satellite for processing data allows for a reduction of power requirements of the sensor satellite. This reduction in power allows for a reduction in complexity of the satellite.

c. Repairability

The monolithic structure of FireSat allows for coverage and detection through the utilization of two satellites, placed in 90 deg offset orbits (Apgar et al., 1999). The highest danger to cost and operation of this type of system, as expressed by Brown and Eremenko, is that a loss in an element will result in a complete loss of capability; whereas a SBG leads to a reduction in capability as a result of the individual module that failed (2006b). Monolithic systems have several options in the event of a failure of a component, either repair or replace the component on orbit, or dispose of the current satellite and replace it with an operational system. The feasibility of repairing or replacing onboard components while in orbit is only found in a single instance in current spacecraft history, and that is the Hubble Space Telescope. As a result of the decommissioning of the shuttle fleet, there are currently no satellites or servicing spacecraft in operation capable of providing reparability services to Hubble. Replacement of an entire satellite is the only recourse for a failed major system component that does not include a redundancy.

A SBG system reduces the cost of the replacement of components, as only a single module is needed, vice the entire constellation. It allows for a new component to be placed into the wireless network, enabling the ability to replace components without the need for complex robotic servicing (Sahel, 2006). An imaging system, such as FireSat, is not restricted by the mean reliability risk of the many components necessary to achieve the detection of wild fires. Each component can be individually replaced in the event of failure. Additionally, each component can theoretically by replaced in the event of large technological improvements, allowing for requirements to be adjusted over the lifetime of the system, by just allowing deployment of additional modules (Brown & Eremenko, 2006a). This simple change in architecture provides a means to distribute schedule and operation risk, reducing overall cost of the program.

4. Launch Segment

The capability to deliver a space system to orbit is an additional factor that can greatly impact the size and weight allowed for a satellite. If unaccounted for, the launch system could result in major delays in a program due to redesign to match the requirements of the desired launch system. There are several considerations to make when choosing a launch vehicle; and the ability to launch several SBG segments in one launch is a consideration that could result in reduced program costs.

SBG architecture allows for reduction in the size of the system, payload, and propulsion systems. This allows for the flexibility to select from many different launch vehicles (Orndorff & Zink, 2010). In modeling a SBG system, the cost impact is reduced because of the small size of the satellite, allowing individual modules to launch separately on smaller vehicles, or as add-ons to other launches. Additionally, De Weck states, when calculating the launch costs to achieve 99.9 percent probability of successful on-orbit operations, the reasonable launch cost and fractionation penalty assumption result in launch costs reduction in two or more than a traditional system. This analysis removes the potential of a SBG system providing value if all modules do not make it to orbit (De Weck et al., 2004).

Analysis and research done by the SMAD authors determined the most cost effective launch system would be the Pegasus rocket. Calculations of the FireSat system provide approximately 225 kg of total satellite weight. Based on the *Pegasus User's Guide* Release 6.0, the Pegasus launch system can deliver FireSat to the requisite orbit, as seen in Figure 6 (*Pegasus User's Guide*, 2007).

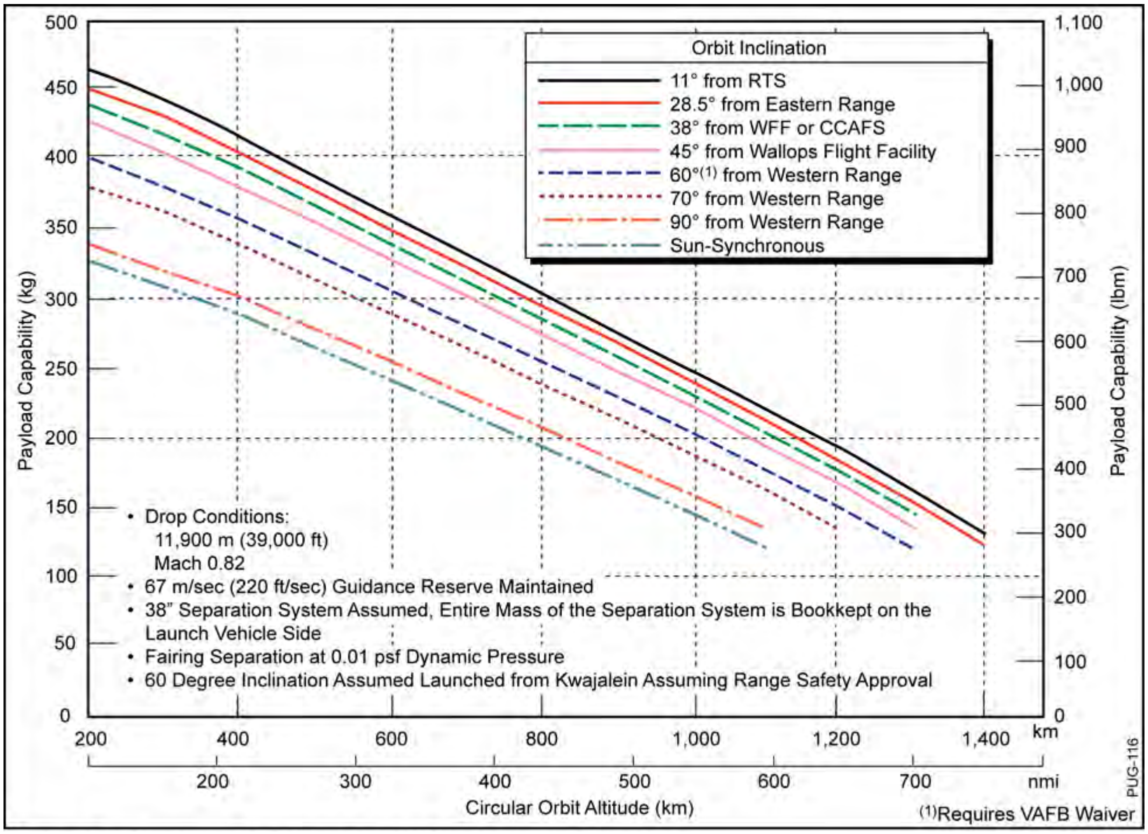


Figure 6. Pegasus XL without Hydrazine Auxiliary Propulsion System Performance Capability (from *Pegasus User's Guide*, 2007, p. 11).

The SBG design comparison results in a reduction of total weight to 85kg. The separation of the communication suite from the imaging platform also reduces the size of the sensor platform, allowing the analysis of placing several satellites in the Pegasus XL launch system.

5. System Communications

The backbone of each and every space-based system is communications. This is no less true for nanosatellites in LEO than for military communications satellites, such as the Advanced Extreme High Frequency system. Communications are required to transmit payload information to the desired ground stations, provide Telemetry, Tracking, and Control (TT&C), and afford relay of information between satellites, in some cases.

Within a monolithic construct, each satellite is comprised of the communications downlink as well as TT&C uplink components. This is another piece, including the sensor payload and the processing and storage required for that payload, which drives up costs. The communications requirement adds power and weight to the overall satellite design, and those elements grow with the complexity required of the communications suite.

The following section examines the difference between the requirements of satellite to satellite communications and that of a ground communications link that would be established for a SBG.

a. Space-Based Communications

A benefit of a SBG is that the core communication satellite, or central hub, provides a dedicated, high-bandwidth ground link that is several times larger than a single satellite system, allowing for transmission of hundreds or more megabits (Orndorff & Zink, 2010).

Operational today, there are a few examples of satellites that utilize space communications, relaying information from one satellite to another. These typical satellites convey large communications packages, often times allowing for seamless communications between command nodes across the world. A SBG system is fully dependent on space based communications, if at a smaller scale, than most global or planetary relay systems. Communications between systems in a SBG involves just a replacement of the current monolithic data bus structure, and replacing it with wireless data links between the individual satellites (Saleh, 2006).

SBG architecture is fixed around a centralized communications hub, which would serve as the pathway for TT&C and sensor data. Each additional module is designed such that it could join and integrate into the hub LAN, similar to current terrestrial wireless networks available around the world. This enables a hub and spoke system configuration, allowing the communication satellite to contain the router for the LAN (Orndorff & Zink, 2010). The network would allow for positional and health data from each spoke to be independently reported to the hub system. An example of this is approached in the F6 and the Mission-Aware Network Architecture for Small-Satellite Adaptive Systems (MANASSAS). These systems utilize a cognitive networking approach, giving the systems the ability to change and modify its behavior in flight. This allows for a significant enhancement in a systems effectiveness and flexibility (Waite, 2011).

The TT&C of individual modules would be relayed through the communications hub. Orndorff and Zink propose that each module should contain its own capability of determining its location, utilizing GPS or star and horizon trackers (Orndorff & Zink, 2010). While a GPS system is adequate for a Low Earth Orbit (LEO) system, Medium Earth Orbit and GEO systems would require the use of stellar locating sensors, further complicating the system architecture. A separate location computation could utilize interferometer architecture. This would require each module to report information on different frequencies. This would allow a single interferometer to establish the relative position of each module, and utilize its own location device to relay the full details of the constellation position to the TT&C stations. The utilization of interferometers is prevalent in several terrestrial systems, to include the new P-8A Poseidon aircraft, which uses interferometer measurements to accurately maintain current locations of sonobuoys based on relative location to the aircraft, and the aircraft's geolocation (Department of the Navy Office of the Chief of Naval Operations, 2014).

The concept of SBG systems is to utilize current wireless technology to achieve the constellation integrated LAN. There are several different options to examine in determining the best system to select for this LAN. Initial studies indicate that RF transmissions are more favorable at distances below several hundred meters, with modular distance of several kilometers favoring laser transmissions. The highly favorable frequency bands for RF are the V or W band (Sahel, 2006). An experiment conducted onboard the Mir space station, designated at the Mir Wireless Network Experiment (MWNE) provided risk mitigation for the ISS, in proving the concept of a space based wireless network. The MWNE was conducted onboard the Space Shuttle mission STS-74 and Mir. The experiment operated a 2.4 GHz wireless network at a power output of 50mW (Lofton & Conley, 1997). This concept, although not performed for satellite to satellite communications, does provide the basis that a 2.4 GHz network can be established between systems in orbit.

Current wireless technology allows for transmission on 2.4 GHz or 5 GHz frequency bands, operating with an average power output of 20 dBm. This design, assuming a wireless link based on current 802.11 series, weighs tens of pounds, requires little power, and fit into small spaces, even when including a 12 dB antenna for transmission (Orndorff & Zink, 2010). In this study, a wireless link of 2.4 GHz was utilized, as this frequency spectrum has been previously tested in spaceflight, as referenced in Lofton and Conley MWNE report (1997). Several wireless LAN systems on the market today allow for dual transmission on these frequencies. The router examined in this research was the Asus RT-AC68U, based on a top pick by CNET.com reviewers. As a top pick, the router provided a continuous 381 Mbps throughput speeds at a distance of 30 m, which was stated as “still very impressive” (Ngo, 2014). The assumption is that due to the decrease in atmospheric attenuation, the router would prove as capable at a 50 m distance, as compared to the 30 m in an Earth bound test. This router, even as a larger model, has a weight of only 26.3 oz., takes up a total of 0.00234 m^3 (Ngo, 2014).

b. Ground Communications

An additional requirement of every space based system is the communications of data to the user, and directives to the system. The required downlink to the ground station, as simply described by Jerry Sellers, is information on the health and status of the spacecraft as well as the designed payload. This link has the largest amount of data to transport, and as a result, is often the limiting factor in the amount of information that can be collected, stored, and transmitted. When examining the varying requirements of a SBG, there are two primary avenues to explore in determining the architecture of the ground communications network.

(1) Single Site Support

Single site support architecture is what is primarily utilized in today's space missions. The construct that information from the downlink, as well as the uplink, is focused under a single unit, often with various link locations throughout the country or world. For utilization in SBG architecture, this would form as a central network storage area, similar to the network cloud concept being utilized by several different companies today. All of the information gathered by all the satellites would be transmitted to a central location, with each payload being specifically encoded for cataloging in the network. This would require several link sites, and a large area for processing and storage of the data, so that users could retrieve the data as needed. This is simplistic, in that the basic parameters of the architecture are developed, and only the catalog and distribution of payload information would need specific development per program.

(2) Mission Specific Site Supports

A mission specific site support would be ideal for a massive SBG undertaking that included a mixture of civilian and DOD, or open and sensitive information. It would ensure that data would be downloaded to sites only designated to receive that information, limiting the ease of sensitive information being leaked or intercepted. The architecture to utilize this type of support is not as widely utilized in today's market. It is further complicated within a SBG construct. A mission specific, multi-site support would require the communication hub to be capable of individually transmitting data, based on a type of encoding or encryption, to a specific site. Due to the inevitability of having more than one site within the communication field of view, it would also require the ability to perform a downlink on different frequencies, to ensure no cross link of data flowing at the same time. This architecture, while beneficial, may result in an undesired cost increase of a program, due to these requirements.

To allow for direct analysis, a NASA Tracking and Data Relay Satellite (TDRS) was used as a currently operational system that could provide the functions desired of the communications hub for a SBG. The TDRS currently provides simultaneous relay between ground stations and multiple satellites (Bell, 2014). The scope of a communications hub would be reduced slightly, in that the satellite would only be required to connect to the wireless network, and not be responsible for relay to multiple systems within different orbits. SMAD provides, in table 20-16, as Space Systems cost estimate of each TDRS, which estimates an average cost of $172 million per unit (Apgar et al., 1999). More information on the capabilities of TDRS can be found on the NASA Space Communications and Navigation web site (Mai, 2013).

Separating the sensor element from the communication suite of a single unit can give increased capability, but it does add additional costs to the program. The cost savings are realized within the combination of several sensor units, each utilizing a single communications hub, for data transmission and reception. The communications platform would add to the program cost, but the effectiveness of adding several sensors under a single program would result in overall cost savings.

D. CHAPTER SUMMARY

The research completed and analyzed in this chapter shows the specific correlations and explicit detailed differences between a common monolithic space system and that of a SBG system. Each area examined is required for detailed acquisitions, and must be done in conjunction with all others in order to prevent delays and cancellations in a program. Each section provided the required inputs and resulting outputs for the FireSat

SBG system. All data inputs and outputs calculated for FireSat SBG Sensor Design can be found in Figure 7 thru Figure 14 appendix.

The DARPA F6 project is a prime example of a program omission that resulted in a million dollar system being canceled, in that the program lacked an overall integrator. Additionally, Brad Tousley stated that the system was not focused on a traditional mission set, “only data transfer—there was nothing else happening” (Ferster, 2013).

IV. APPLICATION OF STUDY

A. ANALYSIS SUMMARY

The previously stated research and analysis conducted provides a breakdown of a monolithic imaging platform into a SBG sensor system, and the estimated cost savings of that single system. It shows how removing the communications component from the sensor platform would change the overall structure of the architecture requirements, and how this separation can benefit a program. The following chapter utilizes the assumption that the majority of systems would experience the same relative cost savings, and shows the estimated program savings when combining FireSat with three currently operational systems in orbits. The systems utilized were the Fermi Gamma-ray Space Telescope (FGST), Galaxy Evolution Explorer (GALEX), and Swift Gamma Ray Burst Explorer (SWIFT) spacecraft. These systems were selected due to their relatively close orbital and inclination to each other, and the assumption FireSat could operate within those parameters. Additionally, these sensor payloads all utilize an imaging type sensor, so a general relationship can be made between them. Table 4 provides basic details on each system. Further information can be found on the NASA National Space Science Data Center (Bell, 2014). An estimated payload size was determined based on a default assumption that the payload was approximately 26 percent of the total spacecraft weight. The space segment portion of the cost analysis was used, assuming the operational costs between a fully monolithic system and a SBG sensor would be similar, due to the mission being unchanged. This is a rough assumption, as the type of ground architecture utilizing a single point download system, as described earlier, could reduce the cost of the program due to the reduction in overall footprint required of the systems as a whole. Additionally, the launch costs were not included, as the launch platforms of the original systems was not researched.

Table 4. System Example Parameters (from Bell, 2014)

System	NORAD ID	Perigee (km)	Apogee (km)	Inclination (deg)	Total Mass (kg)	Estimated Payload Size (kg)
FireSat	Example	700.0	700.0	55.0	175.0	28.1
FGST	33053	527.9	545.9	25.6	4303	1160
GALEX	27783	685.3	690.3	29.0	280.0	75.6
SWIFT	28485	554.0	570.8	20.6	1470	397

B. COST COMPARISONS

This cost savings determined are primarily found in the R&D and manufacturing of the system architecture. The cost savings are fully realized when combining several systems into a single architecture system, due to the savings provided by a sensor only platform, when compared to the monolithic system. Additional savings include reliability of each module and module replacement; however those specific topics were not fully researched.

The cost analysis of FireSat was compiled utilizing the data found in the Appendix, compared to the results provided in SMAD, specifically utilizing table 20-24 on page 816. The \$60.1 million estimate was based on the Small Satellite Cost Model (SSCM) provided in table 20-6. This space based cost was approximately 44 percent of the total estimated total cost of the system over 5 years, as provided by table 20-24 (Apgar et al., 1999). The space segment sum of components estimated cost of the sensor FireSat system, as a result of the limited communications requirement, resulted in a cost of \$23.9 million, saving just over \$36 million (60 percent) as compared to the SMAD estimate.

A cost analysis of the four systems, developed as only sensor platforms, and the addition of a TDRS type communication hub was done to fully realize the savings capable of SBG architecture. In order to remain within the scope of this thesis, a few assumptions were made toward the three operational systems.

- The space segment cost of each system matched FireSat, in that it was 44 percent of the total operational cost
- The reduction to a sensor only platform reduced the space segment cost of each system to 40 percent of the original space segment cost, as per the FireSat example.
- The TDRS cost accurately represents the space segment cost of a communication hub.

The total lifetime cost of each operational system was researched and converted into FY15 dollars, and was determined to be $233.2 million for FGST (SLAC National Acceleration Laboratory, 2007), $160.8 million for GALEX (Clark, 2012), and $317.7 million for SWIFT (Neal, 2004). Under the assumption each system followed a similar cost fallout as described in earlier chapters, in that 70 percent of costs were utilized for acquisitions, Table 5 provides a breakdown of lifetime cost to estimated SBG Space Segment cost.

Table 5. Space Segment Cost Analysis Results

System	Operational Cost ($M)	Estimated Space Segment Cost ($M)	Estimated SBG Space Segment Cost ($M)
FireSat	$85.86	$60.1	$23.9
FGST	$233.2	$163	$65.3
GALEX	$160.8	$113	$45.0
SWIFT	$317.7	$222	$90.0

Upon combining all four systems into a SBG, the expected savings is approximately $224 million. When combining with the TDRS, with a unit cost of $172 million, the total savings provided by the SBG space segment alone is $52 million.

C. RECOMMENDATIONS

This thesis presented arguments that SBG architectures could provide beneficial improvements that can result in reduced cost of a space system. A detailed breakdown of an example system combined with assumed similarities to three additional, currently operational systems, shows that there is a realistic cost savings to the space segment alone, when combining multiple systems into SBG architecture. Research also theorizes that SBG architecture could provide a level of flexibility without resulting in an increase in cost when adding additional sensors to orbit, as compared to current concepts.

A broad analysis was done by Charlotte Mathieu (2006) in her paper assessing the fractionated spacecraft concept. Her conclusion was:

> Fractionated spacecraft could deliver more value to customers than traditional ones for a given mission and at a given level of performance. As demonstrated in the architectural analysis, the more fractionated the architecture is, the more expensive it is. But in terms of value delivered to potential customer the "best" fractionated architectures are different in various fields of application (communications, navigation, etc.). (p. 136)

V. CONCLUSIONS

This thesis researched the development of a SBG, focusing on the estimated advantages as well as an estimated cost savings for the space segment architecture. The full realization of cost savings requires further research, but initial analysis shows that there is a point in which cost savings will be understood with a SBG over a monolithic architecture.

A. KEY POINTS AND RECOMMENDATIONS

The thesis attempted to answer two questions.

- Can the SBG architecture reduce cost of space systems?

Research and analysis of FireSat, as provided by formulas in SMAD, shows that approximately 60 percent of savings per system can be gained. When combining several systems together, this cost will offset a requirement to have a centralized communication hub for SBG architecture.

- Could a SBG allow mission flexibility without substantial cost increases?

Research provides examples of how the flexibility SBG could provide throughout a system's lifecycle can save millions of dollars in expenses. This savings is difficult to fully capture, as it is dependent of the specifics of each architecture, however a safe hypothesis is that the added flexibility provide will not increase cost, and may prove to decrease the costs over a system's lifetime.

The SBG architecture shows promise in having the capability to provide the DOD with cost effective space systems, and could result in a whole new paradigm. Research done by Ms. Mathieu (2006) shows that private sector organizations have little incentive to move toward a fractionated architecture, but the government has the best position to enable this\ transition to SBG systems. This requires focused research and contracting in order to ensure mission capability and integration is captured. The DARPA F6 project is a casualty of this lack of focus, and will result in further delay in acceptance of SBG architectures for future implementation.

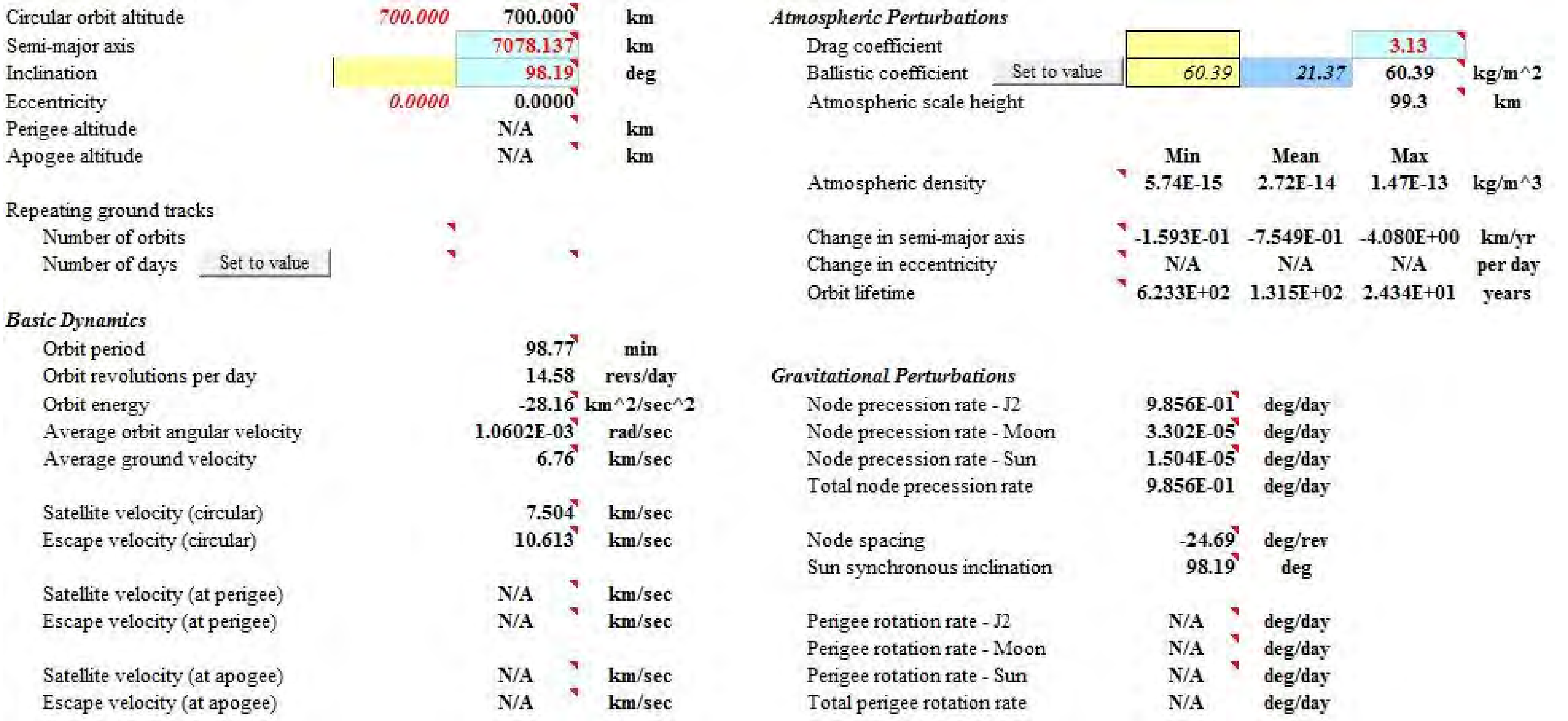
B. AREAS FOR FUTURE RESEARCH

The research presented in this thesis took a defined separation of a sensor platform and the communication suite, and adding three similar systems to the architecture to realize an estimated cost savings to the space segments of these architectures. Further research and analysis should be done to attempt to capture cost savings in relation to communications and operations of SBG, as well as risk-cost comparisons that a SBG provides. The risk-cost comparison includes further system fractionation, reliability benefits with SBG elements, and mission complexity requirements as a result of fractionated systems. In specifics to reliability, Brown and Eremenko express the benefit in the quote:

> If treated as a design parameter endogenous to the design process, the reliability of each module can be independently set to maximize the net value delivered by the system in light of considerations such as: the different paces of obsolescence for different technologies may make the early replacement of some modules (most notably C&DH) desirable; the cost of implementing a given degree of reliability may be starkly different across modules; and the degree to which the health of a particular module is vital to the capability of the overall system (i.e., its covariance with the other modules) may differ depending on the system architecture, degree of homogeneous fractionation, and the types of connectivity. (2006. p. 10)

Additionally, technological advances over extended life mission should be considered, as obsolescence does not threaten the entire system, just the single portion that platform is placed (Orndorff & Zink, 2010).

APPENDIX: FIRESAT SBG SENSOR DESIGN

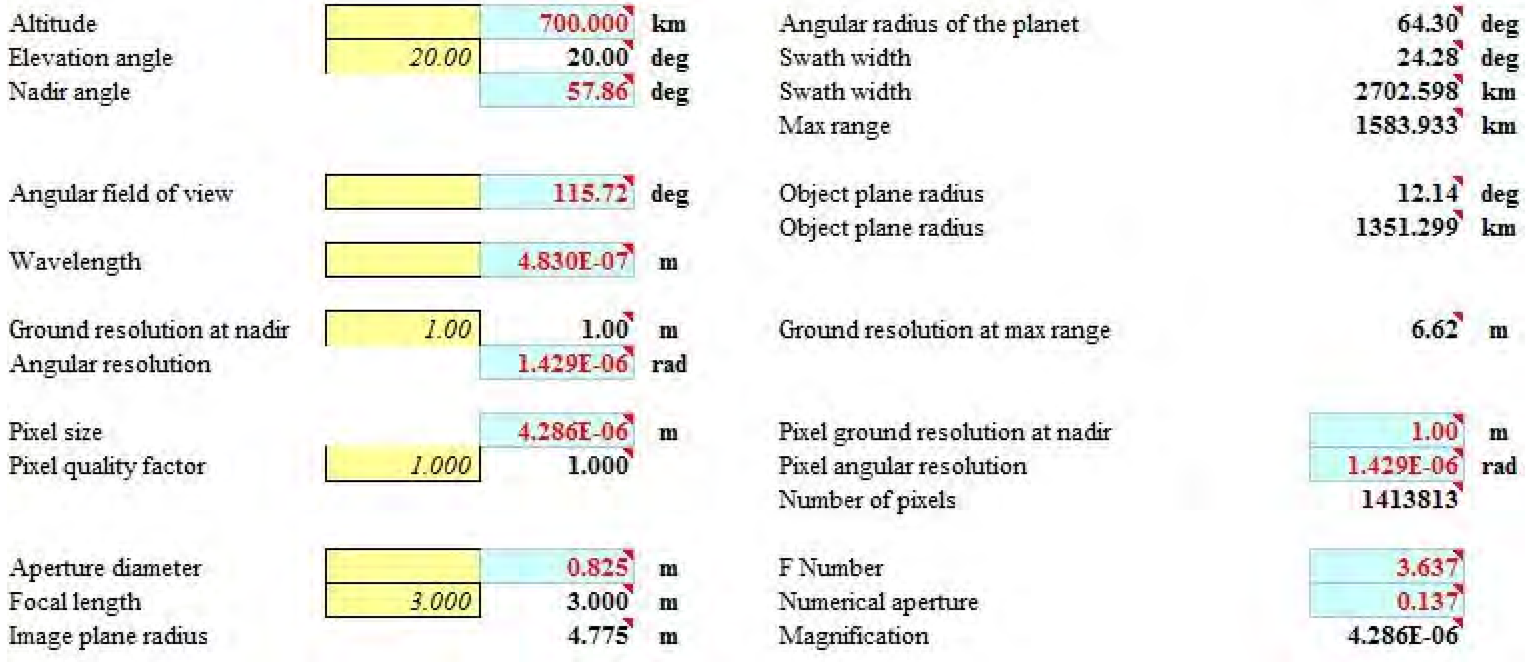


Circular orbit altitude	700.000	700.000	km
Semi-major axis		7078.137	km
Inclination		98.19	deg
Eccentricity	0.0000	0.0000	
Perigee altitude		N/A	km
Apogee altitude		N/A	km
Repeating ground tracks			
Number of orbits			
Number of days	Set to value		
Basic Dynamics			
Orbit period		98.77	min
Orbit revolutions per day		14.58	revs/day
Orbit energy		-28.16	km^2/sec^2
Average orbit angular velocity		1.0602E-03	rad/sec
Average ground velocity		6.76	km/sec
Satellite velocity (circular)		7.504	km/sec
Escape velocity (circular)		10.613	km/sec
Satellite velocity (at perigee)		N/A	km/sec
Escape velocity (at perigee)		N/A	km/sec
Satellite velocity (at apogee)		N/A	km/sec
Escape velocity (at apogee)		N/A	km/sec

Atmospheric Perturbations					
Drag coefficient				3.13	
Ballistic coefficient	Set to value	60.39	21.37	60.39	kg/m^2
Atmospheric scale height				99.3	km
		Min	Mean	Max	
Atmospheric density		5.74E-15	2.72E-14	1.47E-13	kg/m^3
Change in semi-major axis		-1.593E-01	-7.549E-01	-4.080E+00	km/yr
Change in eccentricity		N/A	N/A	N/A	per day
Orbit lifetime		6.233E+02	1.315E+02	2.434E+01	years

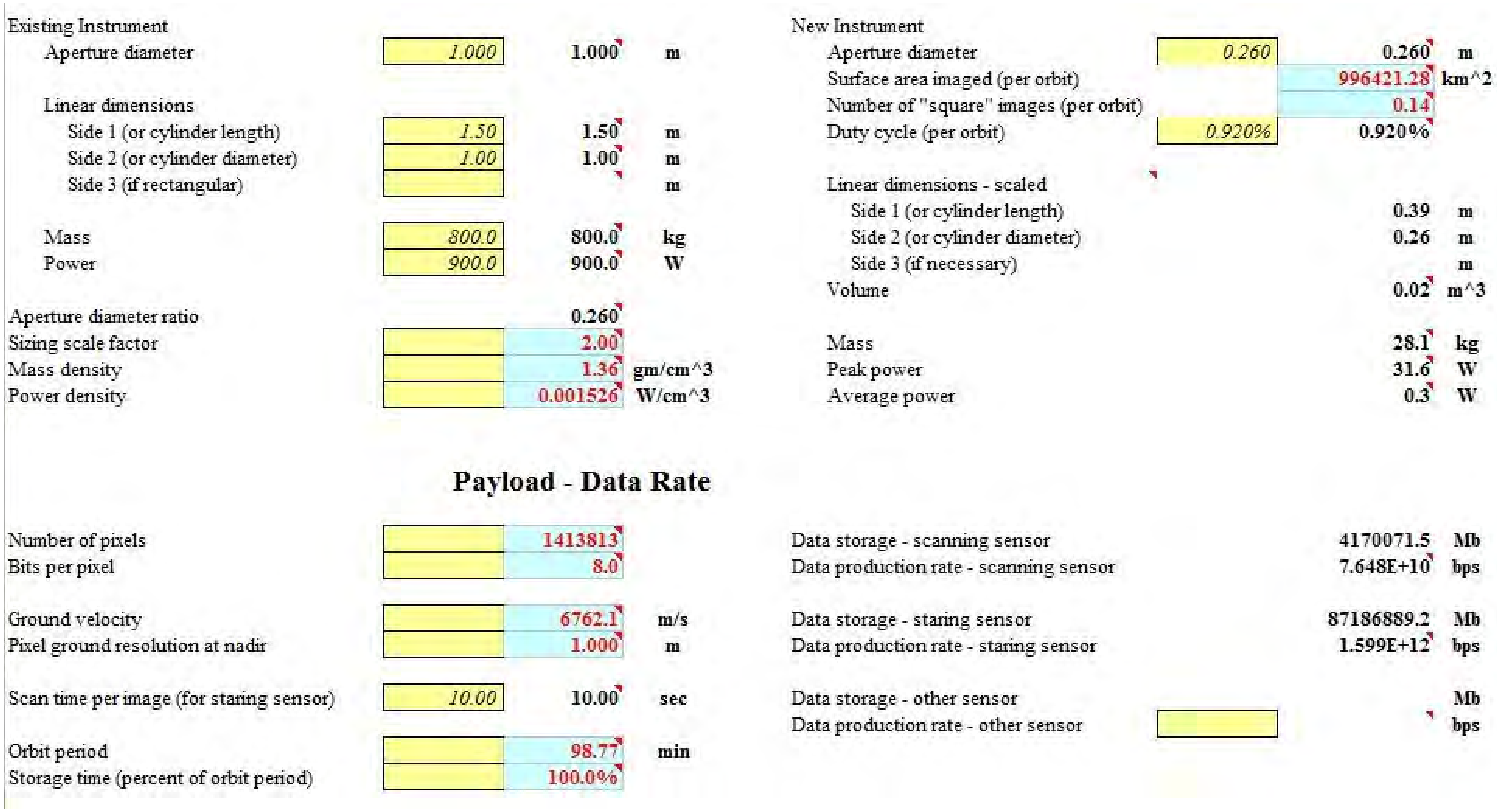
Gravitational Perturbations		
Node precession rate - J2	9.856E-01	deg/day
Node precession rate - Moon	3.302E-05	deg/day
Node precession rate - Sun	1.504E-05	deg/day
Total node precession rate	9.856E-01	deg/day
Node spacing	-24.69	deg/rev
Sun synchronous inclination	98.19	deg
Perigee rotation rate - J2	N/A	deg/day
Perigee rotation rate - Moon	N/A	deg/day
Perigee rotation rate - Sun	N/A	deg/day
Total perigee rotation rate	N/A	deg/day

Figure 7. Orbital Inputs



Altitude		700.000	km	Angular radius of the planet	64.30	deg
Elevation angle	*20.00*	20.00	deg	Swath width	24.28	deg
Nadir angle		57.86	deg	Swath width	2702.598	km
				Max range	1583.933	km
Angular field of view		115.72	deg	Object plane radius	12.14	deg
				Object plane radius	1351.299	km
Wavelength		4.830E-07	m			
Ground resolution at nadir	*1.00*	1.00	m	Ground resolution at max range	6.62	m
Angular resolution		1.429E-06	rad			
Pixel size		4.286E-06	m	Pixel ground resolution at nadir	1.00	m
Pixel quality factor	*1.000*	1.000		Pixel angular resolution	1.429E-06	rad
				Number of pixels	1413813	
Aperture diameter		0.825	m	F Number	3.637	
Focal length	*3.000*	3.000	m	Numerical aperture	0.137	
Image plane radius		4.775	m	Magnification	4.286E-06	

Figure 8. Optics Payload Information



Existing Instrument			
Aperture diameter	*1.000*	1.000	m
Linear dimensions			
Side 1 (or cylinder length)	*1.50*	1.50	m
Side 2 (or cylinder diameter)	*1.00*	1.00	m
Side 3 (if rectangular)			m
Mass	*800.0*	800.0	kg
Power	*900.0*	900.0	W
Aperture diameter ratio		0.260	
Sizing scale factor		2.00	
Mass density		1.36	gm/cm^3
Power density		0.001526	W/cm^3

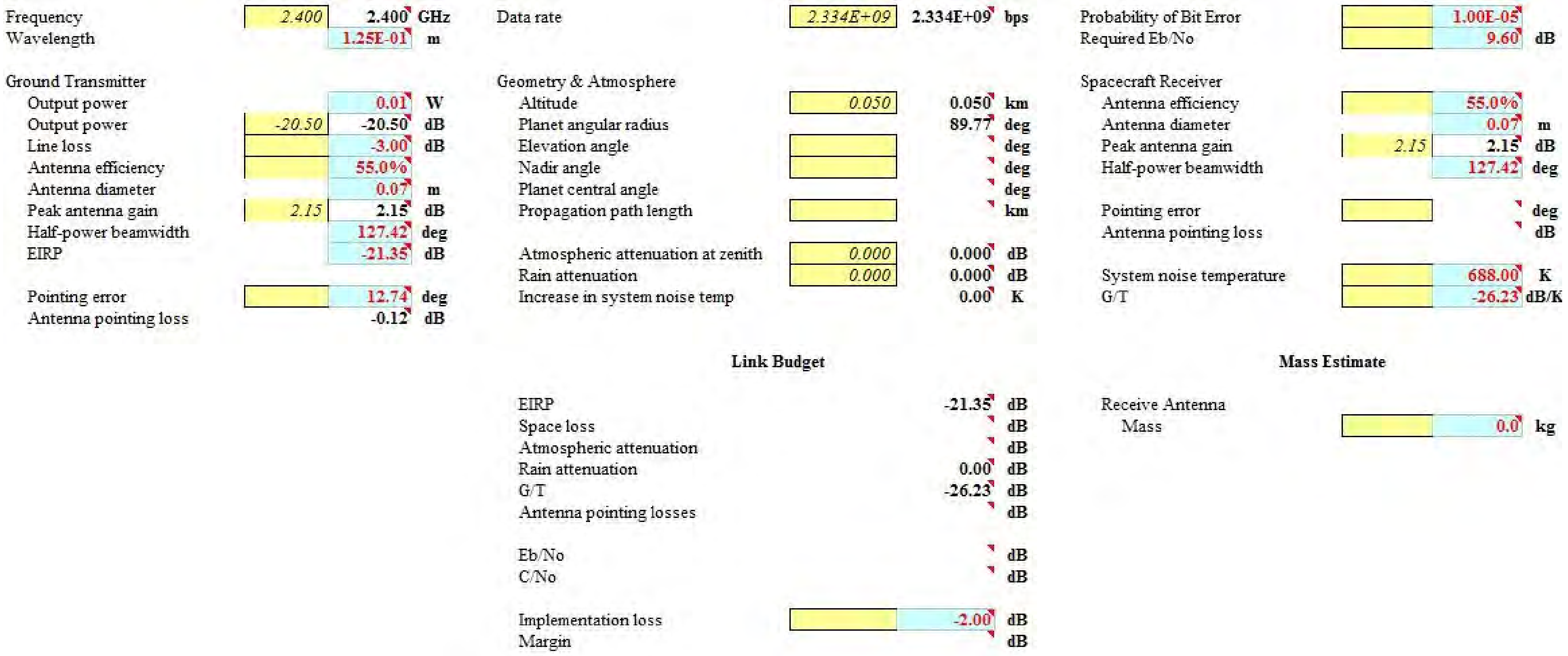
New Instrument			
Aperture diameter	*0.260*	0.260	m
Surface area imaged (per orbit)		996421.28	km^2
Number of "square" images (per orbit)		0.14	
Duty cycle (per orbit)	*0.920%*	0.920%	
Linear dimensions - scaled			
Side 1 (or cylinder length)		0.39	m
Side 2 (or cylinder diameter)		0.26	m
Side 3 (if necessary)			m
Volume		0.02	m^3
Mass		28.1	kg
Peak power		31.6	W
Average power		0.3	W

Payload - Data Rate

Number of pixels		1413813	
Bits per pixel		8.0	
Ground velocity		6762.1	m/s
Pixel ground resolution at nadir		1.000	m
Scan time per image (for staring sensor)	*10.00*	10.00	sec
Orbit period		98.77	min
Storage time (percent of orbit period)		100.0%	

Data storage - scanning sensor		4170071.5	Mb
Data production rate - scanning sensor		7.648E+10	bps
Data storage - staring sensor		87186889.2	Mb
Data production rate - staring sensor		1.599E+12	bps
Data storage - other sensor			Mb
Data production rate - other sensor			bps

Figure 9. Payload Physical Sizing



Frequency	2.400	2.400	GHz
Wavelength		1.25E-01	m
Ground Transmitter			
Output power		0.01	W
Output power	-20.50	-20.50	dB
Line loss		-3.00	dB
Antenna efficiency		55.0%	
Antenna diameter		0.07	m
Peak antenna gain	2.15	2.15	dB
Half-power beamwidth		127.42	deg
EIRP		-21.35	dB
Pointing error		12.74	deg
Antenna pointing loss		-0.12	dB

Data rate	2.334E+09	2.334E+09	bps
Geometry & Atmosphere			
Altitude	0.050	0.050	km
Planet angular radius		89.77	deg
Elevation angle			deg
Nadir angle			deg
Planet central angle			deg
Propagation path length			km
Atmospheric attenuation at zenith	0.000	0.000	dB
Rain attenuation	0.000	0.000	dB
Increase in system noise temp		0.00	K

Probability of Bit Error		1.00E-05	
Required Eb/No		9.60	dB
Spacecraft Receiver			
Antenna efficiency		55.0%	
Antenna diameter		0.07	m
Peak antenna gain	2.15	2.15	dB
Half-power beamwidth		127.42	deg
Pointing error			deg
Antenna pointing loss			dB
System noise temperature		688.00	K
G/T		-26.23	dB/K

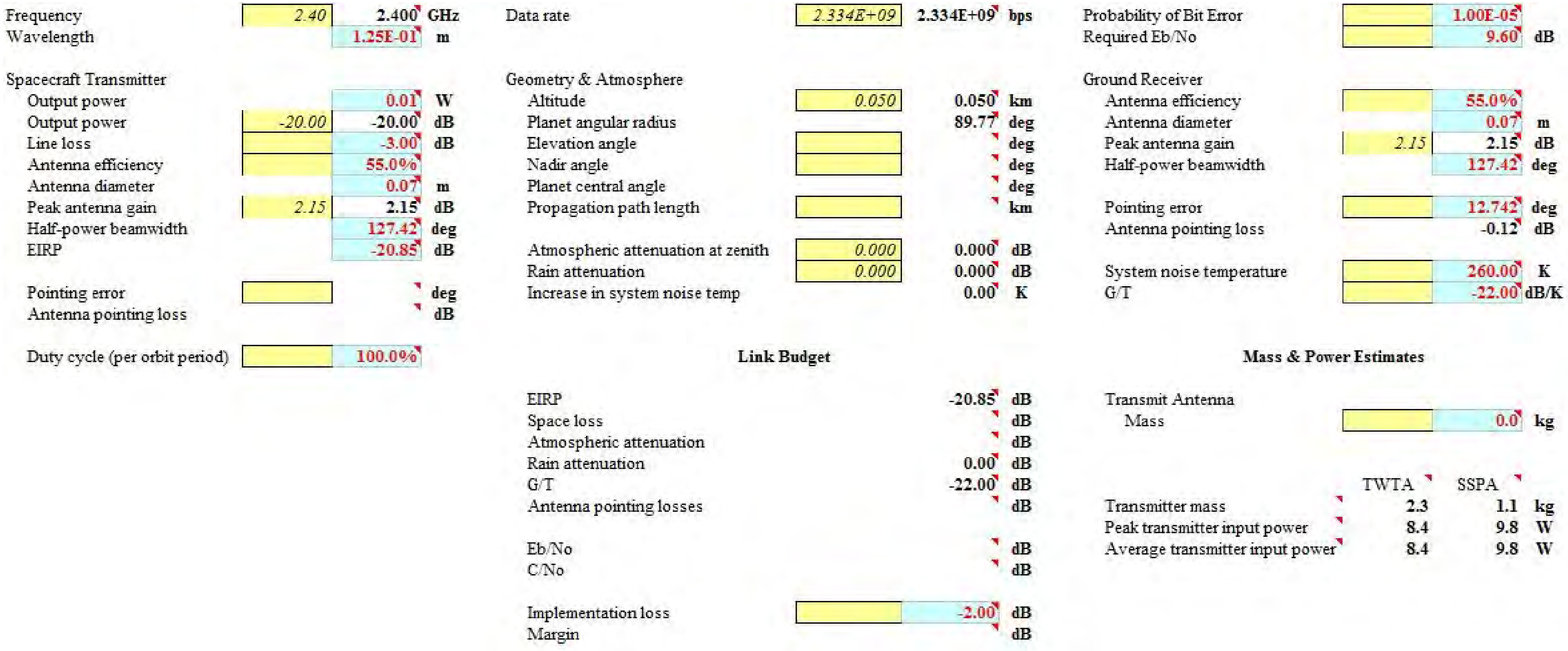
Link Budget

EIRP		-21.35	dB
Space loss			dB
Atmospheric attenuation			dB
Rain attenuation		0.00	dB
G/T		-26.23	dB
Antenna pointing losses			dB
Eb/No			dB
C/No			dB
Implementation loss		-2.00	dB
Margin			dB

Mass Estimate

Receive Antenna			
Mass		0.0	kg

Figure 10. Uplink Communications Inputs



Frequency	2.40	2.400	GHz
Wavelength		1.25E-01	m
Spacecraft Transmitter			
Output power		0.01	W
Output power	-20.00	-20.00	dB
Line loss		-3.00	dB
Antenna efficiency		55.0%	
Antenna diameter		0.07	m
Peak antenna gain	2.15	2.15	dB
Half-power beamwidth		127.42	deg
EIRP		-20.85	dB
Pointing error			deg
Antenna pointing loss			dB
Duty cycle (per orbit period)		100.0%	

Data rate	2.334E+09	2.334E+09	bps
Geometry & Atmosphere			
Altitude	0.050	0.050	km
Planet angular radius		89.77	deg
Elevation angle			deg
Nadir angle			deg
Planet central angle			deg
Propagation path length			km
Atmospheric attenuation at zenith	0.000	0.000	dB
Rain attenuation	0.000	0.000	dB
Increase in system noise temp		0.00	K

Probability of Bit Error		1.00E-05	
Required Eb/No		9.60	dB
Ground Receiver			
Antenna efficiency		55.0%	
Antenna diameter		0.07	m
Peak antenna gain	2.15	2.15	dB
Half-power beamwidth		127.42	deg
Pointing error		12.742	deg
Antenna pointing loss		-0.12	dB
System noise temperature		260.00	K
G/T		-22.00	dB/K

Link Budget

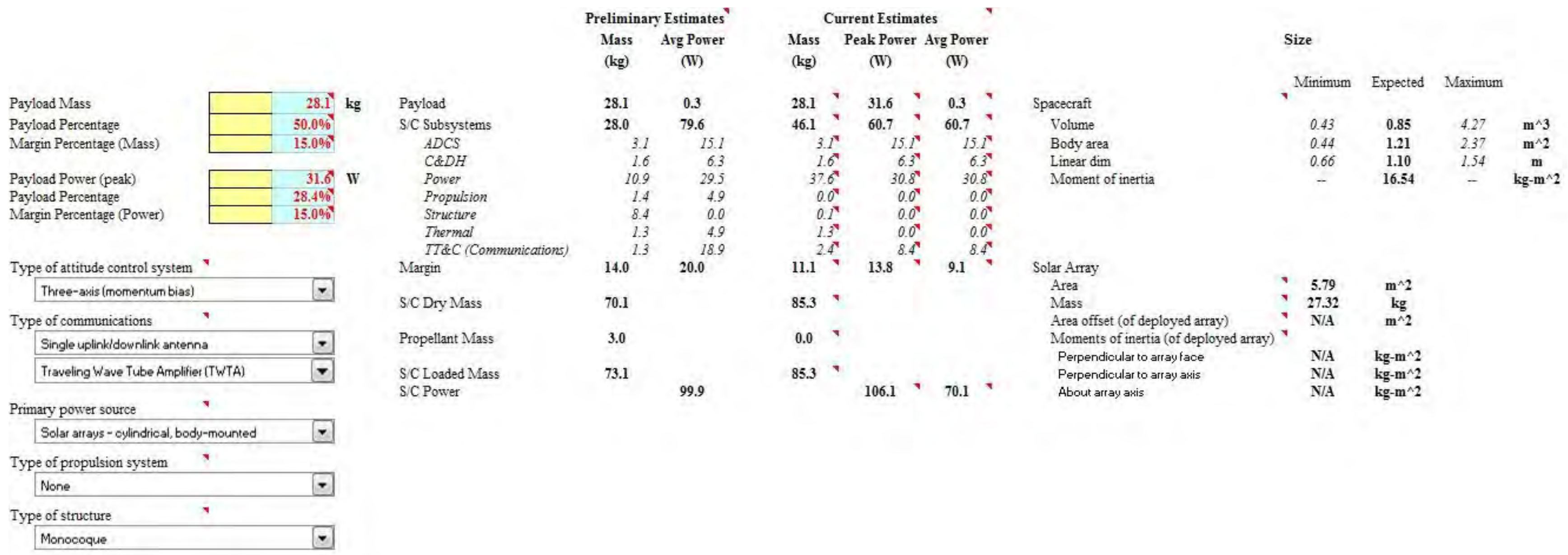
EIRP		-20.85	dB
Space loss			dB
Atmospheric attenuation			dB
Rain attenuation		0.00	dB
G/T		-22.00	dB
Antenna pointing losses			dB
Eb/No			dB
C/No			dB
Implementation loss		-2.00	dB
Margin			dB

Mass & Power Estimates

Transmit Antenna			
Mass		0.0	kg

	TWTA	SSPA	
Transmitter mass	2.3	1.1	kg
Peak transmitter input power	8.4	9.8	W
Average transmitter input power	8.4	9.8	W

Figure 11. Downlink Communications Input



Payload Mass	28.1	kg
Payload Percentage	50.0%	
Margin Percentage (Mass)	15.0%	
Payload Power (peak)	31.6	W
Payload Percentage	28.4%	
Margin Percentage (Power)	15.0%	

Type of attitude control system
Three-axis (momentum bias)

Type of communications
Single uplink/downlink antenna
Traveling Wave Tube Amplifier (TWTA)

Primary power source
Solar arrays - cylindrical, body-mounted

Type of propulsion system
None

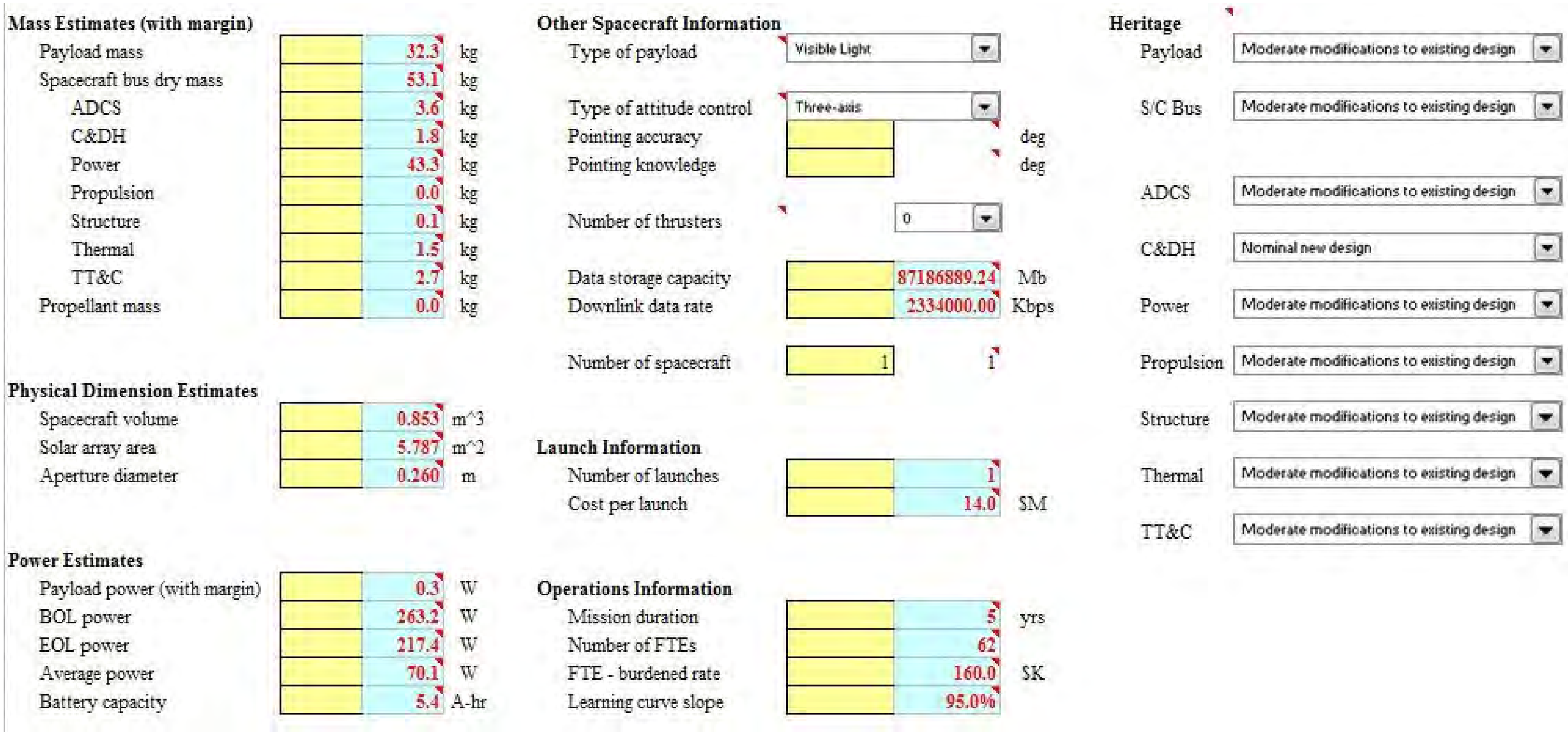
Type of structure
Monocoque

	Preliminary Estimates		Current Estimates		
	Mass (kg)	Avg Power (W)	Mass (kg)	Peak Power (W)	Avg Power (W)
Payload	**28.1**	**0.3**	**28.1**	**31.6**	**0.3**
S/C Subsystems	**28.0**	**79.6**	**46.1**	**60.7**	**60.7**
ADCS	*3.1*	*15.1*	*3.1*	*15.1*	*15.1*
C&DH	*1.6*	*6.3*	*1.6*	*6.3*	*6.3*
Power	*10.9*	*29.5*	*37.6*	*30.8*	*30.8*
Propulsion	*1.4*	*4.9*	*0.0*	*0.0*	*0.0*
Structure	*8.4*	*0.0*	*0.1*	*0.0*	*0.0*
Thermal	*1.3*	*4.9*	*1.3*	*0.0*	*0.0*
TT&C (Communications)	*1.3*	*18.9*	*2.4*	*8.4*	*8.4*
Margin	**14.0**	**20.0**	**11.1**	**13.8**	**9.1**
S/C Dry Mass	**70.1**		**85.3**		
Propellant Mass	**3.0**		**0.0**		
S/C Loaded Mass	**73.1**		**85.3**		
S/C Power		**99.9**		**106.1**	**70.1**

Size	Minimum	Expected	Maximum	
Spacecraft				
Volume	*0.43*	**0.85**	*4.27*	**m^3**
Body area	*0.44*	**1.21**	*2.37*	**m^2**
Linear dim	*0.66*	**1.10**	*1.54*	**m**
Moment of inertia	--	**16.54**	--	**kg-m^2**

Solar Array		
Area	**5.79**	**m^2**
Mass	**27.32**	**kg**
Area offset (of deployed array)	**N/A**	**m^2**
Moments of inertia (of deployed array)		
Perpendicular to array face	**N/A**	**kg-m^2**
Perpendicular to array axis	**N/A**	**kg-m^2**
About array axis	**N/A**	**kg-m^2**

Figure 12. Final Design Sizing Input



Mass Estimates (with margin)

Payload mass	32.3	kg
Spacecraft bus dry mass	53.1	kg
ADCS	3.6	kg
C&DH	1.8	kg
Power	43.3	kg
Propulsion	0.0	kg
Structure	0.1	kg
Thermal	1.5	kg
TT&C	2.7	kg
Propellant mass	0.0	kg

Physical Dimension Estimates

Spacecraft volume	0.853	m^3
Solar array area	5.787	m^2
Aperture diameter	0.260	m

Power Estimates

Payload power (with margin)	0.3	W
BOL power	263.2	W
EOL power	217.4	W
Average power	70.1	W
Battery capacity	5.4	A-hr

Other Spacecraft Information

Type of payload	Visible Light	
Type of attitude control	Three-axis	
Pointing accuracy		deg
Pointing knowledge		deg
Number of thrusters	0	
Data storage capacity	87186889.24	Mb
Downlink data rate	2334000.00	Kbps
Number of spacecraft	1	1

Launch Information

Number of launches	1	
Cost per launch	14.0	$M

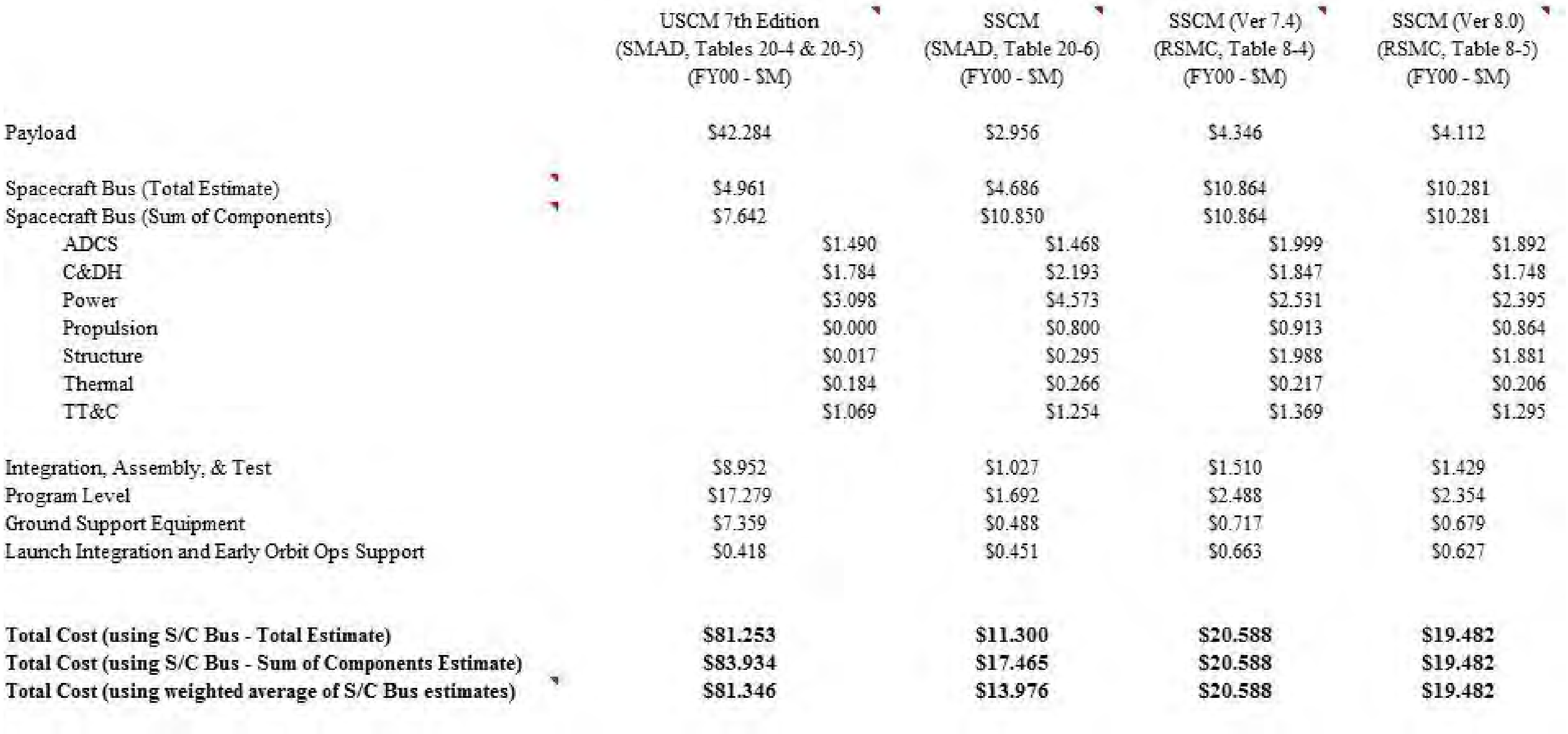
Operations Information

Mission duration	5	yrs
Number of FTEs	62	
FTE - burdened rate	160.0	$K
Learning curve slope	95.0%	

Heritage

Payload	Moderate modifications to existing design
S/C Bus	Moderate modifications to existing design
ADCS	Moderate modifications to existing design
C&DH	Nominal new design
Power	Moderate modifications to existing design
Propulsion	Moderate modifications to existing design
Structure	Moderate modifications to existing design
Thermal	Moderate modifications to existing design
TT&C	Moderate modifications to existing design

Figure 13. Cost Estimate Inputs



	USCM 7th Edition (SMAD, Tables 20-4 & 20-5) (FY00 - $M)		SSCM (SMAD, Table 20-6) (FY00 - $M)		SSCM (Ver 7.4) (RSMC, Table 8-4) (FY00 - $M)		SSCM (Ver 8.0) (RSMC, Table 8-5) (FY00 - $M)	
Payload	$42.284		$2.956		$4.346		$4.112	
Spacecraft Bus (Total Estimate)	$4.961		$4.686		$10.864		$10.281	
Spacecraft Bus (Sum of Components)	$7.642		$10.850		$10.864		$10.281	
ADCS		$1.490		$1.468		$1.999		$1.892
C&DH		$1.784		$2.193		$1.847		$1.748
Power		$3.098		$4.573		$2.531		$2.395
Propulsion		$0.000		$0.800		$0.913		$0.864
Structure		$0.017		$0.295		$1.988		$1.881
Thermal		$0.184		$0.266		$0.217		$0.206
TT&C		$1.069		$1.254		$1.369		$1.295
Integration, Assembly, & Test	$8.952		$1.027		$1.510		$1.429	
Program Level	$17.279		$1.692		$2.488		$2.354	
Ground Support Equipment	$7.359		$0.488		$0.717		$0.679	
Launch Integration and Early Orbit Ops Support	$0.418		$0.451		$0.663		$0.627	
Total Cost (using S/C Bus - Total Estimate)	**$81.253**		**$11.300**		**$20.588**		**$19.482**	
Total Cost (using S/C Bus - Sum of Components Estimate)	**$83.934**		**$17.465**		**$20.588**		**$19.482**	
Total Cost (using weighted average of S/C Bus estimates)	**$81.346**		**$13.976**		**$20.588**		**$19.482**	

Figure 14. Space Segment Cost Comparison

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