A Reliability Model for the Design and Optimization of Separated Spacecraft Interferometer Arrays

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Abstract. The concept of using identical spacecraft for space-based optical interferometry is introduced. The built-in redundancy of such a separated spacecraft interferometer (SSI) design not only improves the reliability of the system, but also improves system performance by placing the redundant components where they can be used during nominal operations. Five metrics have been developed to compare SSI designs. These include 1) total system reliability, 2) specific system reliability, 3) cost per image, 4) time to produce an image, and 5) reduced mission effectiveness due to partial system failure. The reliability model incorporates both combinatorial analysis and Markov modeling to evaluate different SSI designs on the basis of these five metrics. The results indicate that the modular and multifunctional spacecraft (MAMS/C) design rates higher than the single function spacecraft design (SFD) for all five metrics under the assumed mission parameters. These parameters include the number of small satellites in the array, the failure rate of the three components within the array, and the mission design life. For small arrays with extremely reliable components and short mission design lives, the current NASA SFD array with only one combiner suffices. This is because the intended design life of the system is shorter than the mean-time-to-failure of the system. For future larger arrays with more realistic component failure rates and longer mission design lives, designs that incorporate both collector and combiner functions on each small satellite bus rate higher. On the basis of these results, rules of thumb have been developed for the design and optimization of SSI small satellite arrays.

Introduction

In optical interferometry, starlight is reflected by at least two separate collector mirrors into the optics of a combiner in which the light is interfered. From the amplitude and phase information of the fringes for a number of different separations and orientations of collectors, a cross-correlation map is formed in the u-v plane. u and v are free variables in the image Fourier transform domain. By taking the inverse Fourier transform, a brightness map (image) of the observed object is formed in the x-y plane.

NASA has identified space based interferometry as a key technology for future space science programs, such as the Origins program, due to the ability of an interferometer to deliver resolutions orders of magnitude greater than those possible by reasonably sized single aperture telescopes. Original concepts for space based interferometers involved a single, monolithic spacecraft measuring tens to hundreds of meters in size [1]. The sole driver of this large size was to provide a long baseline between the optics.

More recently, NASA is now considering distributing the key elements of the interferometer, namely the collector and combiner optics, on a minimum of three separate, smaller spacecraft [2]. This configuration allows for longer baselines than possible with a single satellite. The small satellites only need to be large enough to support their optics payload. As technology progresses, one may envision future arrays of a dozen or more small satellites working synergistically to image extra-solar Earth-like planets. This paper develops a reliability model for separated spacecraft interferometer (SSI) small satellite arrays that may be used to compare and optimize various designs.

All SSI arrays contain three basic elements: collector mirrors that reflect the incoming light to the combiner[s], combiner optics that interfere the light, and small satellite buses to support the previous two elements. Every current SSI design - the NASA Deep Space 3 Interferometer (DS3) (3 satellites), the ESA Free-Flyer Interferometer (7 satellites), and the JPL MUSIC concept (17 satellites) - employs only one combiner satellite. This is defined as a single function design (SFD), as each spacecraft performs the function of either a collector or a combiner, but not both. Preliminary analysis indicates that this may not be the best design as the combiner satellite represents a single point failure. Alternative designs include providing more than one combiner satellite or placing both collector and combiner elements on each satellite bus. Such spacecraft that can serve as both a collector and a combiner are defined as modular and multifunctional spacecraft (MAMS/C).

The current design of DS3 is analogous to that of conventional ground based interferometers. At first, this appears to be a logical course of action as such a design is successful for ground based observatories. However, there are important distinctions between ground-based and space-based interferometers. On the ground, whenever a component fails and renders the system inoperable, a technician simply repairs the interferometer by replacing the faulty component. In space, one does not have this luxury. A failure of a single component in a single spacecraft in a three spacecraft array can yield the entire SFD interferometer useless, even if the other two spacecraft are working perfectly.

Thus, a space-based interferometer must be designed to be extremely reliable, robust, and adaptable to partial failures. It is well known that the best way to improve the reliability of a system is through redundancy - both within a spacecraft and between spacecraft. However, the amount of redundancy required for a series of interdependent spacecraft in the SFD to be robust to partial failures simply costs too much with today's conventional space system designs. The increased redundancy in the SFD would be implemented in one of two ways: 1) insert redundant components in the subsystems of each collector, combiner, and bus (mass and cost penalty) or 2) use components with a higher mean-time-to-failure (mttf) in the collector, combiner, and bus (cost penalty). Both methods increase reliability by improving redundancy or reliability within each spacecraft.

Modular and multifunctional spacecraft place the redundancy between spacecraft and hold the potential to deliver this needed redundancy economically. Specifically, MAMS/C can increase total system reliability while decreasing the required individual spacecraft's reliability and thus the per unit cost of each spacecraft. MAMS/C also improve performance by allowing the redundant components to be used during nominal operations.

First, the five metrics of total system reliability, specific system reliability, cost per image, time to produce an image, and reduced mission effectiveness are developed. Next, the Markov model methodology of analysis is presented. The five metrics and Markov analysis are applied to a specific case study of NASA's proposed three spacecraft Deep Space 3 interferometer. Finally, design rules of thumb are developed based on the preceding work.

Metric 1: Total System Reliability

Define system reliability as the probability of obtaining a fringe (R_{Fringe}) through a separated spacecraft interferometer. Further, divide the separated spacecraft interferometer system into its three core components - the collector, combiner, and bus. The collector component relays starlight to a combiner. The combiner component interferes the starlight, processes the information, and obtains a fringe measurement. Finally, every spacecraft in the system requires a bus, which performs all of the vital spacecraft functions such as attitude determination and control, communication, etc. Thus, the minimum functionality required to obtain an image is one combiner and two collectors on three separate spacecraft, each with their own functioning bus.

Both Figures 1 and 2 illustrate the current SFD three spacecraft design and the proposed MAMS/C design. Notice that each system is modeled as consisting of modular components.

In the Deep Space 3 SFD design, light is reflected from the two collector spacecraft into the optics of the combiner spacecraft, which resides in the plane parallel to the line-of-sight to the star [2]. Only one u-v point is generated for each baseline configuration.

In the MAMS/C design, each spacecraft is at the vertex of an equilateral triangle which is normal to the line-of-sight to the star. Each collector relays its light to the two other spacecraft combiners. The u-v points for three equal length, but different orientation baselines are acquired for each spacecraft array configuration.

The fact that three correlation measurements and thus three image elements can be obtained per orientation, if the three spacecraft are arranged in an equilateral triangle, is a major advantage of the MAMS/C design. In contrast, the Deep Space 3 design only takes one correlation measurement and image element per orientation. Thus, for an image with a given desired number of pixels, the MAMS/C design is three times more efficient than the current DS3 design.

In the current SFD configuration, all six components of the space system are in series (Figure 2), meaning that

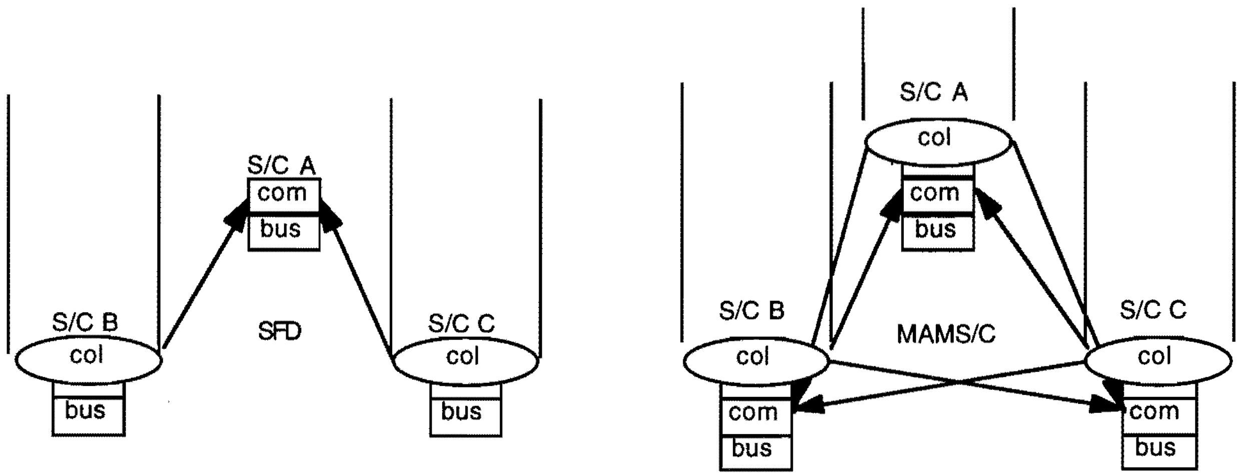

Figure 1: Deep Space 3 Single Function Spacecraft Design (SFD) and Modular and Multifunctional Spacecraft Design (MAMS/C)

if any one component fails, the entire system is useless. Thus, the probability of getting a fringe is the product of the reliability of each component in the system [3].

$$R_{Fringe}=R_{com}R_{col}^2R_{bus}^3 \qquad (1)$$

where R_{com} is the reliability of the combiner, R_{col} is the reliability of the collector, and R_{bus} is the reliability of the spacecraft bus.

The reliability of a component is formally defined as

$$R=e^{-\lambda t} \qquad (2)$$

where λ is the failure rate of the component and t is the time as measured from the beginning of the mission. With this notation, Equation 1 can be rewritten as

$$R_{Fringe} = e^{-(\lambda_{com}+2\lambda_{col}+3\lambda_{bus})t} \qquad (3)$$

Equations 4 through 6 derive an expression for the total system reliability of an SFD separated spacecraft interferometer as a function of any number, n, of spacecraft.

$$R_{Fringe}=R_{com}R_{bus}[1-(1-R_{c1}R_{c2})(1-R_{c2}R_{c3})(1-R_{c1}R_{c3})\ldots(1-R_{cn-1}R_{cn})] \qquad (4)$$

if

$$R_{c1}R_{c2}=R_{c2}R_{c3}=R_{c1}R_{c3}=R_{cn-1}R_{cn}=R_{col}^2R_{bus}^2 \qquad (5)$$

then

$$R_{Fringe}=R_{com}R_{bus}[1-(1-R_{col}^2R_{bus}^2)^{(0.5)(n-1)(n-2)}] \qquad (6)$$

By taking the limit of Equation 6 as n approaches infinity for single function arrays with large numbers of collector spacecraft, R_{Fringe}~$R_{com}R_{bus}$. This is because the combiner spacecraft represents a single point failure for any sized single function spacecraft array.

Figures 1 and 2 also illustrate the functionality of a parallel design with MAMS/C. In this design, each spacecraft contains both a combiner and collector component as well as the required spacecraft bus. The system can lose one collector or two combiners in a specific combination and still achieve the mission.

The probability of obtaining a fringe with this parallel system architecture is one minus the product of the complements of obtaining a fringe through each combiner.

$$R_{Fringe}=1-(1-R_{com}\,R_{bus}(R_{col}R_{bus})^2)^3 \qquad (7)$$

or

$$R_{Fringe} = 1-(1-e^{-(\lambda_{com}+\lambda_{bus})t}(e^{-2(\lambda_{col}+\lambda_{bus})t}))^3 \qquad (8)$$

Equations 9-11 derive an expression for the total system reliability of a MAMS/C array as a function of the number, n, of spacecraft in the array.

$$R_{Fringe}=1-(1-R_{fringe})^n \qquad (9)$$

if

$$R_{fringe}=R_{com}R_{bus}(1-(1-R_{col}^2R_{bus}^2)^{(0.5)(n-1)(n-2)}) \qquad (10)$$

then

$$R_{Fringe}=1-[1-R_{com}R_{bus})(1-(1-R_{col}^2R_{bus}^2)^{(0.5)(n-1)(n-2)})]^n \qquad (11)$$

In this case, taking the limit as n approaches infinity for MAMS/C arrays with large numbers of spacecraft results in $R_{Fringe}=1$.

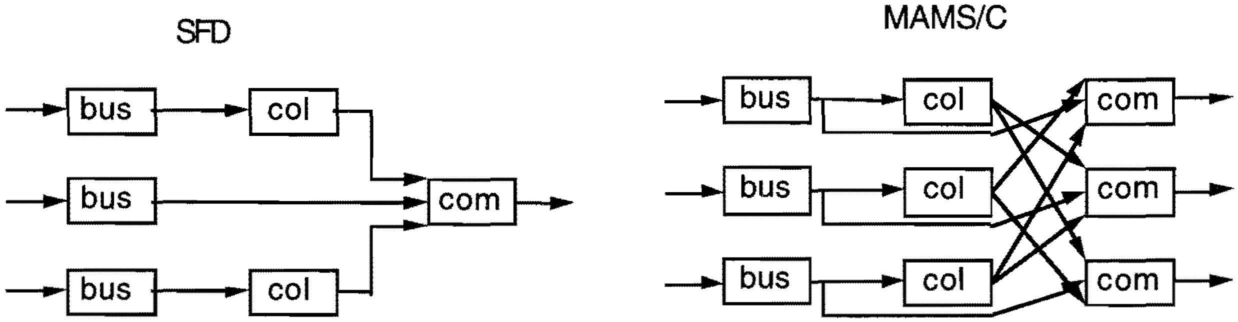

Figure 2: Functionality of the SFD (all components in series) and the MAMS/C Design (some components in parallel)

Now consider some reasonable values for the reliability of components. If the reliability of each component is 0.9 at a given time t, the probability of obtaining a fringe is 0.897 with the MAMS/C design, which is much higher than the 0.531 probability of obtaining the same fringe with the SFD spacecraft. Thus, parallel design with MAMS/C yields superior system performance in an array for any given number of spacecraft. This is true for any component reliability between 0.3 and 0.99 as illustrated in Figure 3. If the metric of performance is total system reliability, then MAMS/C always yields a superior design due to its use of redundancy.

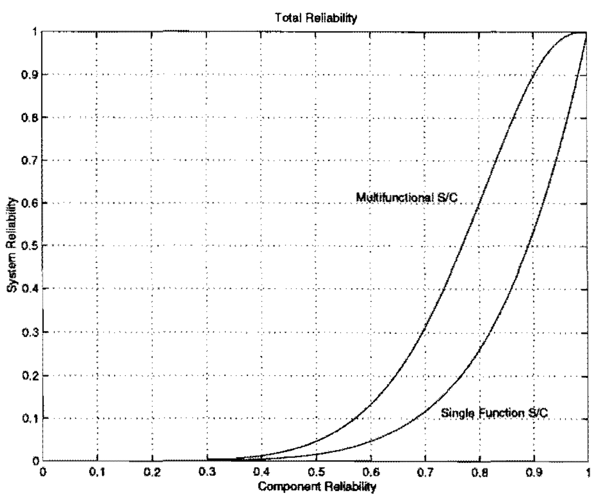

Figure 3: Component Reliability vs. Total System Reliability (R_{col}=R_{com}=R_{bus})

Metric 2: Specific Reliability

Total system reliability may not be the best metric for judgment, however, because it does not tell the entire story. Increased system reliability through redundancy is conventionally accompanied by increased cost. In this case, the cost is the total system mass that must be built and then launched from Earth. A given increase in R_{Fringe} may not be practical if it requires a prohibitively large increase in the mass of the system, such that the system can no longer be constructed under budget or launched on the vehicle of choice. In order to make a more realistic comparison, a relationship between R_{Fringe} and the accompanying mass is needed.

Define specific reliability as the reliability per unit mass of the system. Using the numbers from a preliminary mass budget in a NASA paper on the proposed design for DS3 [4], normalized values for the mass of each component were developed (Table 1).

Table 1: Normalized Mass Values

S/C Component	Estimated Mass (kg)	Normalized Mass Value
Collector	25	1
Combiner	50	2
Bus	150	6

Figure 4 plots component reliability vs. specific reliability for a three spacecraft interferometer. As one can see, the specific reliability of the MAMS/C design remains greater than that for conventional SFD spacecraft for components with a reliability between 0.3-0.97, beyond which the marginal increase in the reliability of the MAMS/C system is no longer worth the accompanying increase in its mass. The exact location of this transition point varies depending on the reliability of the components in the array. Thus, the specific reliability metric allows one to identify which separated spacecraft interferometer architecture makes the most sense at a given time t in the mission for a given set of component mttf's.

Another way to evaluate the relative merit of a distributed satellite system architecture is by determining the required spacecraft reliability for a given total system reliability [5].

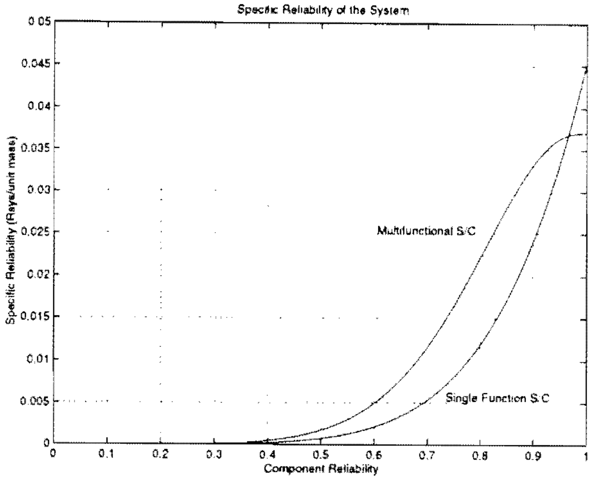

Figure 4: Component Reliability vs. Specific Reliability ($R_{col}=R_{com}=R_{bus}$)

Define spacecraft reliability as the complement of the probability of failure for a single spacecraft. Now allow the following simplification from the previous model:

$$R_{com}R_{bus}=R_{col}R_{bus}=R_{com}R_{col}R_{bus}=R_{s/c} \quad (12)$$

The expressions in Equations 6 and 11 for obtaining a fringe through each system architecture can then simplified as follows:

SFD: $$R_{Fringe}=R_{s/c}[1-(1-R_{s/c}^2)^{(0.5)(n-1)(n-2)}] \quad (13)$$

MAMS/C: $$R_{Fringe}=1-[1-R_{s/c}(1-(1-R_{s/c}^2)^{(0.5)(n-1)(n-2)})]^n \quad (14)$$

where n is the total number of spacecraft in the array. Figure 5 plots the required individual spacecraft reliability vs. the number of spacecraft in the array for a desired R_{Fringe}=0.95 for both architectures.

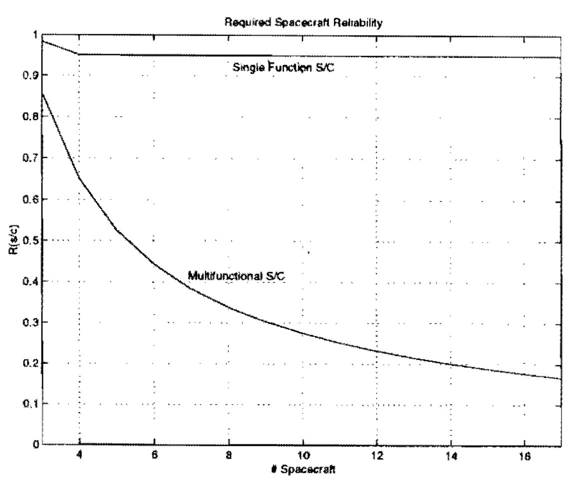

Figure 5: Required Spacecraft Reliability for a Given System Reliability (R_{Fringe}=0.95)

As one can see, for a desired system reliability of R_{Fringe}=0.95 in a three spacecraft array, the required spacecraft reliability is higher in the SFD design ($R_{S/C}$=0.983) than in the MAMS/C design ($R_{S/C}$=0.858). Further, when the array contains four or more spacecraft, the combiner spacecraft in the SFD array must still be built to a 0.95 reliability because it represents a single point failure. The collector spacecraft in the same array may be built to a lower reliability. The necessary individual reliability for MAMS/C, on the other hand, continually decreases as n increases because there are no single point failures in the design.

<u>**Metric 3: Cost Per Image**</u>

One of the three key metrics in the evaluation of a distributed satellite system is the cost per function metric [6]. For the case of an optical interferometer, the final desired function is the production of astronomical images. Each image requires many fringe measurements whose number is approximately equivalent to the number of pixels in the image. Thus, the cost per function (ϕ) for a separated spacecraft interferometer may be defined as

$$\phi=\frac{\text{Total Mission Cost (\$M)}}{\text{Total Number of Images}} \quad (15)$$

Total mission cost includes the costs for project management, system engineering, science teams, interferometer optical instrumentation, spacecraft buses, integration and testing, mission operations, and the launch vehicle [7]. The additional costs for the MAMS/C system over the SFD system manifest themselves as the cost of one extra collector, two extra combiners, three larger spacecraft buses, and the potential additional expense of using a larger launch vehicle. A learning curve 0f 92% was assumed in the construction of the three identical MAMS/C buses.

These additional expenses for the MAMS/C design were estimated based on a cost budget for Deep Space 3 presented at a costing workshop [7] and are listed in Table 2. Originally, the proposed Med-Lite rocket was targeted as the launch vehicle of choice for DS3. The Med-Lite then became the Delta-Lite, which is now the Delta 73xx series. The MAMS/C design contains approximately 200-300 kg more mass than SFD design (depending upon the assumed payload mass fraction), and thus requires use of the next higher version of the Delta rocket, which is the Delta II 7920. The Delta II 7320 costs $40M per launch and the Delta II 7920 costs $50M [8], resulting in an additional expense of $10M for the MAMS/C design.

Table 2: Additional Mission Costs Required for the MAMS/C Design

Item	Additional Cost to Mission ($Million)
1 Extra Collector	4.3
2 Extra Combiners	11.3
3 Larger Buses	20.4
More Powerful Launch Vehicle	10.0
Total	**47.0**

As explained in the next section, a MAMS/C array can produce an image of a given size in less time than a SFD array. Based on the assumptions in Table 3, the number of images obtainable during the mission design life were calculated for each design and are also listed in Table 3.

Table 4 lists the cost per image for DS3, the SFD design, and the MAMS/C design. As one can see, the MAMS/C design yields the lowest cost per image and thus provides the greatest value to NASA and the scientific community. This value is due to the ability of the MAMS/C system to collect more images in a given amount of time and also be functionally capable of collecting images over a longer mission design life due to the inherent redundancy of the system. In other words, the marginal increase in the performance of the MAMS/C system more than outweighs the marginal increase in mission cost incurred by using modular and multifunctional spacecraft over single function spacecraft.

Table 4: Cost Per Image

Design	Cost Per Image ($ Million)
DS3	2.16
SFD	1.18
MAMS/C	0.698

Metric 4: Time To Produce an Image

Another measure of performance of a separated spacecraft interferometer is the amount of time required to produce an image. Each baseline and angular orientation corresponds to a point in the u-v plane, which in turn corresponds to a single pixel in the image. The more baselines per configuration, the fewer reconfigurations of the spacecraft in the array are required to produce an image with a given number of pixels. This in turn decreases the amount of propellant and time to produce an individual image and increases the total number of objects that may be viewed within a given mission lifetime. The number of baselines (#BL) in a separated spacecraft interferometer scales with the number of collectors as

$$\#\,\mathrm{BL} = \frac{n_{col}(n_{col}-1)}{2} \tag{16}$$

where n_{col} represents the number of unfailed collectors in the array.

For a given number of spacecraft, the MAMS/C array has one more collector than the SFD array. Figure 6 compares the number of array reconfigurations required

Table 3: Assumptions and Calculations

Assumption/ Calculation	DS3 Design	Single Function S/C Design	MAMS/C Design
Integration Time [9]	2 min./pixel	2 min./pixel	2 min./pixel
Maneuvering Time	1 min./reconfiguration	1 min./reconfiguration	1 min./reconfiguration
Retargeting Time	4 hours	4 hours	4 hours
Image Size	32x32=1024 pixels	32x32=1024 pixels	32x32=1024 pixels
Bus mttf	96 months	96 months	96 months
Collector mttf	180 months	180 months	180 months
Combiner mttf	48 months	48 months	48 months
Number Baselines/ Configuration	1	1	3
Number Reconfig./ Image	1032	1032	344
Total Time Per Image	53.2 hr	53.2 hr	40.1 hr
Mission Design Life	*6 months	**11 months	**17.5 months
Total # Images	**81**	**148**	**314**

* As designated by NASA.
** As determined by the time at which the probability of failure of the system exceeds 50%.

to produce a 1024 pixel (32x32) image. As one can see, the most pronounced improvements in performance between the two designs occur for smaller arrays with only 3-5 spacecraft. As time passes, individual spacecraft within an array of a given size will fail, n_{col} will decrease, and mission effectiveness measured as the number of baselines available per configuration will degrade.

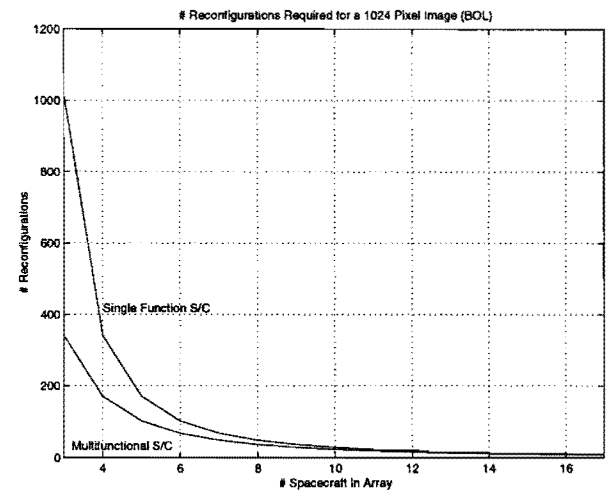

Figure 6: Number of Reconfigurations to Produce an Image (1024 pixels) at t=0

Metric 5: Reduced Mission Effectiveness

Reduced mission effectiveness refers to the ability of a design to continue to perform even when certain components within the design have failed. The technique of Markov modeling was used to evaluate the reduced mission effectiveness of each device. Additionally, two cases were run for each design Markov model. In the first case, the failure rate for each of the three array components (bus, collector, and combiner) were assumed to be identical with a mean-time-to-failure (mttf) of 100 months (8.33 yr.), which corresponds to a failure rate (λ) of 0.01 month^{-1}.

In the second case, more realistic values, taking into account the varying complexities and heritage of the three model components, were estimated. Satellite bus designs have continued to improve over the past 30 plus years. The mttf for commercial communication satellites is typically quoted as 100,000 hr (11.4 yr.) [10]. For this science mission, assuming enough fuel is on board for 8 years worth of maneuvering, let $mttf_{bus}$=8yr=70,080hr. This corresponds to a failure rate for the bus of $1.427\times10^{-5}hr^{-1}$=0.01042month^{-1}.

The collector is the simplest of the three model components, containing only a flat 12 cm 3-axis gimbaled mirror. Thus, the mttf for the collectors should be much higher. Let $mttf_{collector}$=15yr= 131,400hr. This corresponds to a failure rate for the collector of $7.610\times10^{-6}hr^{-1}$=0.005555 month^{-1}.

The combiner represents a newer, high risk space technology with more moving parts. Thus, its reliability will be lower. Let $mttf_{com}$=4yr=35,040hr. This corresponds to a failure rate for the combiner of 2.8539hr^{-1}=0.02083month^{-1}. Table 5 summarizes the mttf and failure rate values for the two cases.

Table 5: Failure Rate Values

Case 1	mttf(months)	λ (month^{-1})
Bus	100	0.01
Collector	100	0.01
Combiner	100	0.01
Case 2		
Bus	96	0.01042
Collector	180	0.005555
Combiner	48	0.02083

Markov Model Methodology

Markov models are used over combinatorial analysis when modeling events that are time dependent and sequential (ie. mutually exclusive) in a system [11].

The two cases studied are the current three single function spacecraft DS3 design and the proposed three modular and multifunctional spacecraft design. First, fault tree diagrams illustrating all the possible different modes of failure for each design were developed (Figures 7 and 8).

In the SFD DS3 design, as illustrated in Figure 7, the system fails when any single component - collector, combiner, or bus - fails. On the other hand, individual components or even combinations of components can fail without leading to system failure in the MAMS/C design. Figure 8 illustrates which component failure combinations will lead to system failure for the MAMS/C design.

From the fault trees, a Markov model illustrating each possible state of the system was created for the two designs. Figure 9 illustrates the Markov Model for the SFD. This model contains only one possible functioning state, which occurs when all six components are working. Otherwise, the system is in a state of failure - no fringes can be measured and no image can be produced. On the other hand, the Markov model for the MAMS/C design was considerably more complicated with 43 mutually exclusive possible states.

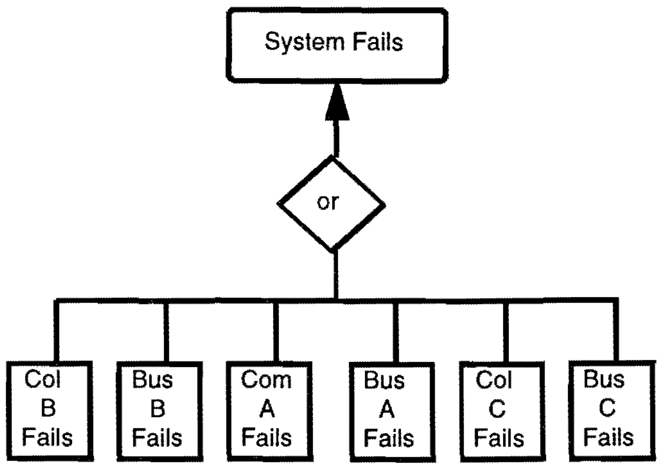

Figure 7: Fault Tree for the Three Single Function Spacecraft Design

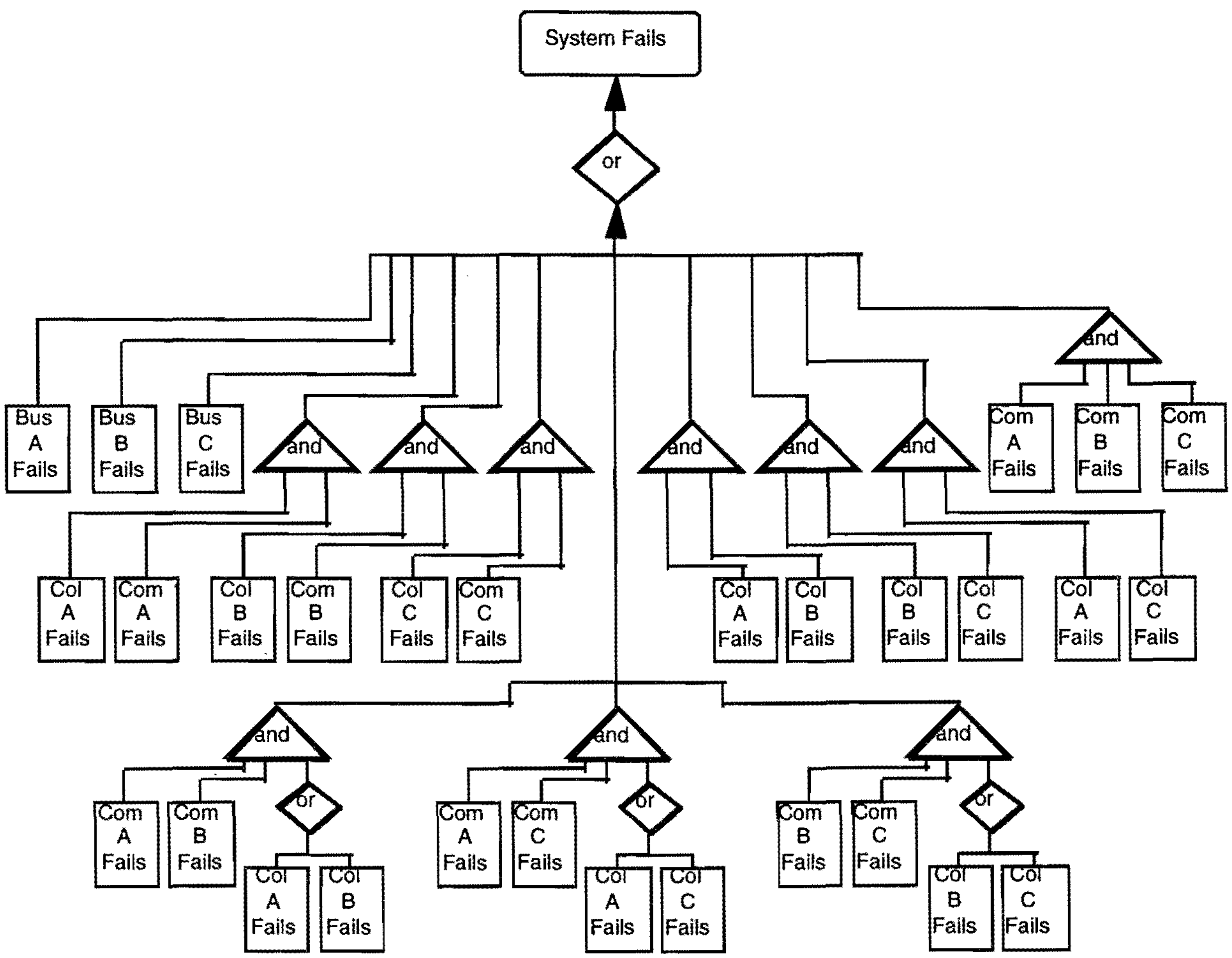

Figure 8: Fault Tree for the Three Modular and Multifunctional Spacecraft Design

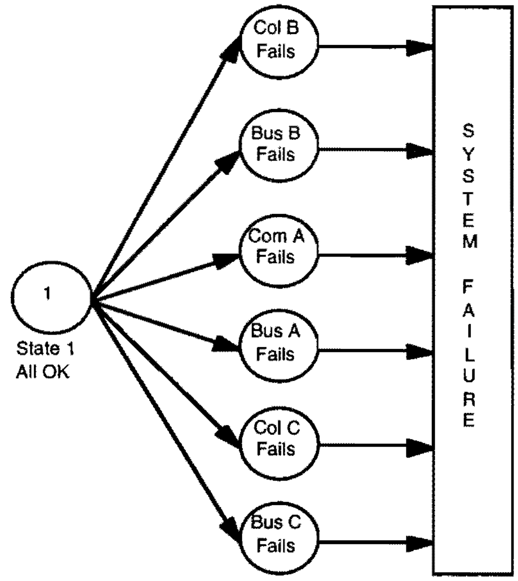

Figure 9: Markov Model of the Three Single Function S/C DS3 Design

From the Markov Model diagrams, a system of differential equations can be written to determine the most probable state of the system at any given time. The SFD system can be described by only one linear first order differential equation. The MAMS/C system requires a set of 43 partially coupled linear first order differential equations to model the system.

Upon solving the system of equations for the MAMS/C design, the model may be simplified from 43 states to the 6 functioning states listed in Table 6 in which the array may still produce an image.

Table 6: Aggregated Functioning States for MAMS/C Model

State #	Functioning State
1	Everything is working.
2	One collector has failed.
3	One combiner has failed.
4	Two combiners have failed.
5	One collector and one combiner have failed.
6	One collector and two combiners have failed.

Markov Model Results

Figures 10-13 illustrate the results for each design in cases 1 and 2. In these figures, system failure is defined as occurring when the probability of failure exceeds 0.5.

In Case 1, all three elements - the spacecraft bus, collector, and combiner - are assumed to have the same failure rate of 0.01 month^{-1}. From the dashed line in Figure 10, one can see that the SFD design is able to produce images for at least 1 year, after which the probability of failure exceeds the probability that the single function spacecraft system is still working. This exceeds the mission design life of DS3 as designated by NASA by six months.

In the MAMS/C design, the system remains in state 1 for approximately 18 months (solid line), after which the probability of failure exceeds the probability that the system is still capable of producing an image, which is the sum of the probabilities of states 1 through 6 (dash-dot lines).

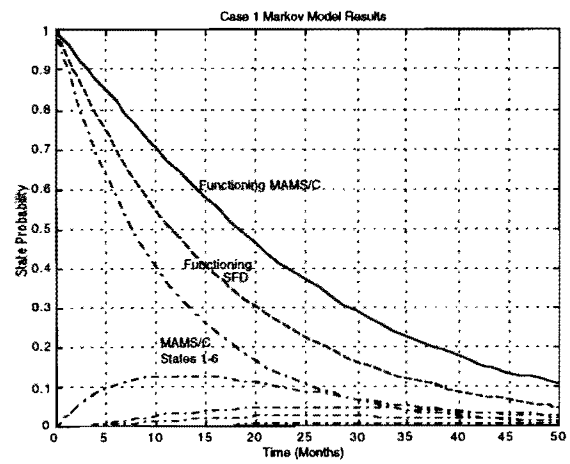

Figure 10: Case 1 - λ_{bus}=0.01, λ_{col}=0.01, λ_{com}=0.01

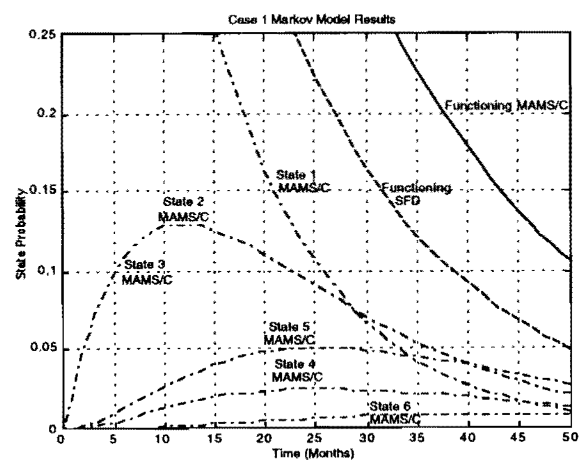

Figure 11: Case 1 Magnified - λ_{bus}=0.01, λ_{col}=0.01, λ_{com}=0.01

In Case 2, more realistic values for the failure rate of each component are used, taking into account the varying complexities and heritage for each component. As illustrated in Figure 12, the period of time for which the probability of success exceeds the probability of failure for the SFD has been reduced to 11 months, but is still greater than the DS3 mission design life of six months. In the MAMS/C design, the system transitions from state 1 to state 3 at approximately 13 months (Figure 13). At t=17.5 months, the probability of failure exceeds the probability that the system is still functioning.

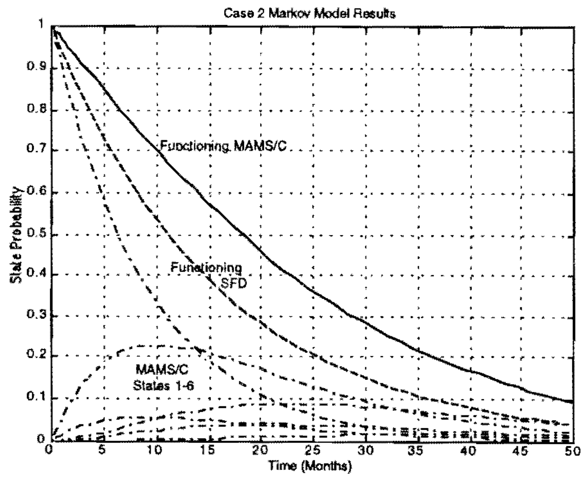

Figure 12: Case 2 - λ_{bus}=0.01042, λ_{col}=0.005555, λ_{com}=0.02083

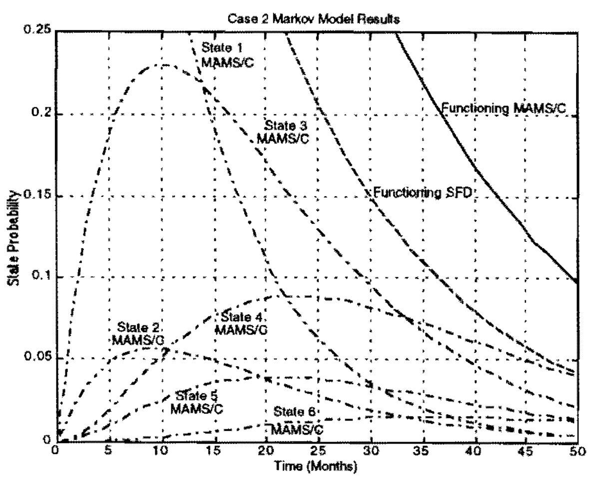

Figure 13: Case 2 Magnified - λ_{bus}=0.01042, λ_{col}=0.005555, λ_{com}=0.02083

Conclusions Based Upon Markov Model Analysis

The Markov model analysis results illustrate the key trends that may be used in the design of small satellite arrays. When the mission design life is short, as is the case for DS3 (only 6 months), both designs meet the mission requirement of operating at the end of 6 months. The MAMS/C will have a higher probability of functioning at this time at the price that it was more costly both to build and to launch.

However, for missions with longer design lives, the MAMS/C design offers more significant advantages over the single function design, namely a higher probability of obtaining images over a longer period of time due to the ability of a MAMS/C array to deal with partial failures and still function at a reduced level of performance. This in turn leads to a lower cost per image.

Many future science missions with small satellite arrays will have much longer design lives than DS3. For example, the Terrestrial Planet Finder (TPF), a possible NASA 5 spacecraft interferometer array that would be placed in a 5 AU heliocentric orbit, would have a minimum mission design life of 5 years (60 months) [1]. Thus, MAMS/C designs have the potential to be better than conventional single function spacecraft designs for future small satellite arrays conducting missions over long design lives. Mission lifetime determines the best design choice for a given mission.

DS3 Case Study Results

Table 7 lists the results of each metric for the two proposed DS3 designs. In the first case, the failure rates for each of the three components in the array model - the bus, collector, and combiner - were assumed to be identical with a mean-time-to-failure of 100 months (λ=0.01 month^{-1}). In the second case, more realistic values taking into account the varying complexities and heritage of the three model components were estimated.

In both cases, the total system reliability of the MAMS/C design was approximately 33% better than the single function design at the end of the mission design life of six months. In other words, the odds of having the capability to still collect images at t=6 months increase by a third when the MAMS/C configuration is used. The MAMS/C design will always have a higher total system reliability for DS3 due to component redundancy.

Initially (near t=0), the marginal increase in reliability provided by the MAMS/C design is not worth the marginal increase in mass it creates. However, a transition occurs beyond which the MAMS/C design provides a higher specific reliability than the single function design. For DS3, this transition point occurs before the mission design life of six months.

Table 7: DS3 Case Study Results

Metric	Case 1 SFD	Case 1 MAMS/C	Case 2 SFD	Case 2 MAMS/C
Total System Reliability (t=6 months)	0.698	0.972	0.684	0.969
Specific System Reliability (t=6 months)	0.0317	0.0360	0.0311	0.0359
Cost per Image ($M)	1.08	0.68	1.18	0.70
Time to Produce an Image (number reconfigurations for a 1024 pixel image)	1024	342	1024	342
Reduced Mission Effectiveness (50% prob. of mission termination) (months)	12	18	11	17

In both cases, the cost per image was less for the MAMS/C design than the SFD design. This means that the marginal increase in performance provided by MAMS/C far outweighs the marginal increase in cost if the mission objective is to produce images.

The single function design provides only one baseline per configuration, while the MAMS/C design provides three. This translates into 1024 reconfigurations for the SFD array and only 342 reconfigurations for the MAMS/C array for a 32x32 pixel image. Thus, MAMS/C arrays can produce an image in less time and with fewer thruster firings than SFD arrays.

Finally, the ability of the MAMS/C design to deal with partial failures gives the MAMS/C system greater reduced mission effectiveness than the single function spacecraft design. This results in a longer mission design life for the MAMS/C array than the SFD spacecraft array.

Design Rules of Thumb for a 3 Spacecraft Array

Rule 1
For a given number of spacecraft with given values for the mttf of components, a constellation consisting entirely of MAMS/C will always have a higher total system reliability than a constellation consisting solely of SFD spacecraft.

This rule flows naturally from the derivation of the formulas for total system reliability and is due to the built in redundancy of the MAMS/C array.

Rule 2
Beyond the transition point where the specific reliability of the MAMS/C array is less than the specific reliability of the SFD array ($Rsp_{MAMS/C} < Rsp_{SFD}$), the MAMS/C design should not be used.
Under these conditions, the cost of building and launching the extra mass inherent in the MAMS/C design is not worth the marginal increase in reliability the design provides.

Rule 3
For a given set of mission parameters, the best compromise between performance and economy over the mission design life is the system with the lowest cost per image.

This is the design which delivers the greatest "bang per buck." In this sense, MAMS/C designs have an advantage in that they provide a longer mission design life in which more images may be taken while simultaneously decreasing the time required to produce each image because more fringes are obtained and thus pixels recorded per configuration - all at a marginal dollar and mass cost.

Rule 4
If system performance is the chief concern, then the MAMS/C design should always be chosen.

For a given number of spacecraft, a MAMS/C array always provides more baselines than a single function design array. Thus, the MAMS/C array can produce an image with a given number of pixels with fewer array reconfigurations than a single function design. This saves both time and fuel, simultaneously allowing the MAMS/C design to take more images in a given time period and increase its fuel-constrained operational life. MAMS/C not only provide redundancy, but do so in a way that also improves nominal performance.

Rule 5
If the mission design life (MDL) is greater than the first state transition (MDL>1st state transition) in the Markov model, then MAMS/C are not needed.

One of the greatest advantages of MAMS/C is their ability to still function under partial failures, what is termed as "reduced mission effectiveness." However, if the mission design life is so short and the reliability of the components so high that no component failures occur before the end of the intended mission design life, then the MAMS/C design does not provide reduced mission effectiveness. The final decision of the array design then depends on the initial system requirements and the trade between performance and cost.

Conclusions

As outlined in this paper, modular and multifunctional spacecraft (MAMS/C) possess the potential to increase total system reliability and specific system reliability while simultaneously decreasing the required reliability of each individual spacecraft in an array and the cost per image. A MAMS/C design will always have a higher total system reliability than a single function spacecraft design. MAMS/C systems also provide more baselines for a given number of spacecraft sized array, allowing an image to be taken in a shorter period of time. Finally, MAMS/C arrays always provide greater reduced mission effectiveness than SFD arrays for missions with long design lives. Thus, depending on the requirements of the array, MAMS/C will likely be a better choice than single function spacecraft for future separated spacecraft interferometers. Also, hybrid designs containing both MAMS/C and single function design spacecraft need to be investigated in the future as they may rate higher for the five given metrics with a lower total system mass and cost.

As part of the New Millennium Program, the primary objective of the DS3 mission is the technology demonstration of formation flying and optical interferometry in space, with the secondary objective being the performance of some unique science. Thus, MAMS/C may not be needed for the technology demonstration DS3 mission, but will be extremely useful for future dedicated space science interferometry missions.

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