MISSION COST REDUCTION

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I. INTRODUCTION

The United States Air Force has historically performed the ICBM launch detection mission using satellites placed in high altitude orbits. This type of mission can also be performed at other altitudes while still maintaining overall mission performance. The choice of satellite altitude drives the size of the constellation, the characteristics of the sensor system and the supporting electronics, and the associated launch costs. The objective of this paper is to demonstrate to a first order that the ICBM launch detection mission can be performed at a lower altitude for a lower overall cost using a constellation of smaller less capable satellites while still maintaining overall mission performance.

Generically, the derivation of space mission costs begins with mission definition and understanding the requirements to perform that mission. Once understood, satellite constellations are designed that fit the mission needs. These constellations vary in size by altitude and coverage requirements. Next, the mission payload is designed dependent on mission requirements and constellation configuration. Supporting electronics and hardware are then included and the impact of their reliability considered. Finally, launch costs required to orbit the satellite constellations are determined. Integration of all of these variables will result in a system cost versus altitude table from which the lowest cost satellite constellation can be determined.

The mission evaluated here is that of ballistic missile launch detection. This system's mission is to detect ballistic missile launches from anywhere on the globe within seconds of cloud break. To perform this mission a satellite constellation must: (1) have uninterrupted global coverage; (2) have a sensor system capable of detecting missile exhaust plumes in a timely manner; (3) use electronics capable of processing the gathered information; and, (4) function reliably over its design life. The rest of this paper is devoted to finding the design solutions that meet these requirements at the lowest cost.

II. CONSTELLATION DESIGN

The use of circular polar orbits is one method that provides uninterrupted global coverage. Authors such as Luders [1], Beste [2], and Rider [3] are noted for their work in identifying techniques to design efficient circular polar orbit satellite constellations.

The techniques proposed by these authors to minimize the number of satellites for a particular level of coverage at a particular altitude are similar. In general, satellites are first placed in an orbital plane such that there are no gaps in coverage between individual satellites. Then multiple orbital planes are placed around the globe and phased with respect to each other to provide the desired coverage. The desired coverage to meet our mission requirement is uninterrupted single global coverage. Uninterrupted single global coverage is defined as having any point on the globe being visible to at least one satellite in the constellation at all times.

Rider's method for determining a minimum constellation to perform uninterrupted single global coverage can be reduced to the following equation (1) subject to restrictions (2) and (3).

$$T = ps \tag{1}$$

In equation (1) T is the total number of satellites, p is the number of satellite rings and s is the number of satellites in a ring. Equation (1) is subject to the following restrictions (2) and (3).

$$c - \cos^{-1}\left(\frac{\cos\Theta}{\cos \Pi/_s}\right) = 0 \tag{2}$$

$$\sin c - \cos\left[\left(\frac{p-1}{2p}\right)\Pi\right] = 0 \tag{3}$$

In equation (2) Θ is the central half angle of a circle on the surface of the Earth that encloses an area capable of being viewed by a satellite. The value c is a distance on either side of the orbit trace defined by satellites in the same orbit ring.

The angle Θ is given by equation (4).

$$\cos(\Theta + \beta) = \left(\frac{\cos\beta}{1 + \frac{h}{R_E}} \right) \tag{4}$$

In equation (4) the value β is the Earth grazing angle; h is the satellite altitude; and R_E is the Earth's radius.

The results of using Rider's method to determine a minimum satellite constellation to perform uninterrupted single global coverage are shown in Table 1. These results reflect a 10° grazing angle, which means that a satellite must be at least 10° above a ground point's local horizon to see it.

TABLE 1.
Constellation Altitude Versus Size
For Uninterrupted Global Single Coverage

Altitude (Kilometers)	Number	Altitude (Kilometers)	Number
500	128	6,000	12
600	104	7,000	10
700	84	8,000	10
800	72	9,000	8
900	66	10,000	8
1,000	54	15,000	8
1,100	50	20,000	6
1,200	45	25,000	6
2,000	28	30,000	6
3,000	18	35,000	6
4,000	15	35,900	6
5,000	15		

III. SENSORS

Ballistic missile launch detection can be performed using short wave infra-red (SWIR) sensors. The design of such sensors can be accomplished to a first order using range equations developed by Hudson [4]. Equation (5) is the idealized range equation for a search type sensor.

$$R_o\,(search) = \left[\frac{\Pi}{2}\, D_o\ (NA)\ D^{*} J\ \tau_a\ \tau_o\right]^{1/2} \left[\frac{\nu c}{\dot{\Omega}}\right]^{1/4} \qquad (5)$$

D_o = Diameter of Optic Entrance Aperture
NA = Numerical Aperture
D^* = Detector D - Star
J = Spectral Radient Intensity of the Target
τ_a = Spectral Transmittance of Atmosphere
τ_o = Spectral Transmittance of Optics
υ = Pulse Visibility Factor
c = Number of Detectors
$\dot{\Omega}$ = Time Rate of Search

For each satellite altitude shown in Table 1, a sensor design based on Hudson's search sensor range equations (5) was determined. Each altitude dependent design varied the number of detectors and the diameter of the optics entrance aperture to vary sensor range, while keeping other variables constant. The variables that were kept constant independent of altitude were the optical design, the detector material D^*, the spectral radiant intensity of the target, and the time to scan the search field. Some of the characteristics common to the sensor design are shown in Table 2. The derived sensor designs were not intended to stress technology but to provide a minimum capability to assure mission performance.

TABLE 2.
SWIR Sensor Characteristics

Characteristic	Description
Optical Design	Schmidt - Cassegrainian, 5 optical elements
Detector Material D*	5×10^{11} cm $(Hz)^{1/2}$ W^{-1}
Target	80,000 W/sr at 2.7 microns
Search Field Scan Rate	60 seconds

Sensor system costs and weights were derived using the cost estimating relationships of the Space Sensor Cost Model [5]. These cost estimating relationships use major sensor design variables to make average unit production cost estimates. System weight is also calculated. Table 3 shows the relationship between satellite altitude and sensor weight and cost in 1990 dollars.

TABLE 3.
Constellation Altitude Versus SWIR Sensor Weight and Cost

Altitude (Kilometers)	Weight (Kilograms)	Cost (1,000s)	Altitude (Kilometers)	Weight (Kilograms)	Cost (1,000s)
500	0.8	6,800	6,000	2.8	10,000
600	0.9	7,142	7,000	4.9	10,908
700	1.0	7,377	8,000	7.0	11,248
800	1.1	7,565	9,000	10.8	11,654
900	1.2	7,753	10,000	15.3	12,123
1,000	1.3	7,906	15,000	24.8	15,927
1,100	1.3	8,050	20,000	54.7	23,831
1,200	1.4	8,187	25,000	198.3	31,642
2,000	1.9	8,959	30,000	423.1	44,639
3,000	2.2	9,581	35,000	748.3	64,694
4,000	2.5	9,979	35,900	816.1	69,486
5,000	2.7	10,279			

IV. ELECTRONICS

Once the optics have gathered the SWIR radiation and it is detected by the detector elements on the focal plane, a signal processor and computer must be used to evaluate the gathered information. This analysis set a requirement for approximately 10,000 operations per second per detector element for the signal processing function. Also required were VLSIC technology and a digital pre-processor. The computer was sized for 10^7 operations per second to handle both sensor and generic satellite functions. Additionally, 2 x 10^7 bytes of memory were allocated for the computer. As with the signal processor, the computer uses VLSIC technology and has a digital pre-processor.

Weight and Cost

Electronics weights and costs were calculated by assessing the on-board computer and the signal processor separately. This was done since the computer requirements remain stable over all altitudes due to fixed housekeeping and communications functions. The signal processor weight and cost, however, scale linearly with the number of detectors in the SWIR sensor, and thus with altitude.

Both weight and cost data were derived from the Space Sensor Cost Model. The model was used to determine weight and cost values for the signal processor and computer, built with S-class components, for an SWIR surveillance satellite in geosynchronous orbit. This data is shown below in Table 4 along with fixed values for the Signal Processor, with cost in 1990 dollars.

TABLE 4.
Baseline Electronics Weight and Cost

Description	Weight (Kilograms)	Cost (1,000s)
Computer	48.6	2,402
Signal Processor (fixed)	9.05	7,307

Cost data for electronic subsystems built with components of other than S-class is calculated by multiplying baseline S-class costs by the relative cost factors in Table 5 [6].

TABLE 5.
Relative Cost Factors

Class	Relative Cost
S	1.00
S-1	0.68
B	0.1525
B-1	0.10375
B-2	0.0636
D	0.055
D-1	0.035

Reliability

If identical electronic subsystems are built with components of varying quality and varying cost their reliability must be evaluated. A reliability trade study must be carried out to quantify the performance lost due to building satellites with lower cost, lower quality components. A representative mission critical device, monolithic bipolar MOS integrated circuit, is chosen as the basis for the analysis.

The MIL-HDBK-217 [7] failure rate model for this type of device is shown in equation (6).

$$\lambda = \Pi_Q \, \Pi_L \, \left[C_1 \, \Pi_T \, \Pi_V + C_2 \, \Pi_E \right] \qquad (6)$$

Π_Q = Quality Factor
Π_L = Learning Factor
Π_T = Temperature Acceleration Factor
Π_V = Voltage Derating Stress Factor
Π_E = Environment Factor
C_1 = Complexity Factor Based on Gate or Bit Count
C_2 = Complexity Factor Based on Packaging

A failure rate of 4.65×10^{-9} failures/hr is typical for an S-class chip. By holding design and environmental factors constant, the device failure rates for different component classes can be reduced to a simple multiple of the component class quality factor (Π_Q) as shown in equation (7).

$$\lambda = \Pi_Q K \qquad (7)$$

Using S-class data, K = 1.86×10^{-8} which yields the failure rates for the component classes given in Table 6.

TABLE 6.
Failure Rates of Different Component Class Devices

Class	Π_Q	Failures Per Hour ($\times 10^{-9}$)
S	0.25	4.65
S-1	0.75	13.95
B	1.0	18.6
B-1	2.0	37.2
B-2	5.0	93.0
D	10.0	186.0
D-1	20.0	372.0

For this study, a subsystem by subsystem reliability analysis was not conducted, but reliability goals for S-class electronic subsystems were set which allowed the resulting system reliabilities for different component classes to be determined. The spacecraft electronics are assumed to be black boxes for which reliability is a function of component reliabilities and their internal circuit design. The "design" can be fixed by setting a reliability goal for black boxes comprising S-class components only. As a worst case, a linear circuit design is assumed, allowing the failure rate for a black box to be represented as in equation (8).

$$\lambda_X = \Pi_{Q_x} \sum_{i=1}^{n} K_i \tag{8}$$

λ_X = Failure Rate with Class x Component
Π_{Q_X} = Class x Quality Factor
K_i = Reduced K Factor, equation (7)
n = Number of Components

Since,

$$R_x = e^{-\lambda_x t} = e^{-t\,\Pi_{Q_x} \sum_{i=1}^{n} K_i} \tag{9}$$

then for two component classes, x and y, the black box reliabilities are related by equation (10).

$$\frac{\ln R_x}{\ln R_y} = \frac{-t\ \Pi_{Q_x} \sum_{i=1}^{n} K_i}{-t\ \Pi_{Q_y} \sum_{i=1}^{n} K_i} = \frac{\Pi_{Q_x}}{\Pi_{Q_y}} \tag{10}$$

To determine electronics reliability versus component class, a five year reliability goal for S-class black boxes is set at 0.995. Using equation (10) the following electronic system reliabilities are calculated.

TABLE 7.
Component Class Versus Five Year Reliability

Class	Five Year Reliability	
S	0.995	(goal)
S-1	0.985	
B	0.980	
B-1	0.961	
B-2	0.906	
D	0.820	
D-1	0.673	

V. SATELLITE AND CONSTELLATION RELIABILITY

The minimum constellation size to perform the mission, determined by Rider's method, must be adjusted to account for satellite losses over the life of the system. With reliabilities of the mission critical electronics known, reliability goals for the overall constellation can be set to determine numbers of satellites required to assure mission performance. Two assumptions implicit in this analysis are that; (1) failure of mission critical electronics results in failure of the satellite; and (2) all satellites in the constellation are launched during the first year of operation, i.e., no replenishment.

The five year system reliability goal was set at 0.995. This means that the probability that the satellite constellation is functioning in five years must be greater than or equal to 0.995. The number of satellites necessary to meet this requirement can be found by the binomial reliability function [8] shown in equation (11).

$$R_{sys} = 1 - \sum_{i=0}^{m-1} \binom{n}{i} R^i \, (1 - R)^{n-i} \qquad (11)$$

R_{sys} = System Reliability
m = Minimum Number of Units Required to be Operating at End of Time Period
n = Number of Units Initially in System
R = Unit Reliability for Time Period

The results of equation (11) applied to constellations using various component classes are shown in Table 8. The minimum number of satellites for proper coverage are listed versus altitude on the left column. Component classes are listed on the top. The body of the table shows the number of satellites using a given component class which must be initially launched to meet the mission performance and reliability goals.

TABLE 8.
Number of Satellites Required for Mission Performance
Initial # Satellites Required for System Reliability 0.995

Alt	# Req'd for Coverage	Class S	S-1	B	B-1	B-2	D	D-1
500	128	131	134	135	140	152	172	215
600	104	107	110	111	114	125	141	179
700	84	87	89	90	93	102	116	146
800	72	75	76	77	80	88	100	127
900	66	68	70	71	74	81	92	117
1000	54	56	58	59	62	67	77	98
1100	50	52	54	54	57	62	72	91
1200	45	47	49	49	51	56	65	83
2000	28	30	31	31	33	36	42	55
3000	18	19	20	21	22	24	29	38
4000	15	16	17	17	18	21	25	32
5000	15	16	17	17	18	21	25	32
6000	12	13	14	14	15	17	20	27
7000	10	11	12	12	13	15	18	23
8000	10	11	12	12	13	15	18	23
9000	8	9	10	10	11	12	15	20
10000	8	9	10	10	11	12	15	20
15000	8	9	10	10	11	12	15	20
20000	6	7	7	8	8	10	12	16
25000	6	7	7	8	8	10	12	16
30000	6	7	7	8	8	10	12	16
35000	6	7	7	8	8	10	12	16
35900	6	7	7	8	8	10	12	16

An alternative to constellation proliferation to maintain system reliability is to use redundant electronics inside individual satellites. Using S-class reliability as a baseline, the number of redundant lower reliability electronics black boxes can be calculated to meet that baseline value. Equation (12) shows the relationship between S-class black box reliability and the number of lower reliability black boxes required to meet S-class component reliability.

$$R\,(S\!-\!class\,) \;=\; 1 - (\,1 - R_x\,)^n \tag{12}$$

Rx = Reliability of class x black box

n = Number of redundant black boxes

The number of redundant black boxes needed to match S-class reliability were calculated and are shown in Table 9.

TABLE 9.
Number of Redundant Black Boxes

Class	Black Boxes
S	1
S - 1	2
B	2
B - 1	2
B - 2	3
D	4
D - 1	5

A satellite constellation using internal redundancy to raise individual satellite reliability to the S-class level will need only the S-class number of satellites (Table 8.) to meet mission performance goals.

VI. CONSTELLATION WEIGHTS AND COSTS

Total weights and costs of the candidate constellations can now be determined by combining the required number of satellites for each component class versus altitude, with electronics and sensor weights and costs. Persistent trends are obvious in the data, including the preliminary results presented here. It should be noted that the results are summations of vehicle weights and costs for given constellations, not values for single satellites.

Weights

Constellation sensor weights are a function of the number of satellites required in a constellation to meet mission performance and individual sensor weight. Constellation sensor weights for proliferated constellations are shown in Figure 1. S, B, and D component classes are shown since they provide maximum and minimum bounds for each altitude. Minima occur for all component classes at 7,000 Km altitude with the weight trend flat for altitudes down to about 3,000 Km. The absolute minimum occurs for S-class.

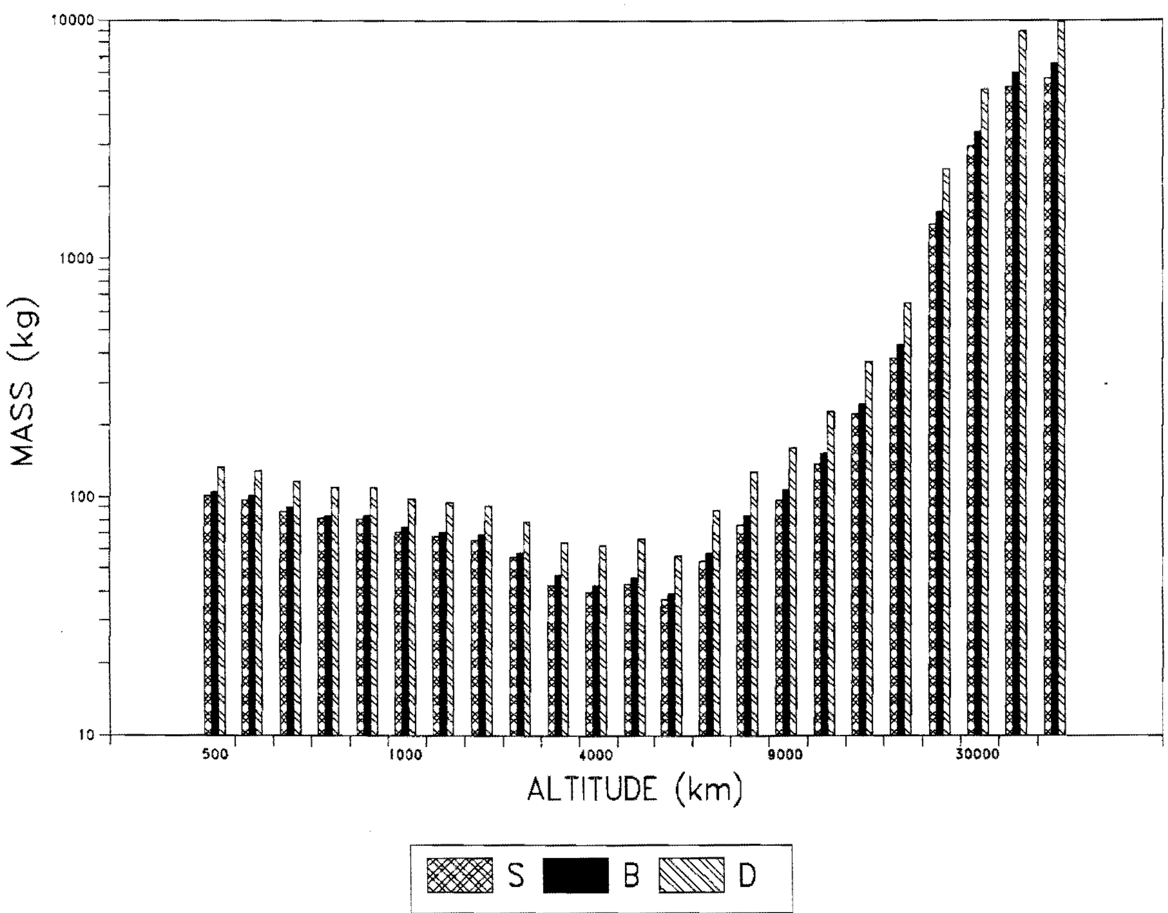

Figure 1. Constellation Sensor Weight (Proliferated)

Constellation Sensor weight for constellations using black box redundancy instead of proliferation are shown in Figure 2. There is no variation in weight for a particular altitude since different component class constellation will have the same number of satellites (and hence sensors) as an S-class constellation. As in Figure 1, minima occur at 7,000 Km altitude with the weight trend relatively flat for altitudes down to about 3,000 Km.

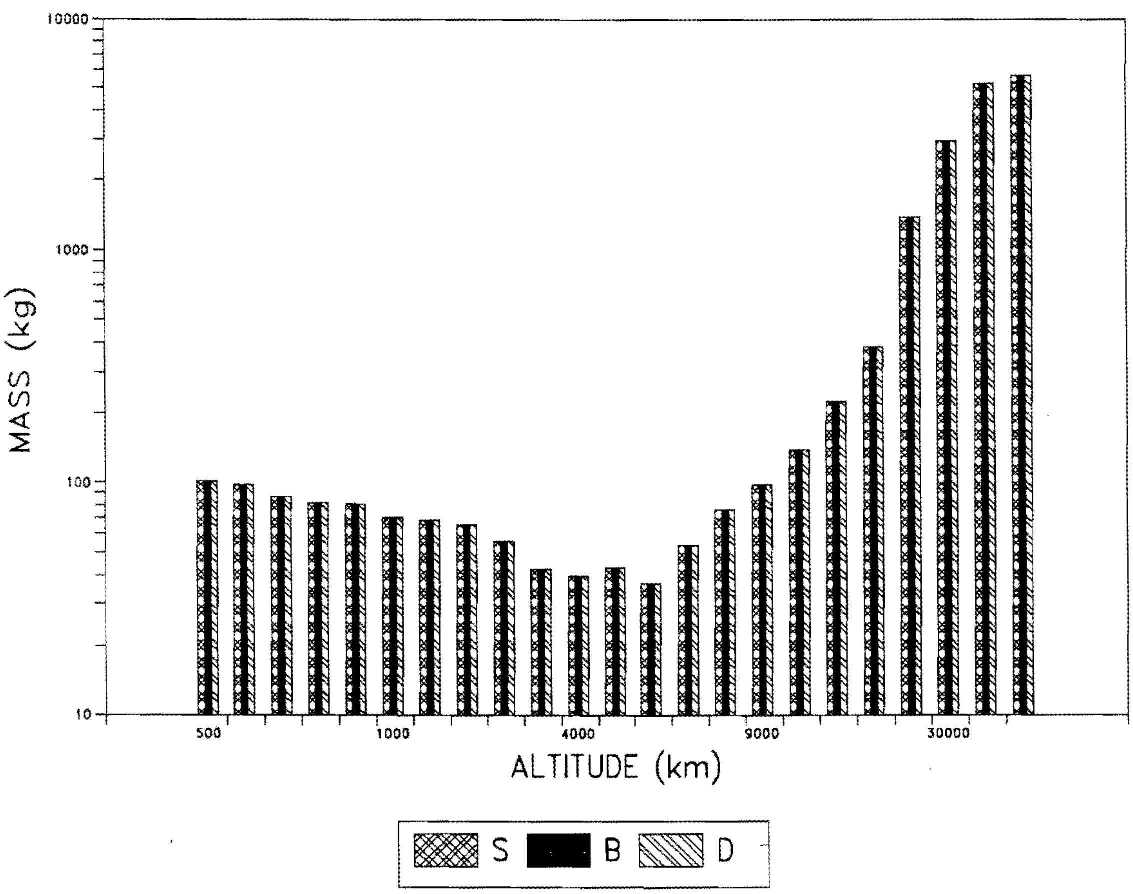

Figure 2. Const. Sensor Weights (Black Box Redundancy)

Constellation electronics weights are a function of the number of satellites required in a constellation to meet mission performance and individual electronics weight. Constellation electronics weights for proliferated constellations are shown in Figure 3. As the weights tend to decrease with altitude, there is a slight knee in the trends around 3,000 to 4,000 Km. The curves are relatively flat farther out in altitude, with all minima occurring at 20,000 Km. Again, the lowest constellation weight occurs for satellites with S-class components.

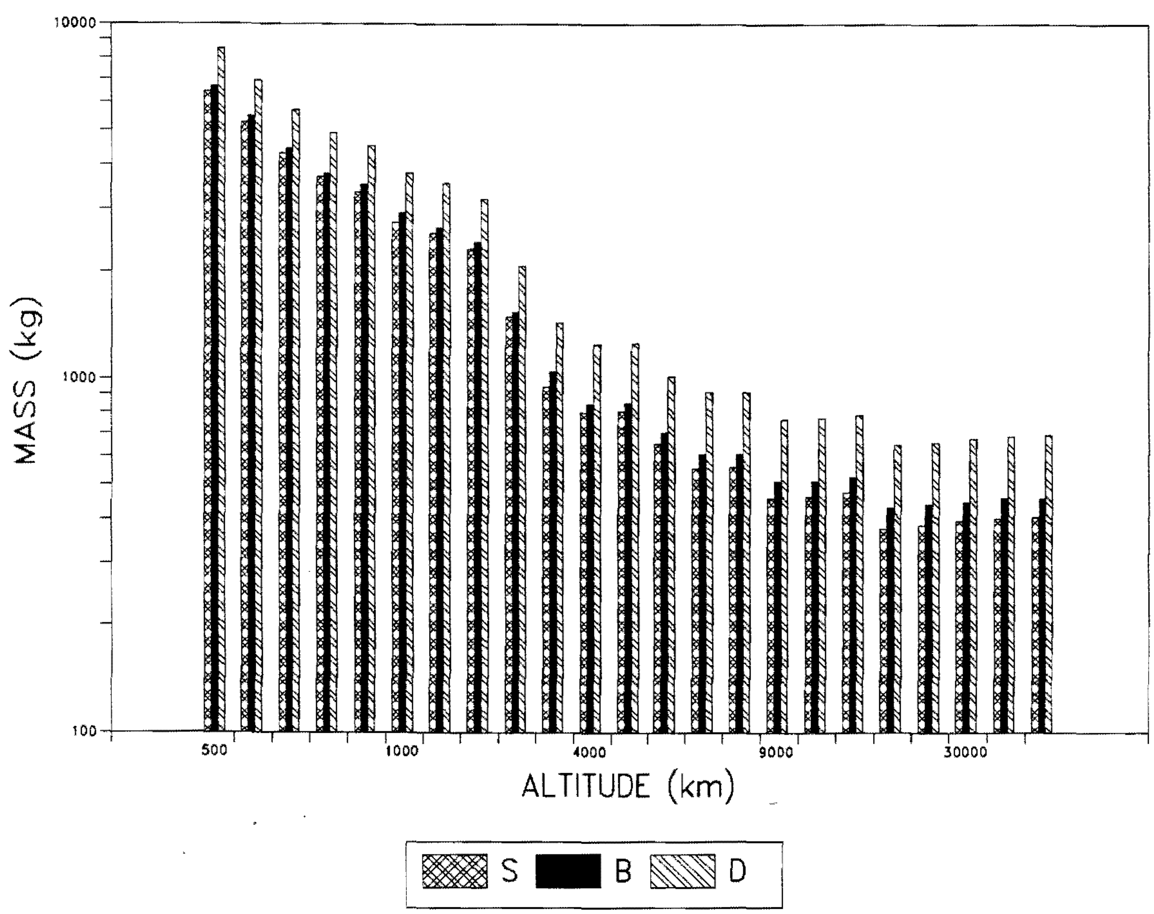

Figure 3. Const. Electronics Weights (Proliferated)

Constellation electronics weights for constellations using black box redundancy are shown in Figure 4. The weight of S-class electronics for each altitude is identical to those in Figure 3. The weight of B and D-class electronics are more however. This is due to the requirements of 2 and 4 black boxes on each satellite for B and D-class respectively. The knee seen in Figure 3 can still be seen in Figure 4. Again, minima occur at about 20,000 Km for the S-class constellations.

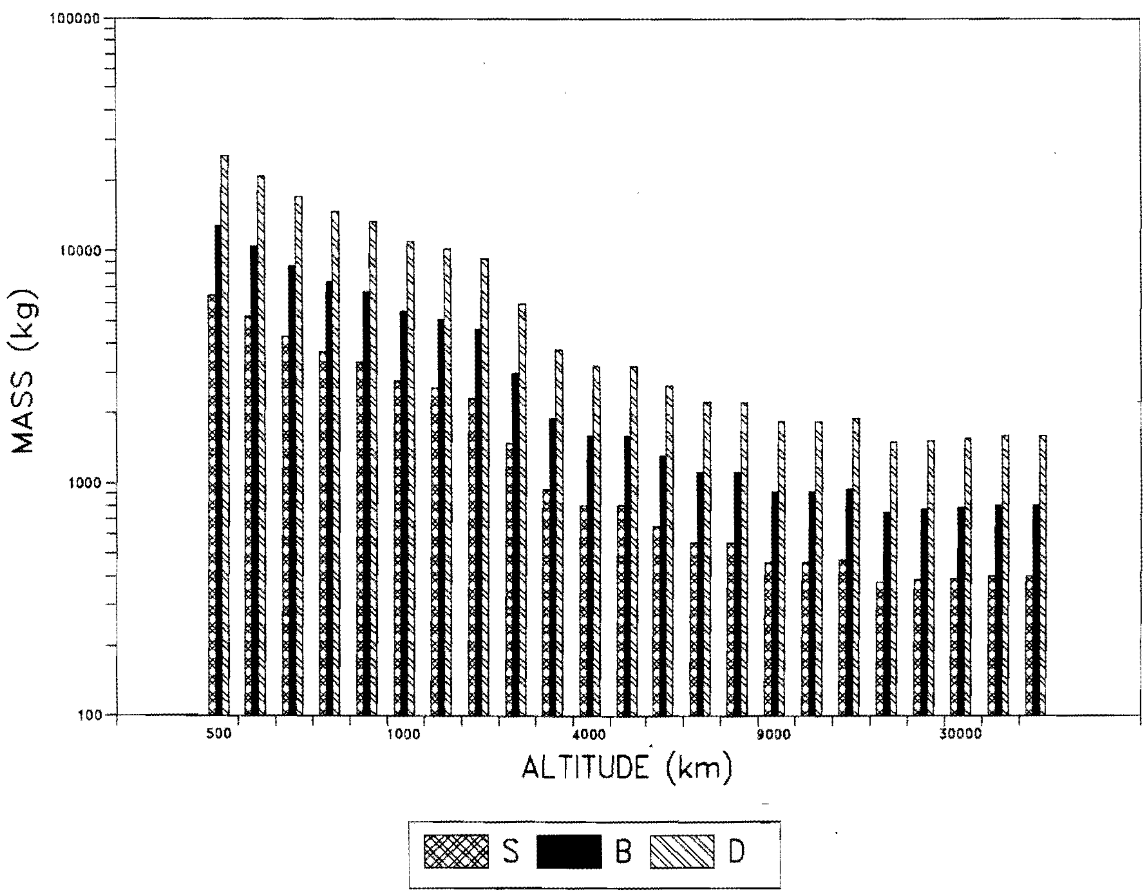

Figure 4. Const. Elec. Weights (Black Box Redundancy)

Combined sensor and electronics weight was assumed to be 40% of the total spacecraft weight. This is based on weight breakdowns of representative systems [9]. Figure 5 shows the combined total weight dependent on component class for proliferated constellations. The lowest weight constellations are centered around 9,000 Km, with the minimum constellation weights biased towards S-class components.

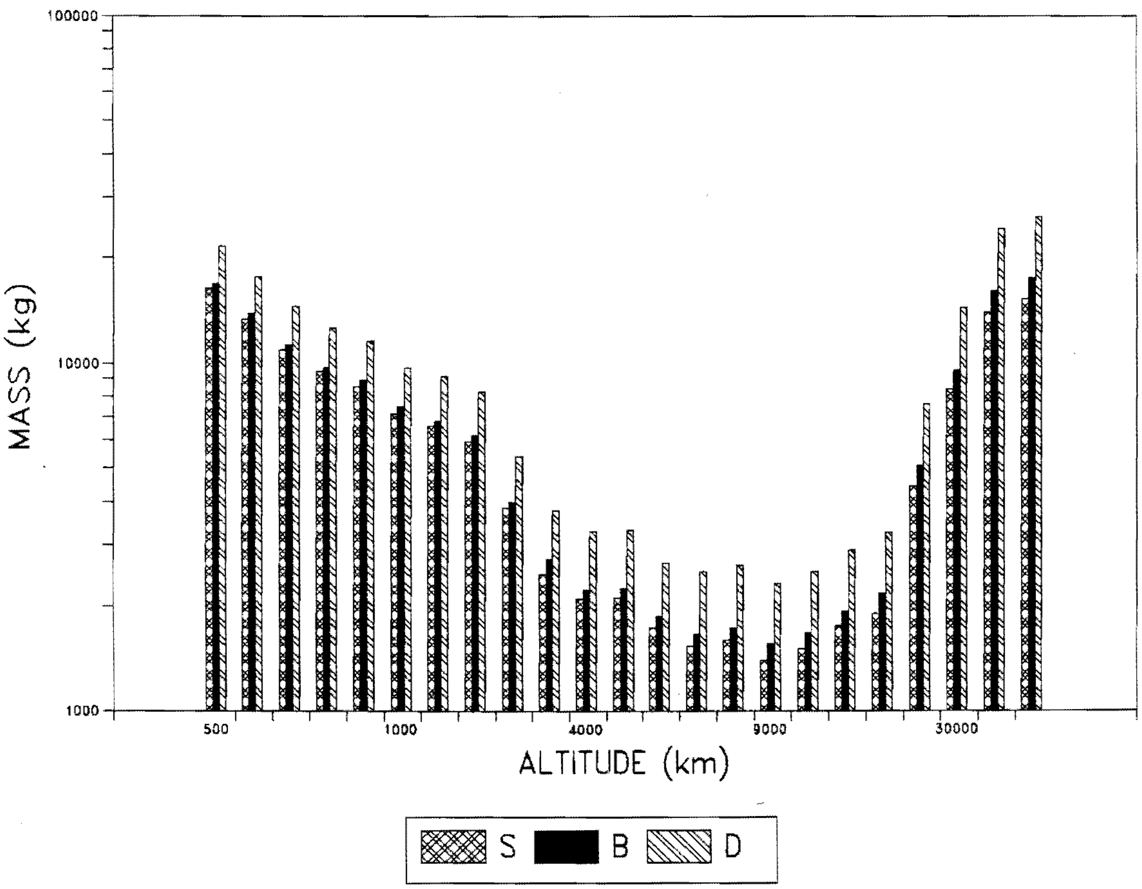

Figure 5. Total Constellation Weights (Proliferated)

Figure 6 shows the combined total weight dependent on component class for constellations using black box redundancy. Again the lowest weight constellations are centered around 9,000 Km, with the minimum constellation weights biased towards S-class components. In general total constellation weights are higher for B and D-class electronic systems using black box redundancy than those constellations using proliferation. They are however lower for altitudes above 25,000 Km.

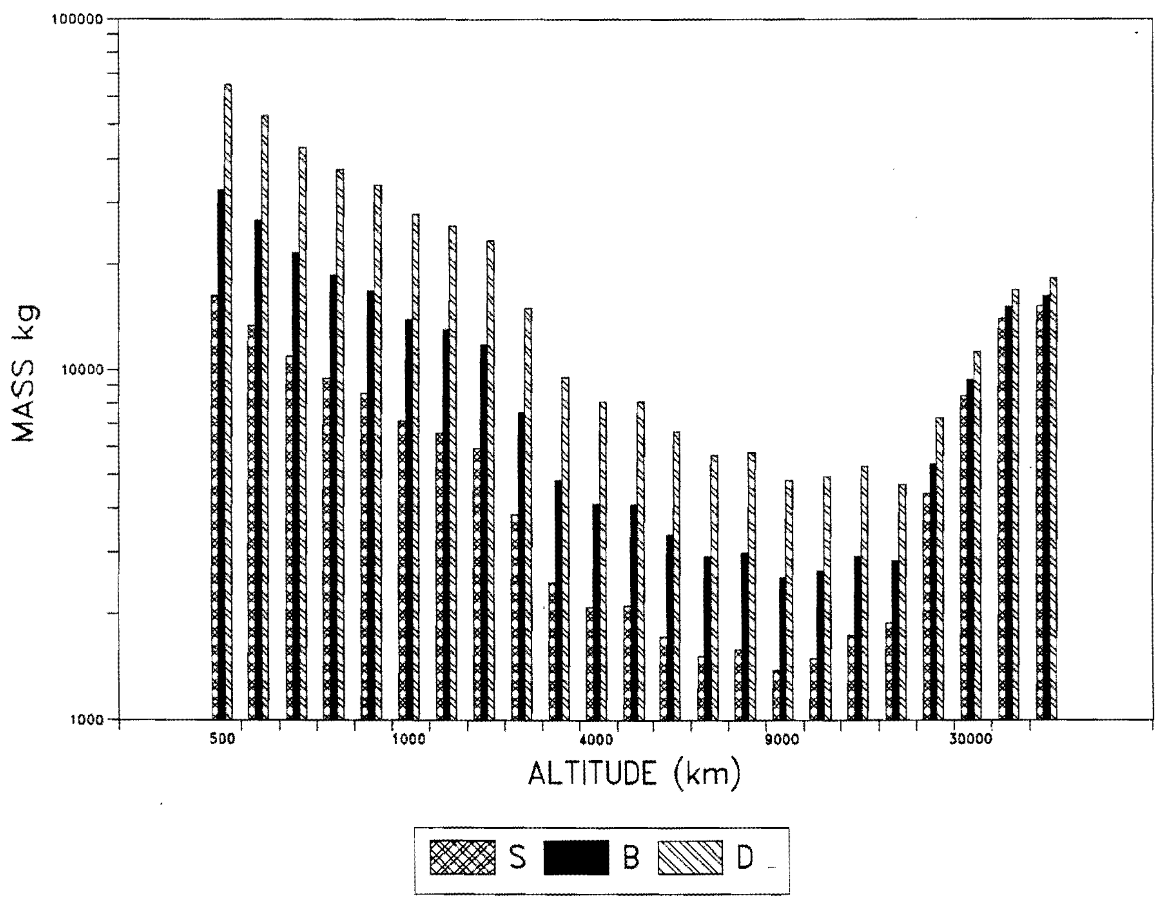

Figure 6. Total Const. Weights (Black Box Redundancy)

Cost

Constellation sensor costs are a function of the number of satellites required in a constellation to meet mission performance and individual sensor cost. The total cost of the SWIR sensors for each proliferated constellation are shown in Figure 7. The trend of the data is similar to the weight data trend. Minima for sensor costs are around 9,000 Km. The lowest total sensor cost of all the constellations is for an S-class constellation at this altitude.

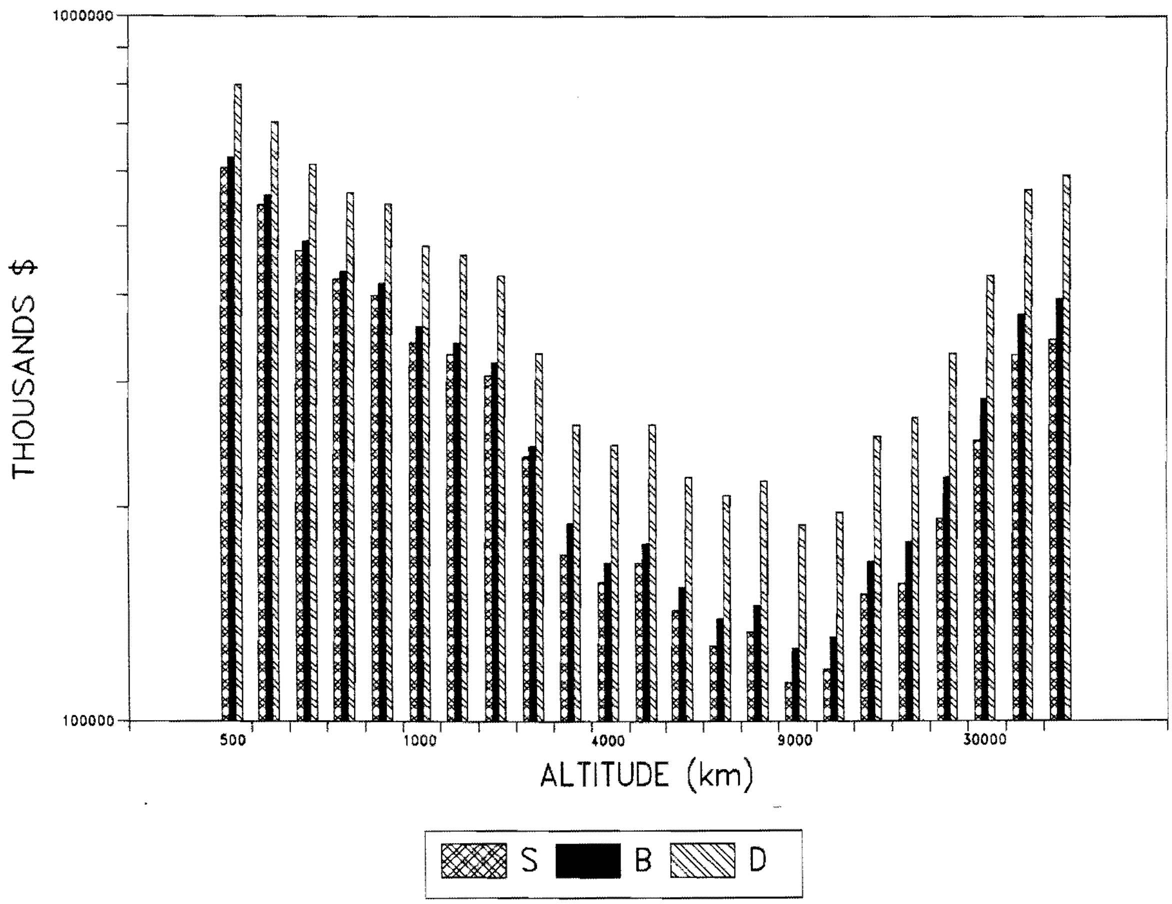

Figure 7. Constellation Sensor Costs (Proliferated)

Constellation sensor costs for constellations using black box redundancy instead of proliferation are shown in Figure 8. There is no variation in cost for a particular altitude since each different component class constellation will have the same number of satellites and sensors as an S-class constellation. As in Figure 7, minima occur at around 9,000 Km.

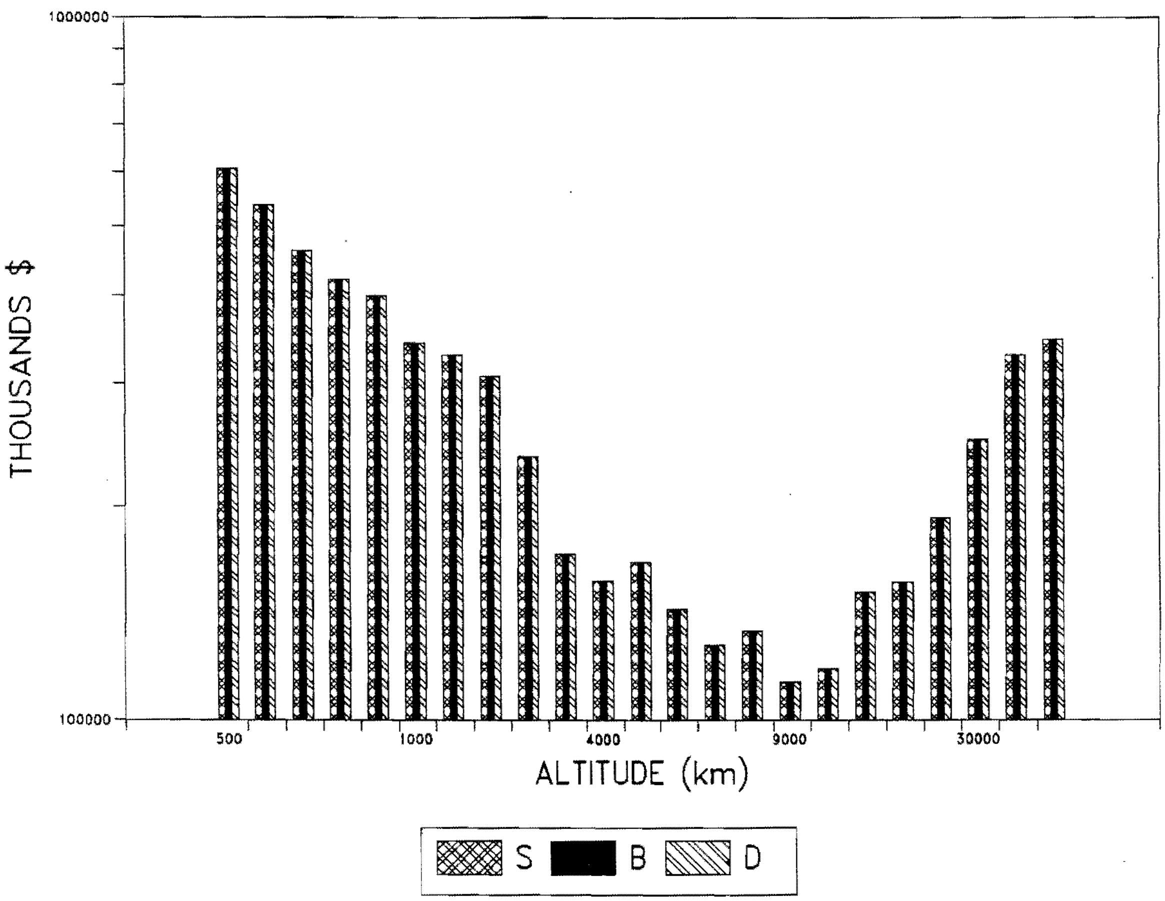

Figure 8. Const. Sensor Costs (Black Box Redundancy)

Constellation electronics costs are a function of the number of satellites required in a constellation to meet mission performance and individual electronics cost. The total electronics cost for proliferated constellations shown in Figure 9 exhibits a characteristic similar to that of sensor costs since electronics cost scales with the number of detector elements. The minimum cost electronics are biased towards the D-class components and occur at 9,000 Km.

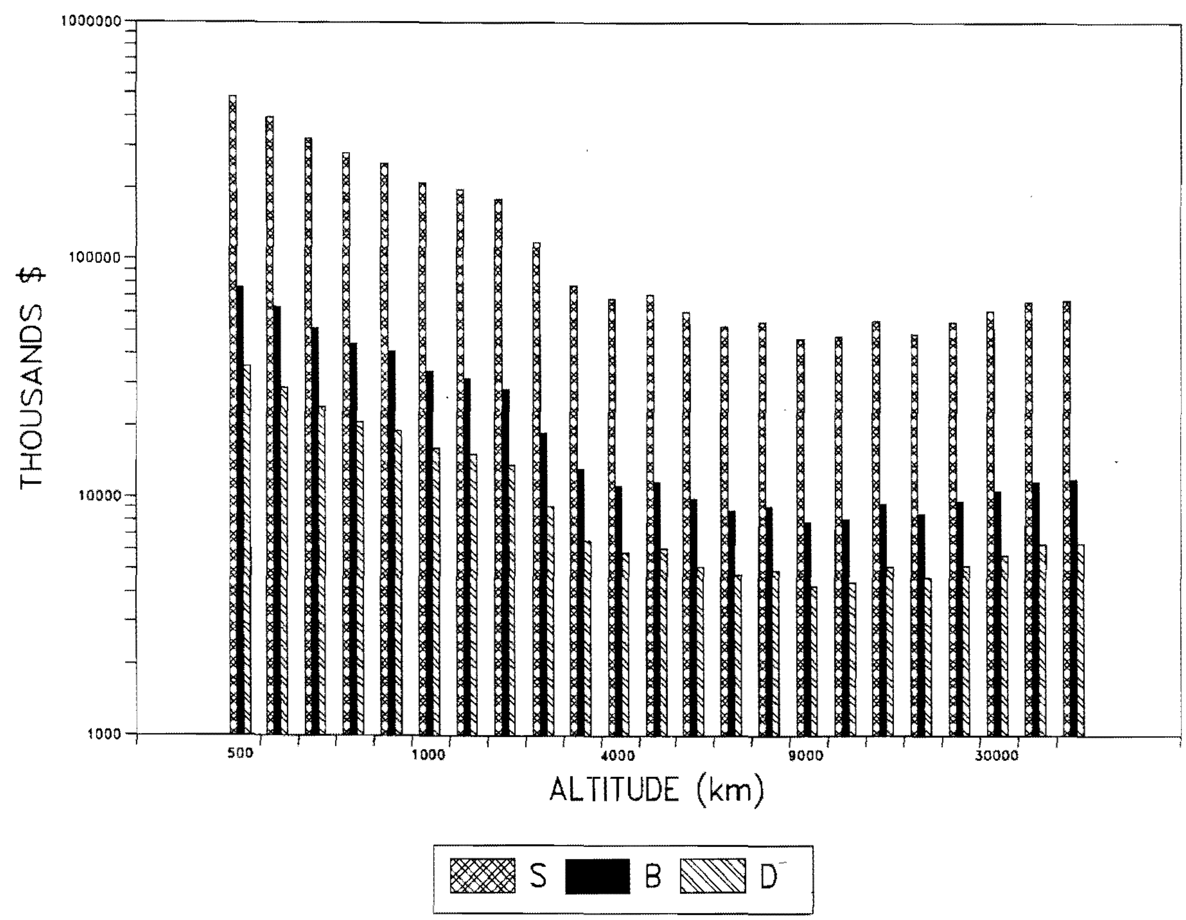

Figure 9. Constellation Electronics Costs (Proliferated)

Constellation electronics costs for constellations using black box redundancy are shown in Figure 10. The cost of S-class electronics for each altitude is identical to those in Figure 9. The cost of B and D-class electronics is more however. This is due to the requirement of 2 and 4 black boxes on each satellite for B and D-class respectively. The minimum cost electronics are biased towards the D-class components at 9,000 Km.

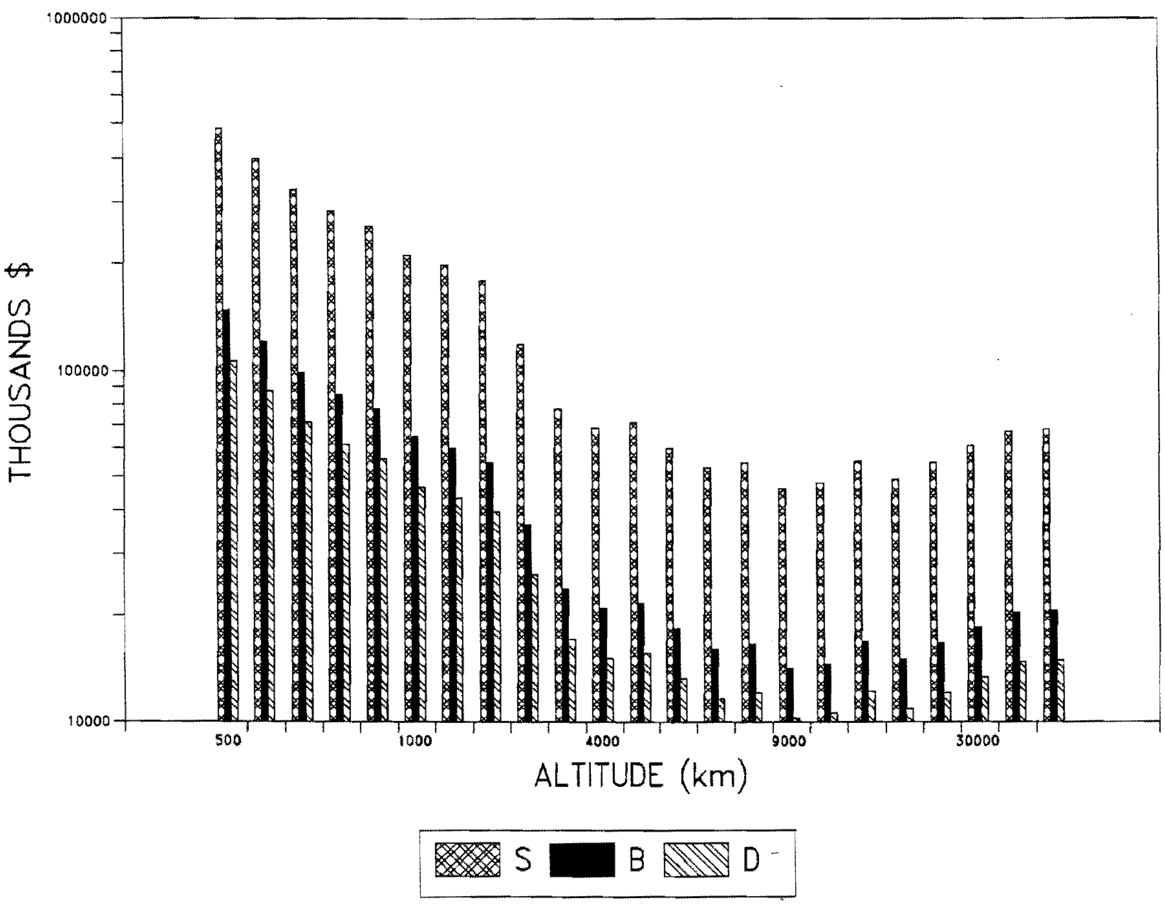

Figure 10. Const. Elec. Costs (Black Box Redundancy)

Combined sensor and electronics costs were assumed to be 67% of the vehicle production cost. This factor is supported by these components being more complex than older technology standard to satellites. Total constellation costs are shown in Figure 11 for proliferated constellations. The trend in total vehicle cost reflects the character of the component weight results.

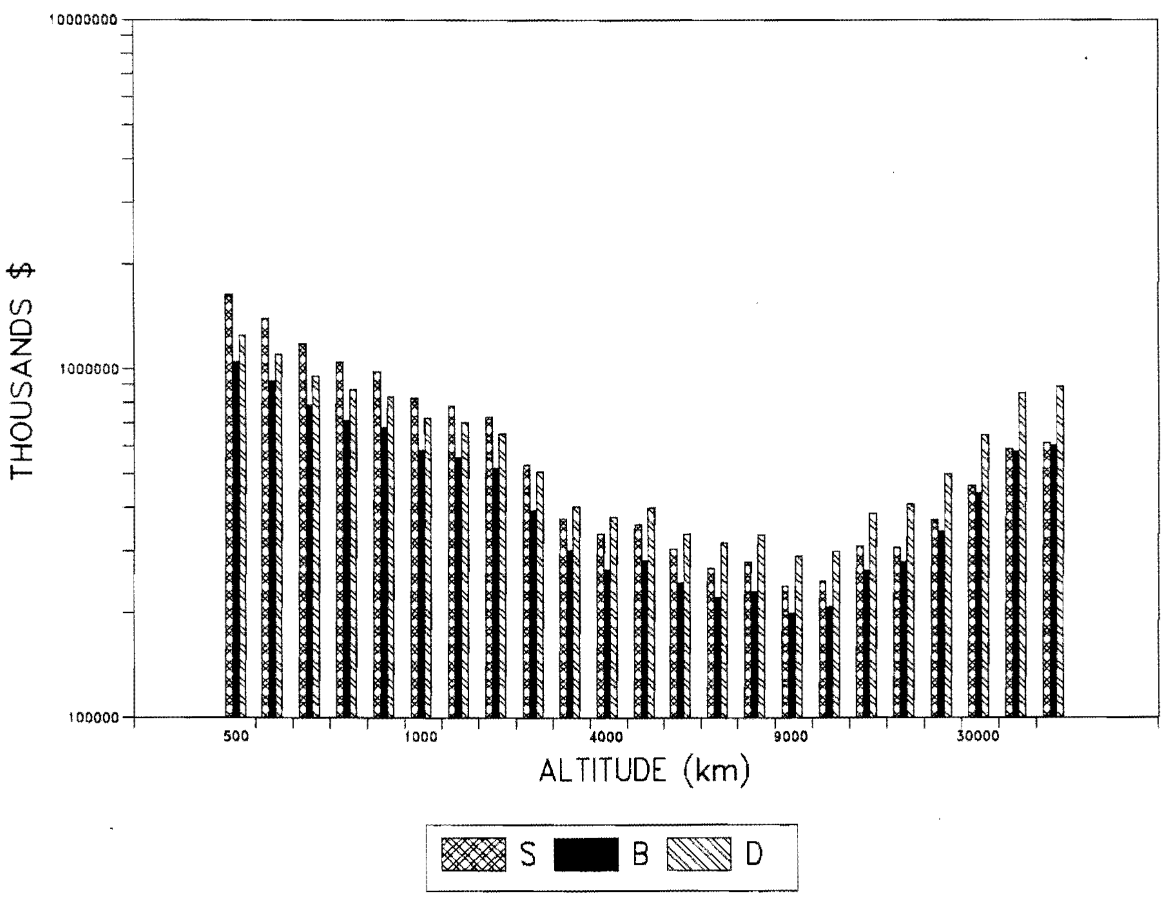

Figure 11. Total Constellation Costs (Proliferated)

Figure 12 shows the combined total cost dependent on component class for constellations using black box redundancy. Again the lowest cost cosntellations are centered around 9,000 Km. The lowest cost constellations are all D-class, with 9,000 Km as the lowest.

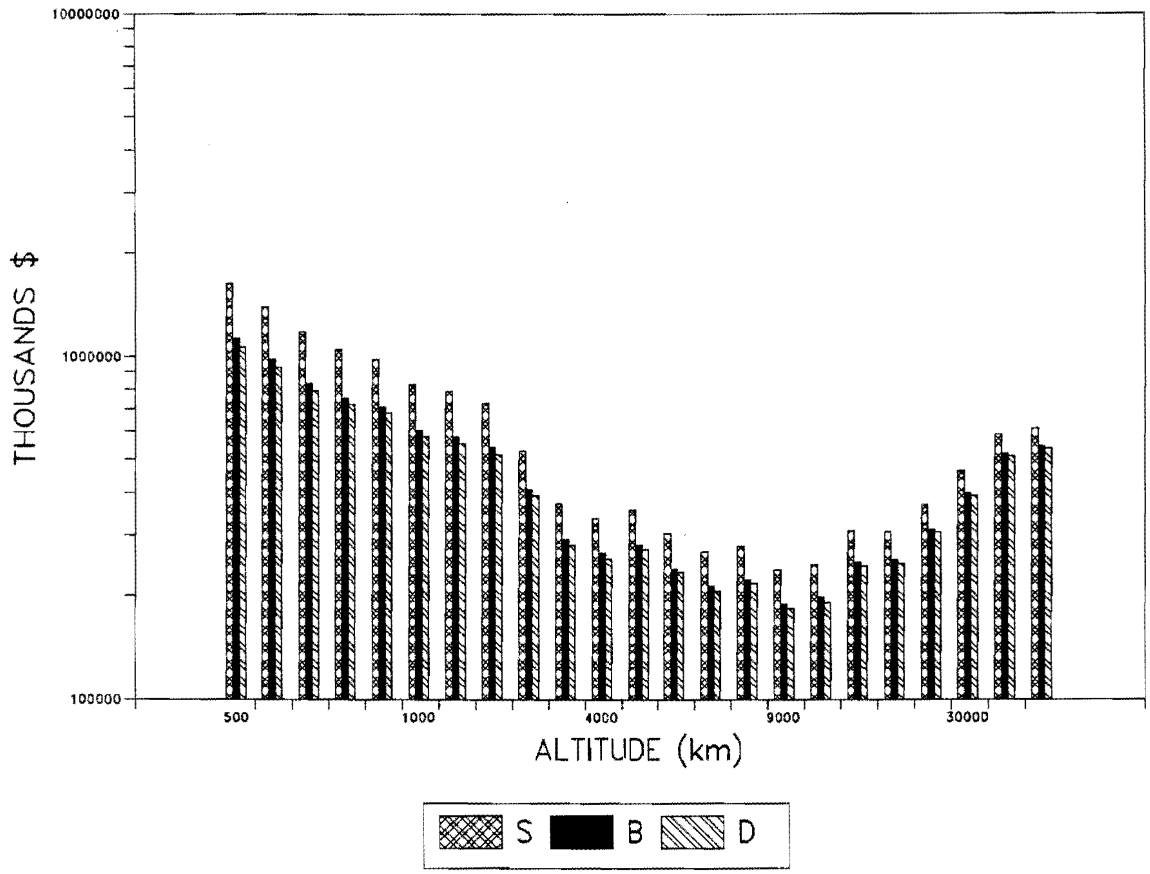

Figure 12. Total Const. Costs (Black Box Redundancy)

The costs associated with launching the satellite constellations must also be considered. A Martin Marietta cost model [10] was used to determine these costs. Launch costs are modeled as a function of payload weight to an operational orbit. The costs associated with launching the various proliferated constellations are shown in Figure 13.

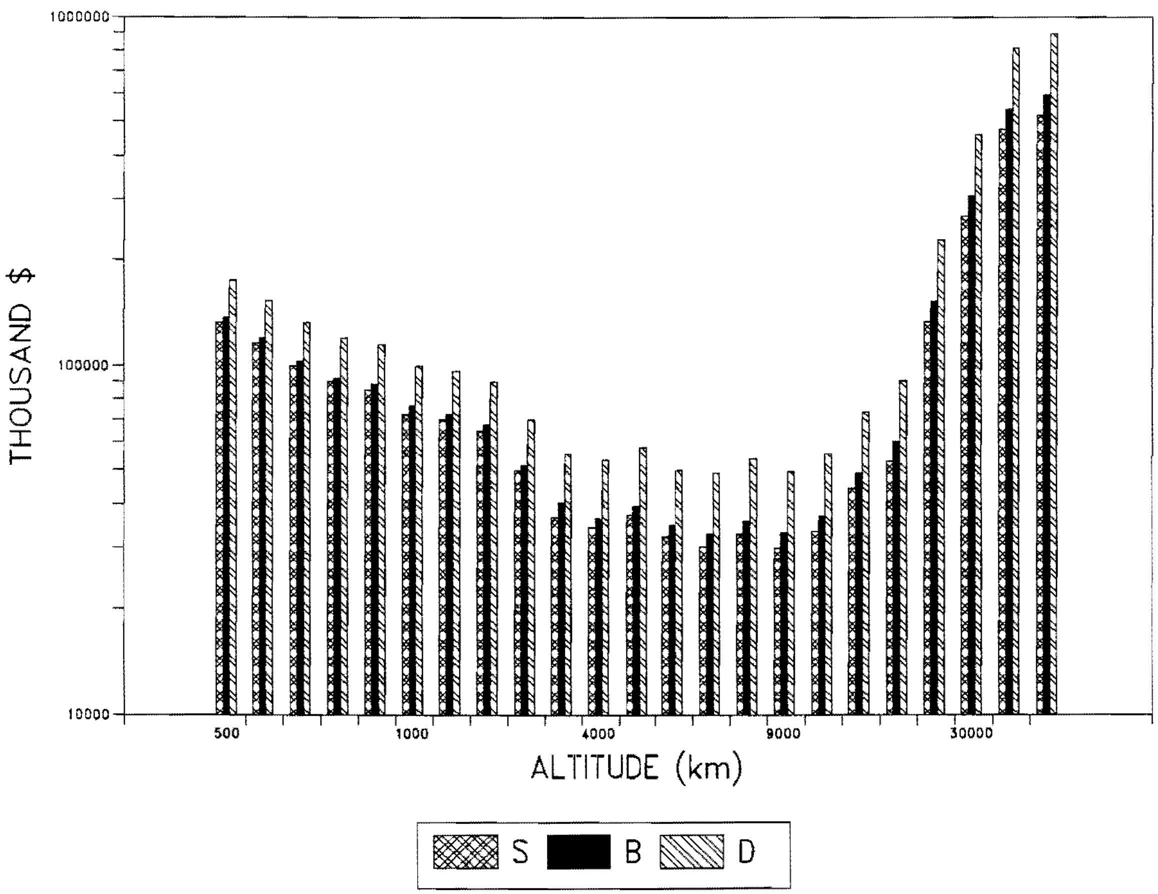

Figure 13. Constellation Launch Costs (Proliferated)

The same cost model was used to determine costs associated with launching the constellations using black box redundancy. These costs are shown in Figure 14.

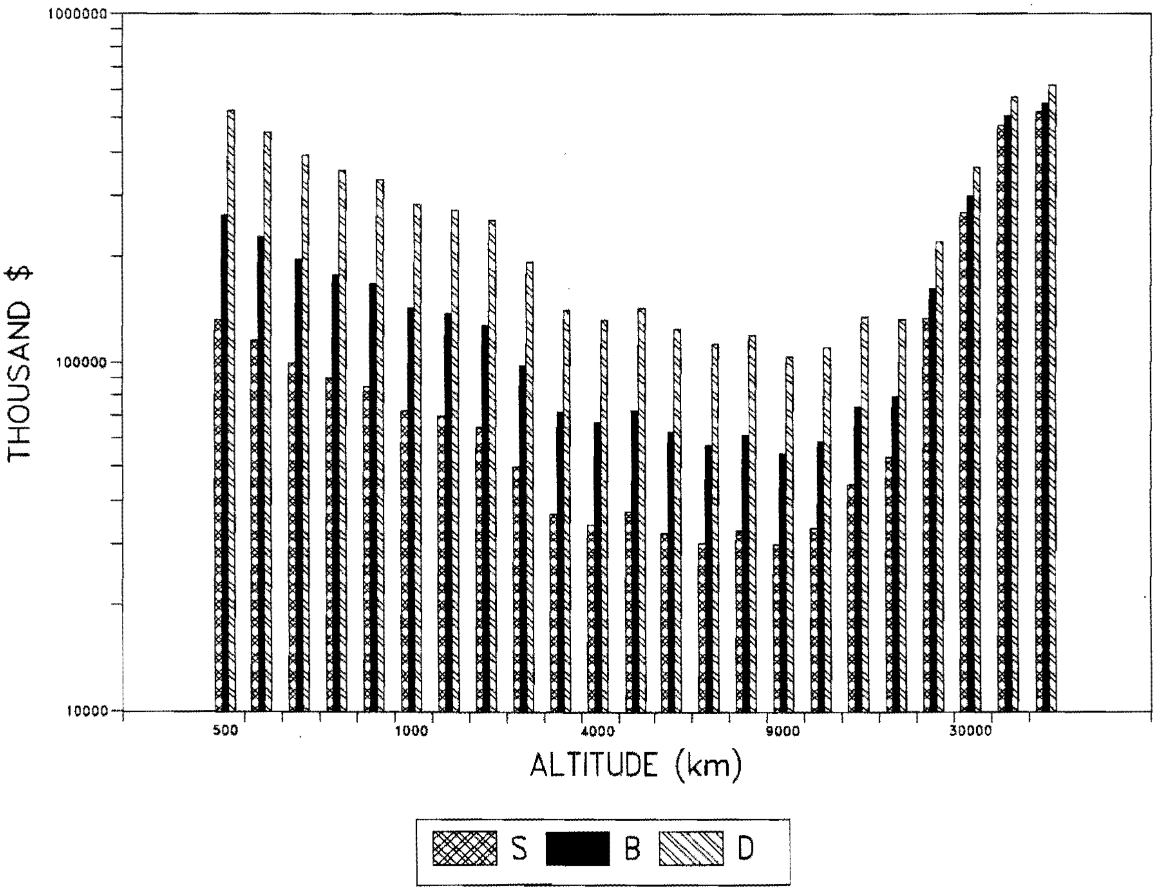

Figure 14. Const. Launch Costs (Black Box Redundancy)

Finally, total costs for proliferated constellations are shown in Figure 15. Total costs are the combined total constellation costs and launch costs. The minimum cost for all component class constellations occurs at 9,000 Km, with B-class the lowest at $233M. The high altitude component satellite constellations are less cost effective than most of the other constellations examined. Only the highly proliferated low altitude constellations rival them in terms of highest total system cost.

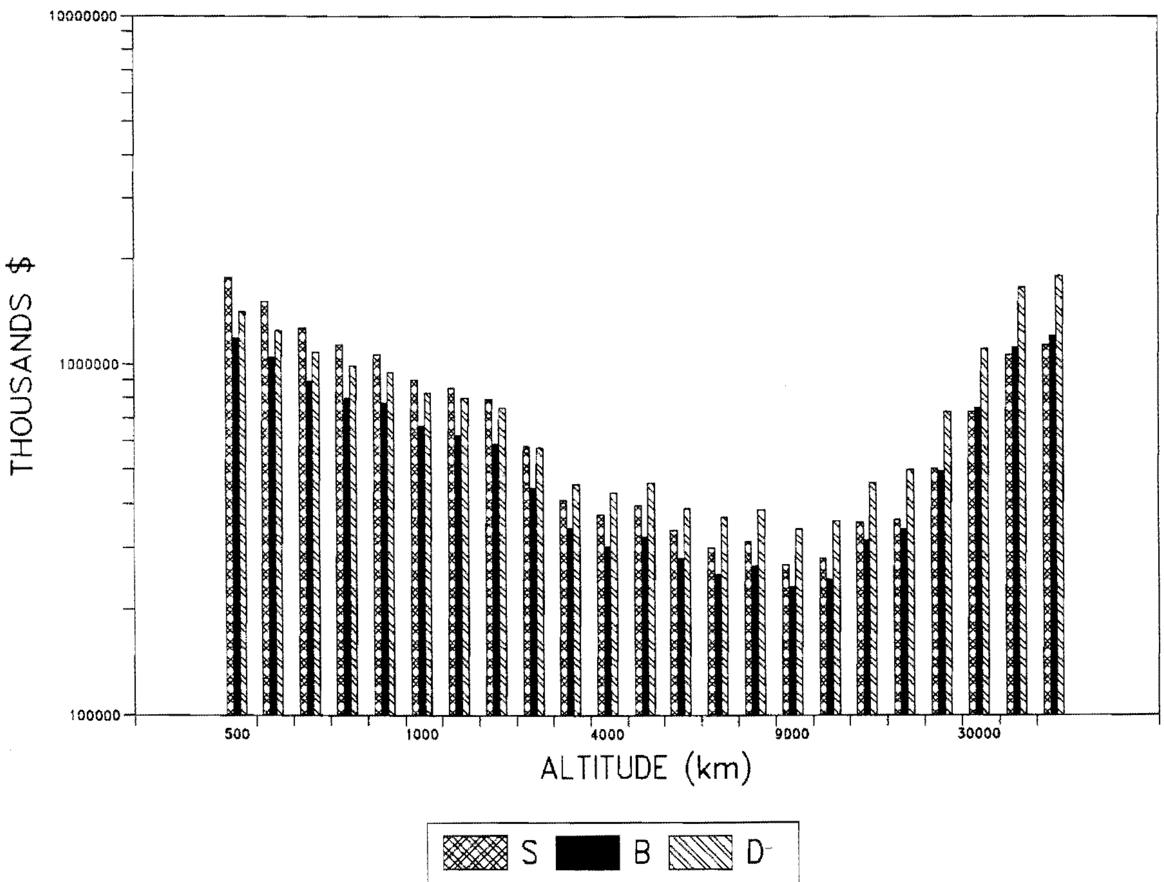

Figure 15. Total Costs (Proliferated)

The total costs for constellations using black box redundancy are shown in Figure 16. The minimum cost for all component class constellations occurs again at 9,000 Km, with B-class the lowest at $238M. Again, the high altitude component satellite constellations are less cost effective than most of the other constellations examined.

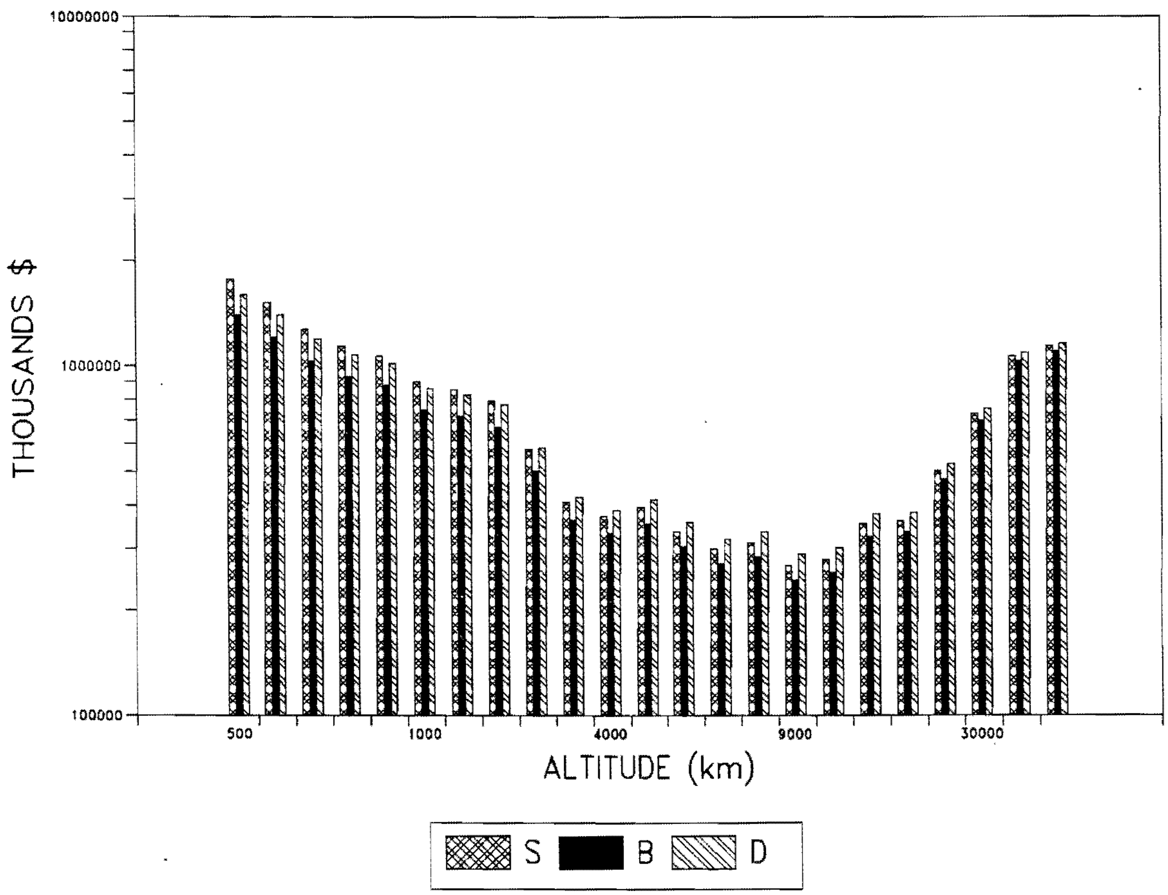

Figure 16. Total Costs (Black Box Redundancy)

VII. SUMMARY

The ICBM launch detection mission can be performed from a range of altitudes with equal success. A total system cost evaluation can be used as a method for determining which altitude is most desirable. The historical approach of high altitude, high component reliability constellations can be shown to be more expensive than lower altitude alternatives. The use of the lower altitude alternatives can reduce mission costs. Very low altitude highly proliferated constellations however, can also be shown to be very expensive. This leaves the middle altitude region. This middle altitude region shows promise for the performance of the ballistic missile launch detection mission based on the combined costs of constellation configuration, sensor payload, electronics reliability and associated launch costs.

VIII. AUTHORS

Mr. Anthony Dietl and Mr. Robert Nordyke are Space Systems Analysts at Thomas/Scifers Inc in El Segundo specializing in satellite survivability and space architecture studies.

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