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Volume III: LOVES Computer Simulations, Results, and Analyses

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MANNED SYSTEMS UTILIZATION ANALYSIS (STUDY 2.1)

FINAL REPORT

Volume III: LOVES Computer Simulations, Results, and Analyses

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FOREWORD

This report has been prepared to document the capabilities of the LOVES computer code and to present to the customer, NASA, the results of analyses performed in the demonstration process. The analyses were performed at various stages of program development and, in most cases, led to modifications or additions to the code.

For the most part, different versions of the program were used to conduct the various analyses. The differences were, however, differences in refinement rather than in basic program function. Since most of the analyses were comparative in nature, the early analyses should be as valid as the later ones even though they may not compare on a one-for-one basis. The comparisons and trends revealed in Section 5, should be useful as an aid in directing future detailed studies of satellite servicing policies, vehicle operations, and satellite procurement requirements. Several analyses utilizing a solar electric propulsion stage (SEPS) in conjunction with a chemical tug were also conducted. Though not exhaustive, they do show how such a vehicle might be used to advantage in an earth orbital program.

This effort has been performed by The Aerospace Corporation as part of Study 2.1, Manned Systems Utilization Analysis under NASA contract NASW-2727. The NASA Study Director is Dr. J. W. Steincamp, NASA MSFC, Code PD 34.

ACKNOWLEDGMENTS

This report is a compilation of analyses that were performed to demonstrate the capabilities of the LOVES computer code. They represent the product of the computer code at various stages in its development. Results of many of the analyses provided valuable insights into additions and modifications that were needed, and were ultimately incorporated into the code. Appreciation is expressed to Mr. V. Huff and Dr. J. W. Steincamp of NASA for their support of, and contributions to, the development and demonstration of the LOVES code.

A number of people at The Aerospace Corporation were involved in supporting the coding and demonstration efforts. Primary among those were: Messrs. T. Trafton, M. G. Wolfe, and R. R. Wolfe for the design of the space serviceable payloads used in the analyses, Messrs. H. Hecht and J. Johnson for their reliability assessments and Mr. S. T. Wray for the design, implementation and modification of the LOVES computer code.

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1. INTRODUCTION/BACKGROUND

The LOVES computer program was developed primarily to study the concept of space serviceable spacecraft. These spacecraft are composed of a few long life, nonreplaceable units (NRUs) and a larger number of space replaceable modules (SRUs). The philosophy associated with the concept is to replace only the equipment that has failed when a satellite becomes inoperative rather than replacing the entire spacecraft.

In the process of generating data for the analyses, the LOVES code operates in the following manner. The program first delivers an original inventory of satellites to orbit according to a predetermined schedule. Once the satellites are on station, the program computes a random time to failure for the replaceable and nonreplaceable modules comprising the satellites. The computation is based upon the known reliability expressions for the modules and a random number selected for the reliability value. The program then stores the failure information for subsequent action. When the appropriate clock time arrives in the simulation, the modules are declared to have failed. Replacement modules are put in a waiting line (queue) with other replacement modules awaiting flights, according to their priority. Highest priority modules are assured of waiting in line no longer than some maximum time specified by the analyst. If such a module has not been launched by the end of the time period specified, a dedicated flight is scheduled to deliver it to orbit. These modules are referred to as mandatory payloads. All other replacement modules are considered to be spares and are flown as payloads of opportunity.

When a flight becomes available, as many modules as can be carried by the vehicle are removed from the front of the queue and loaded aboard the flight. Once in orbit, the replacement modules are substituted for the failed modules and the process is repeated. This iterative process is continued for all satellites in the mission model until the simulation is terminated.

Multiple simulations are then performed in a Monte Carlo process to arrive at statistical distributions of various logistics elements. Some of the more important elements reported on are: module failures, module replacements, equivalent satellite procurement, vehicle flights, vehicle load factors, expended vehicles, flight charges for each satellite, flight charges for each satellite system, satellite availabilities, and satellite system availabilities.

The LOVES code can be made to operate using warnings of impending failure rather than the failure event itself. All that is required is to specify the reliability parameters associated with the warning event instead of (or in addition to) the expression for the failure event. The LOVES code assures that the warning event always precedes the failure event in case both are specified to be computed.

The program does not normally consider a replacement module to be a mandatory payload unless the failure of the module for which the replacement is scheduled has caused the satellite to fail. For example, only the replacement module for the fourth module failure in a redundant set requiring one of four modules to be functional would be designated a mandatory payload. If, however, the requirement was for two of the four modules to be functional, the replacement modules for both the third and fourth module failures would be declared to be mandatory payloads. Replacements for spare modules can also be declared mandatory by using an override option available in the program, if the analyst so desires.

The LOVES program was used to perform a number of analyses of the geosynchronous portion of the 1973 NASA Automated Payload Model (Ref. 1). The basic purpose of the analyses was to perform tradeoffs between the relatively new concept of space servicing of automated satellites and the present concept of replacing the entire satellite with another, upon failure. The space serviceable satellites are redesigned versions of their expendable counterparts (Ref. 2). The satellite's basic objectives and function remain the same but the equipments have been modularized and the satellite reconfigured to provide easy access to the modules for changeout, upon failure. The changeout is accomplished using an automated service unit attached to (or an integral part of) an appropriate upper stage vehicle. The tradeoffs were made utilizing various upper stages, thereby providing an opportunity to perform tradeoffs among the candidate upper stages also.

In addition to the primary analyses comparing satellite and upper stage options, a number of LOVES analyses were made to investigate the sensitivity of the space servicing concept to various mission model, satellite, vehicle, and operational parameters. The objectives of the various analyses conducted were:

1. Investigate the effect of the service unit weight on the flight requirements for the geosynchronous mission model.
2. Investigate the effects of utilizing a large tank Centaur upper stage for extended missions of up to seven days. The additional time involved eases the mission velocity requirements but the boiloff of cryogenic propellants is greatly increased. Therefore, the vehicle performance becomes a tradeoff between these two opposing factors.
3. Investigate the effect the degree of replaceability on redundantly configured elements has on the flight requirements of a typical satellite mission model.
4. Investigate the effects of varying the response time to a failure on the flight requirements and satellite availability. This was done by varying the maximum time that a mandatory payload was permitted to remain in the waiting queue prior to launch.
5. Investigate the sensitivity of flight requirements and satellite availabilities to the number of spare modules incorporated in the satellite design. This was done by limiting the number of spare module replacements that were permitted to enter the waiting queue.

The LOVES program has undergone substantial revisions during the course of its development and with further use will undoubtedly undergo more. As with most programs of this nature, the results obtained give rise to additional questions to be answered that in turn lead to additional modifications to the program code. The analyses performed in conjunction with the program development have been very informative; however, for an indepth investigation of all facets of the space servicing concept, they need to be expanded considerably using the latest satellite and vehicle designs.

2. SUMMARY

The LOVES computer program was employed to analyze the geosynchronous portion of the NASA's 1973 automated satellite mission model from 1980 through 1990. The objectives of the analyses were twofold. One objective was to demonstrate the capability of the LOVES code to provide the depth and accuracy of data required to support the analyses. The other was to tradeoff the concept of space servicing automated satellites composed of replaceable modules against the concept of replacing expendable satellites upon failure.

Three options were identified and investigated in the space service concept:

1. Return both the service unit and the spent modules
2. Return only the service unit
3. Return nothing

The first objective was attained in the process of attaining the second. The computer code proved to be an invaluable tool in analyzing the logistic requirements of the various test cases required in the tradeoff. Numerous independent verifications of program results were made as the analyses progressed to assure the satisfactory operation of the code.

The tradeoffs proved the space servicing concept to be superior to the satellite replacement concept. In all cases where the same upper stage and operational constraints were used, the concept of space servicing proved to be more economical than its expendable satellite counterpart, in spite of the heavier satellite weights associated with the space serviceable design. In only one instance was the use of expendable satellites shown to be superior to the use of space serviced satellites. In that instance different upper stages were employed in the analysis of the two concepts. The FCT was used as the upper stage in the expendable satellite concepts, whereas the Centaur was used in the space serviced case. The space service option in question was the most demanding of the three options, i.e., the one requiring the return of the service unit and the spent modules. For that option with its demanding performance requirements, the high boiloff rate of the large tank Centaur proved to be too much of a handicap for the space service operation to overcome.

For comparable conditions, the space service concept shows cost benefits over its expendable satellite counterpart ranging from $150M to $350M over an eleven year period. The exact figure depends on the upper stage employed and on the space service option selected. If one takes into consideration the additional programs not included in the synchronous orbit sample and the fact that the space program will be an ongoing effort, the savings should be much greater than the indicated figures. These amounts of money should be more than enough to compensate for the expenditures required to develop an operational space service capability.

If operations employing the Transtage/Kick combination are taken as the reference for the expendable satellite concept, a space serviced concept utilizing the Centaur upper stage could provide savings ranging from $290M to $500M. The FCT could provide benefits ranging from $635M to $725M. The limited SEPS analysis indicates that utilizing the SEPS in conjunction with the FCT could provide at least an additional $150M in benefits, resulting (conservatively) in a minimum savings of approximately $785M.

The lower figures quoted in the above ranges apply to the space service option requiring the return of the service unit and the spent modules. The higher values refer to the option that returns nothing from orbit. If the space replaceable modules and the service unit could be left in orbit, substantial additional savings could be realized. Additional savings of $90M to $210M are indicated depending on the upper stage being utilized. There must be a tradeoff of these savings against the savings that may be realized in refurbishing and reusing the modules and service unit rather than procuring new ones for each service mission. Even if just the spent modules could be left in orbit, a sizeable additional benefit could be accrued compared to returning both the service unit and the SRUs. Returning payloads is expensive. As a rule of thumb, each pound of payload that must be returned reduces the payload that can be deployed by 2 to 3 pounds.

In summary, it should be said that the concept of space servicing offers the potential for substantial savings in the cost of operating automated satellite systems. The benefits attainable by space servicing satellites over the continued use of expendable satellites is not unlike the benefits the Shuttle will provide over the continued use of expendable launch vehicles. It appears, therefore, that continued study and development of the concept should be given serious consideration.

3. SPACE SERVICING VERSUS EXPENDABLE SATELLITES

3.1 GENERAL

A primary objective of the analyses performed was to demonstrate the capability of the LOVES computer program to analyze the logistics requirements of future space program concepts. One of the most interesting and potentially rewarding of these concepts is the concept of space servicing of automated satellites. Unlike the existing practice employing expendable satellites, the space servicing operation replaces only those modules of equipment that have failed, rather than the entire satellite. In theory at least, this should result in lower procurement costs and lower flight rates for a given mission model. However, the increased weight of the satellites as redesigned for space servicing and the requirement for the use of a service unit (representing additional weight) tend to offset the advantages of the concept. To determine which of the two concepts was the best, a series of analyses was performed to support a tradeoff between them.

In order for the space service concept to be competitive there must be sufficient traffic to be able to share service flights among a number of different satellites, thereby providing high vehicle load factors. The goal of fully loaded, shared, service flights suggests that the total space program being serviced must be of fairly sizeable proportions and/or the availability requirements of the satellite programs involved must not be excessively demanding. To permit a valid tradeoff to be made, it was necessary to select a satellite mission model large enough to provide the demand for shared logistics flights. The synchronous equatorial missions from the October 1973 NASA Automated Payload Model were chosen because they appeared to meet the necessary requirement.

Detailed definitions of the redesigned spacecraft have been previously reported in Reference 2. Not all of the satellites operating in the synchronous equatorial regime were amenable to redesign for space servicing. Therefore, those that could not be redesigned remained configured as expendable satellites in the model. The physical definitions of those expendable satellites used in the analyses were contained in the expendable satellite volume of NASA/MSFC's Index of Defined Satellites, dated 7 February 1974 (Ref. 3).

The results of the analyses were obtained in terms of equivalent spacecraft procurement, vehicle procurement, vehicle flights, and spacecraft system availability. In order to get the data in a form that could be compared, the data was reduced to the common base of cost. Prior to the costing operation, results were normalized to relate to the same average satellite availability. The normalization process was approximate and involved modifying all results to relate to a common level of payload procurement. This presumes equal availability for equal payload procurement, which is a reasonable approximation, although not exact.

Three basic sets of comparisons were made in the course of performing the analyses. The first set was performed to compare the two servicing concepts, i.e., expendable satellite operations versus space serviced operations. In this set, the upper stage vehicle remained fixed for both concepts. The expendable satellite operation was actually compared against three different space service options:

1. The first option provided for the retrieval and return of the failed modules that had been replaced.
2. The second option permitted the failed modules to be left in orbit but provided for the return of the service unit.
3. The third option permitted both the failed modules and the service unit to be left in orbit.

The second set of analyses was performed to compare the effects of employing different upper stage vehicles. In this set, the servicing concept remained fixed while the vehicles were varied. Five basic vehicles or vehicle combinations were employed in the analyses. They were:

1. An expendable Transtage, 10 feet in diameter and 20 feet in length.
2. A two-stage Transtage/Kick combination where the Kick was expended but the Transtage was recovered.

3. A recoverable large tank Centaur, 15 feet in diameter and 28 feet in length.

4. A recoverable, FCT, 15 feet in diameter and 30 feet in length.

5. The FCT used in conjunction with a SEPS. Due to time and budgetary constraints, analyses employing the SEPS were limited to a couple of demonstration runs.

The source of data for the physical and performance characteristics of the 28-ft Short Reusable Centaur, the 20-ft Reunsable Transtage and the Synchronous Equational Kick Stage (used in conjunction with the Transtage in some analyses) was an internal memorandum written for the USAF/SAMSO Upper Stage assessment of November 1973. The physical and performance characteristics of the full capability cryogenic tug were obtained from the NASA baseline tug document (Ref. 4). Data for the SEPS was obtained from Rockwell International final SEPS study documentation (Ref. 5). Pertinent characteristics of the various vehicles are included in Appendix C.

In addition to the comparisons made within each of the above sets, a number of cross comparisons between the two sets were also made once the data was available. The third set of analyses were performed to gain some insight into the sensitivity of the space servicing concept to the various parameters involved. For example, the sensitivity of the concept to the weight of the service unit employed was investigated. The effects of varying the maximum waiting time until launch of a replacement module was another of the additional analyses performed.

Since the time that the above definitions were current, several design iterations have been accomplished on the satellites comprising the NASA's Mission Model, as well as on the composition of the model itself. Likewise, several updates in the USAF's interim upper stage designs have been accomplished within the past year. Many additional changes will undoubtedly be made as requirements and ground rules continue to evolve; however, each iteration produces a representative model against which a valid analysis, using the LOVES computer program, can be performed. Since the analyses performed were comparative in nature, drastic changes would have to occur in the mission model and/or vehicle definitions to invalidate the results obtained from the LOVES analyses.

3.2 EXPENDABLE SATELLITE OPERATIONS VERSUS SPACE SERVICED SATELLITE OPERATIONS

Both the expendable satellite model and the space serviced satellite model have their origin in the October 1973 NASA Automated Payload Model. In looking at the 95 satellite programs in the reference model and assessing the feasibility of redesigning the satellites for space servicing, it was found that 42 of the 95 programs were reasonable candidates for reconfiguration (Ref. 2). The satellites comprising the 42 programs were subsequently redesigned for space servicing. Of the 42 satellite programs, 23 were scheduled for operation in synchronous equatorial orbit. Considering all synchronous equatorial satellite programs, only six could not be reconfigured for space servicing without a severe weight penalty. Therefore, the synchronous equatorial program appeared to be a "natural" for use in the comparative analyses. It provided the necessary high percentage (80%) of redesignable satellites to make a good comparison and it constituted a large enough segment (25%) of the reference satellite model to be a representative sample.

Detailed reliability and design data for the space serviceable version of the satellites was obtained from the FY 74 NASA Study 2.1 (Ref. 2). These data were used as input to the LOVES computer program to obtain the results used in the comparative analysis. The satellite weights, after redesign for space servicing, were considerably higher than their expendable counterparts; in some cases almost twice as heavy.

Although the weights of the expendable satellites were considerably less than the space service versions, their functional configurations were considered to be equivalent. The expendable satellites were presumed to consist of the same number and arrangement of modules as their space serviceable counterparts. In that way, the servicing concepts incorporating the two satellite configurations could be compared on a one-for-one basis. The basic difference, of course, was the fact that in servicing a satellite failure in the expendable model, the satellite had to be replaced with another satellite. In the case of the space serviceable model, only failed modules were replaced.

Reliability data was presented in the form:

$$R = e - \left(\frac{t}{\alpha}\right)^{\beta}$$

where α and β are Weibull parameters representing the failure or warning characteristics of the individual modules and t is the time. By supplying a random number R_n (from a random number generator) for R, random failures or warnings of the modules are simulated by the program. Each time a module is replaced by a new module, the time to failure or warning of the new module is computed using the reliability expression. Weibull parameters may be given for either or both of the warning and failure events. If either the warning or failure event (α_W or α_F) is zero, no warning or failure can occur for the module and the time to warning or failure (t_W or t_F) is considered to be zero. If α_W is zero and α_F is not, $t_W = 0$ and t_F is defined by the expression

$$t_F = -\alpha_F \left(\ln R_n\right)^{1/\beta_F} .$$

where R_n is a random number generated by the program. If both α_W and α_F are not zero, t_W is defined by the expression

$$t_W = -\alpha_W \left(\ln R_n\right)^{1/\beta_W}$$

and t_F being conditional on having a t_W is expressed as follows:

$$t_F = -\alpha_W \left\{ \ln \cdot R_n \left[e^{-\left(\frac{t_W}{\alpha_F}\right)^{1/\beta_F}} \right] \right\}^{1/\beta_F}$$

Once the times have been computed, the program puts the events on a calendar for execution at the proper time in the simulation. In those cases cases where a space serviceable counterpart for an expendable satellite had been determined to be impractical, estimates of the reliability and redundancy levels of the expendable satellite were provided. No reliability or module redundancy information was provided in the reference satellite model.

A number of optional initial conditions, limits, ground rules, and procedures were arbitrarily set when initiating the analyses. The program has some 40 to 50 optional input variables and operational switches to provide analytical flexibility. All of these must be specified and/or set before an analysis can be performed. More important than the actual values or settings provided was the fact that in performing tradeoff studies those applicable to both service concepts were held constant throughout the analyses. One of the more sensitive variable features was the one concerned with the number of spare modules left remaining operational (in a redundant set of modules) when a mandatory replacement flight was scheduled. For the analyses conducted, the program was set to initiate a mandatory replacement flight when the last spare module in the redundant set had failed. In other words, the satellite was still operational but one more failure in that particular set of redundant modules would cause the satellite to fail. The analyst has to specify in the input data the number of modules that must remain functional in order that the satellite remain operational; however, he can choose to initiate a mandatory replacement flight upon the failure of any module. Another important and sensitive feature to be specified is the one defining the maximum time that a mandatory replacement module may be left in the loading queue before it must be flown on a replacement flight. The program permits different times to be specified depending on whether the satellite is functional or out of service at the time a failure occurs. Since the program was set to react with mandatory flights while the satellites were still operational, both values were set for the same period, 90 days. A partial list of other optional values and/or settings imposed for the analyses are as follows:

1. Service mission duration of 7 days
2. Service unit weight of 182 kg (400 lb)
3. Service unit capacity of 16 modules
4. Shuttle/tug turnaround time of 15 days
5. Unlimited vehicle fleet size
6. Random failure option set
7. Reusable vehicle option set
8. Satellite replacement policy set to leave failed satellite in orbit
9. Space service policy set to return service unit and failed modules
10. Satellite/module procurement and checkout time set to zero.

The initial analyses used to compare the two servicing concepts were performed using only the Centaur upper stage. The FCT design had not been finalized at the time and the Transtage had insufficient performance to service the geosynchronous orbit. Reference runs using the Transtage and Transtage/Kick combination were also made in the case of the expendable satellite model. In the former case, the Transtage was expended, whereas, with the Transtage/Kick combination, the Transtage was recovered wherever possible.

Results of the analyses are presented and discussed in Section 5.0. Figures 5-1 and 5-2 summarize the results obtained from the analyses.

3.3 CANDIDATE UPPER STAGE COMPARISONS

Three basic vehicles were included in the comparative analysis of upper stage configurations. The three, that were previously identified, were the Transtage, the Centaur, and the MSFC full capability tug. In addition, two other configurations were given limited consideration. The first was the Transtage/Kick combination that was employed only in the expended satellite concept. The second configuration was the FCT used in conjunction with a SEPS stationed in synchronous equatorial orbit. The SEPS operation can be restricted to synchronous orbit in a so called "scooter" mode or it can operate in a combined ascent/scooter mode. In the latter mode, the SEPS meets the FCT at some intermediate altitude where they exchange payloads. The SEPS then delivers the FCT payload to synchronous orbit while the FCT delivers the SEPS payload to the shuttle for return to earth. The FCT/SEPS combination was analyzed for only one of the three space service options, i.e., the mode where the failed modules and the service unit are returned to the shuttle. Time and budget constraints did not permit the evaluation of the SEPS in other service modes or with the Centaur or the Transtage. It is entirely possible that the Transtage could be used in the space servicing concept if it were operated in conjunction with a SEPS.

In order to obtain a good comparison of the candidate upper stages, it was necessary to extend the relatively short orbital lifetime of the Centaur in order to be compatible with the seven-day life of the FCT. Data available from the USAF/SAMSO upper stage assessment of November 1973 was used to synthesize a seven-day version of the vehicle. Additions were made to the fuel cell cryogens, the avionics package, pressurant gas, and the data management system. The attitude control system was changed to incorporate the FCT system and additional propellant added for a seven-day mission. No provisions were made for additional insulation of the main propellant tanks; however, the boiloff rate was reduced since General Dynamics Corporation personnel had estimated that it could be cut (from 8.6 kg (19 lb)/hr to approximately 5 kg (11 lb)/hr) by a minor redesign to further impede thermal leakage between the oxidizer (LOX) and fuel (LH_2) tanks. The lower boiloff rate was used to perform the analysis.

With the storable propellant Transtage, excessive propellant boiloff during the extended missions was not a problem. However, the added power and attitude control capability required for the seven-day mission necessitated certain changes. Batteries were replaced by fuel cells and the ACS hardware was modified to provide additional propellant. In the analyses, the Transtage was used either as an expended stage delivering satellites to orbit, or as a recoverable first stage of a multistage vehicle using an expendable Kick stage for payload deployment. The Transtage was not analyzed as a candidate vehicle for space servicing since it provided no payload return capability. It could, however, have been tried in the expendable mode for the space servicing option that leaves the failed modules and the service unit in orbit.

The vehicles were compared utilizing both the expendable satellite concept and the three space service options. Previous comparisons of the expendable satellite concept and the space service concept were based on the space service option that returns the failed modules and the service unit. The primary reason for returning the modules is to refurbish and reuse them, thereby reducing the replacement module cost. However, since no benefits from module refurbishment and reuse were applied in the analysis, a better comparison might be obtained by comparing the space service option that leaves the failed modules in orbit and returns the service unit. One might also want to tradeoff the benefits resulting from service unit reuse versus the performance penalties associated with its return. If it could be manufactured at a reasonable cost, it would probably pay to expend it in orbit.

The values and settings of the optional program features were the same for the upper stage comparisons as they were for the servicing concept comparisons. In this way, cross comparisons could be made for various combinations of servicing concepts and upper stage applications.

Results of the analyses are presented and discussed in Section 5.0. Figures 5-1 and 5-2 summarize the results obtained from the analyses.

4. SPACE SERVICING SENSITIVITY ANALYSES

4.1 GENERAL

After investigating the feasibility of the space servicing concept, it was desirable to evaluate the sensitivity of the concept to variations in operational, vehicle, and mission model parameters. One of the parameters of greatest interest was the service unit weight. Due to the general interest of the industry in the space servicing concept, a number of preliminary service unit designs had been completed by various contractors, including The Aerospace Corporation. The designs represented considerable variation in concept, configuration, and weight. The method of module changeout employed by the service unit and the actual configuration of the service unit are important factors in the space servicing concept, but the most important from a logistics point of view is the weight of the unit. An abbreviated analysis was conducted to investigate the effect of service unit weight on the flight requirements of the three space service options.

Another parameter of considerable interest was the duration of the orbital service mission. Of particular interest was how the flight requirements of a fixed design cryogenically fueled upper stage vehicle vary as the duration of the mission is varied. The primary factor contributing to the variation in flight requirements is the boiloff of the cryogenic propellants during the mission. Other less significant factors that also vary with mission time and also affect the performance of the vehicle (such as additional weight associated with increased power, attitude control, and maneuvering requirements) were also considered in the analysis.

In considering the space serviceable satellite designs, the question arose as to the degree of modularization or replaceability that should be incorporated into the redundant configuration of a functional element. For example, should the element consist of four identical, independently replaceable modules in parallel, or should some of the redundancy be incorporated internally so that the number of replaceable modules is reduced to two? The limiting condition, of course, is where all of the redundancy is contained within a single replaceable module. An abbreviated analysis was performed to establish the relationship between flight requirements and level of replaceability for various values of satellite availability.

The final sensitivity analysis performed was one to determine the effect of varying the maximum time allotted to respond to a failure on flight requirements and satellite availability.

4.2 SENSIVITITY OF FLIGHT REQUIREMENTS TO SERVICE UNIT WEIGHT

The sensitivity of space servicing logistics to the service unit weight is really a function of two parameters. One, of course, is the weight of the service unit and the other is the payload capability of the upper stage. At the time the analysis was performed, the capability to analyze a Tug/SEPS vehicle combination had not been incorporated into the LOVES program. Therefore, the only two vehicles that were employed in the space service concept at that point in time were the Centaur and the FCT. While the actual results of the analysis will vary with the vehicle employed, the trend should be the same regardless of the vehicle used. For the analysis performed, the FCT was used as the upper stage. All three of the space service options were evaluated to get an appreciation of the sensitivity of each to the weight of the service unit.

For those cases where the service unit weight was greater than the return capability of the vehicle, neither the service unit or the failed modules could be returned from orbit. Therefore, in that region, the program was not representative of any option requiring a payload return. The only option to which the region applied was the option where the service unit and the modules were both expended in orbit. In those cases where the service unit weight was less than, but close to, the roundtrip capability of the vehicle, some of the lighter modules could be returned with the service unit or the service unit alone could be returned. However, the region was not totally representative of the option requiring both to be returned. The limit on service unit weight for a meaningful analysis for the option where the modules and the service unit are both returned is approximately 650 kg (1430 lb).

This weight is equal to the difference between the roundtrip capability of the vehicle and the weight of the heaviest module that must be transported. Similarly, the service unit weight limit for the option, returning only the service unit, is represented by the difference between the delivery capability of the vehicle (with a return payload equal to the weight of the service unit) and the heaviest module to be transported. This value was approximately 864 kg (1900 lb). Therefore, regions representing service unit weights close to, or beyond, the vehicle roundtrip payload weight are undefined regions for the options requiring any kind of payload return.

In the option where both the service unit and the modules are expended, the delivery capability of the vehicle (with no return payload) was the limiting weight for the service unit. Beyond that point, the vehicle would have to be expended to perform the mission. Since the maximum service unit weight considered in the analysis was 1136 kg (2500 lb) no problems were encountered with undefined areas for that option.

Results of the analyses are presented and discussed in Section 5.0. Figure 5-6 summarizes the results obtained from the analyses.

4.3 SENSITIVITY OF UPPER STAGE VEHICLE PERFORMANCE TO MISSION DURATION

In the analyses conducted, the two-day version of the Centaur vehicle was extended to a seven-day version so that the Centaur could be directly compared with the FCT that was designed for a seven-day mission. The rate of propellant boiloff used for the extended version of the Centaur was less than for its two-day counterpart, but it was still excessive for extended duration missions. The analysis was performed to determine the sensitivity of vehicle performance (and therefore flight requirements) to the duration of the mission.

The basic procedure used in the LOVES code to assess the capability of the upper stage is to compute the vehicle propellant margin remaining each time a new payload is put on the vehicle. In doing this, the computation proceeds in a direction opposite to the actual mission profile. In other words, the first computation made determines the propellant required for the return flight of the upper stage, while the last computation determines the propellant required for the ascent flight from parking orbit to operational orbit. In previous computations, the losses experienced during the mission were assumed to be evenly distributed throughout the mission as an average loss rate. This assumption was then accounted for by using an effective Isp based on the ratio of the quantity of propellant used to the quantity originally available. This procedure was too inaccurate when applied to a vehicle with the high boiloff rate of the Centaur.

In a revised procedure, each segment (leg) of the mission profile was considered to be a two-burn operation. The first burn was to initiate the maneuver and the second burn, after an intermediate coast period, was to terminate the maneuver. Each burn was presumed to account for one-half of the velocity increment required for the total maneuver. Since the duration of the burns was only a small percentage of the total time required for the complete segment, all of the boiloff was considered to occur during the coast phase of the maneuver. The procedure then for computing the segment propellant requirement was to determine the propellant expenditure for the terminal burn, compute the propellent loss due to boiloff during the coast phase, and then after decrementing the onboard propellant by the sum of these two, compute the propellant required to perform the initial burn.

Results of the analyses are presented and discussed in Section 5.0. Figure 5-7 summarizes the results obtained from the analyses.

4.4 SENSITIVITY OF THE FLIGHT REQUIREMENTS TO THE DEGREE OF REPLACEABLE REDUNDANCY

The question of the degree of redundancy required in the space serviceable satellites and the manner in which the redundancy should be configured was a question raised many times in the design process. When the computer program development reached the point that this type of analysis could be performed, an arbitrary mission model was synthesized to investigate the problem. The mission model consisted of five systems, each of which contained nine satellites for a total of 45 satellites. The ninth satellite in each system was considered to be an active spare. Each satellite was composed of one nonreplaceable unit (NRU) and three functional elements designed for space servicing. The weight of each of these elements was 273 kg (600 lb) making the total satellite weight equal to 1090 kg (2400 lb).

The functional elements were considered to be series elements (as was the NRU); however, they were themselves presumed to contain triple redundancy.

For purposes of analysis, the redundancy was configured in three different ways (Fig. 4-1). In the first case, the redundancy was considered to be completely internal to the functional element, with the element consisting of a single space replaceable module. In the second case, the functional element was presumed to consist of two space replaceable modules, functionally in parallel, with each having a single level of internal redundancy. In the third case, the functional element was presumed to consist of four space replaceable modules, functionally in parallel, with no internal redundancy within the modules. Equivalent reliability parameters for the three versions of the space replaceable modules were analytically computed and supplied as input to the computer simulation. The capacity of the service unit was adjusted for each of the configurations analyzed to provide an equivalent capacity for each of the three configurations.

An early version of a FCT was used as the upper stage logistics vehicle since it provided an equivalent roundtrip capability for all three module configurations. With the weight of the service unit subtracted, there remained a roundtrip capability to synchronous orbit of approximately 1410 kg (3100 lb). This vehicle could accommodate five of the single module elements, 10 of the two module elements, and 20 of the four module elements. In each case, the total weight of modules that could be accommodated was 1364 kg (3000 lb).

The replacement philosophy applied in the analysis was to provide for module replacement only when the module had failed in service. No action was to be initiated in the event of a warning. Since the single module functional element contained all of its redundancy internally, no service action could be initiated until the satellite itself had failed. In order to compare configurations on an equal basis, an option in the program was set to preclude scheduling a service flight for any module until the last module in the redundant set had failed, thus causing the satellite to fail. However, for those functional elements comprised of two or more replaceable modules, the modules were available for transport on other scheduled flights as payloads of opportunity.

One of the objectives of the analysis was to compare the logistics requirements generated by the three configurations on the basis of equal availability and to do it in a more exact sense than had been done previously.

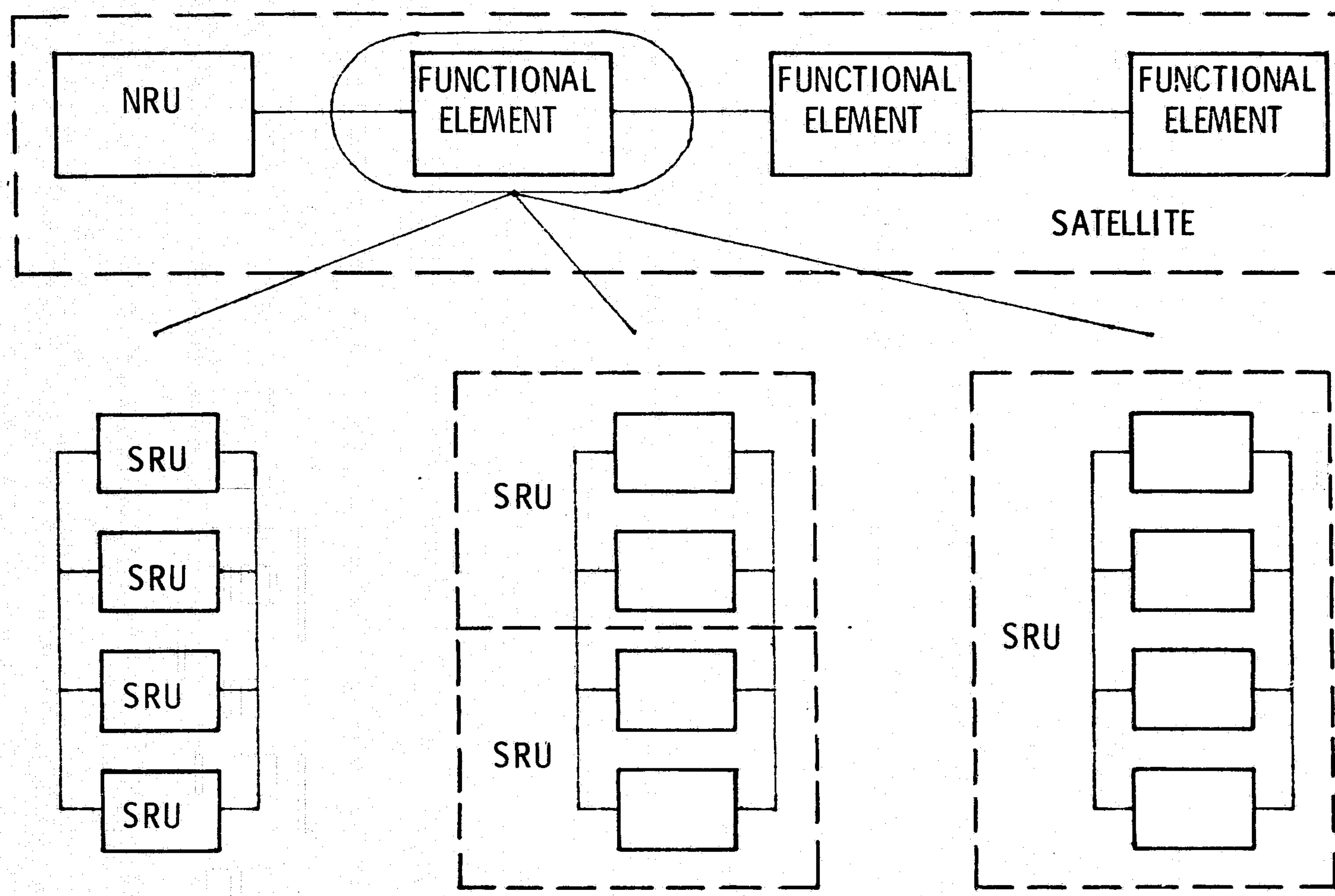


Figure 4-1. Levels of Replaceability in Element Containing Redundancy

To do this some parameter had to be varied in order to obtain a set of values for system availability for each condition investigated. The operational parameter that was varied to obtain the set of system availabilities required was the delay incurred between the time of failure and the time a replacement flight is launched. Plots of system availability versus various logistics requirements were then constructed for each of the three functional element configurations. With these plots in hand, it was then possible to construct curves of module configuration versus logistics requirements with availability as a parameter.

Results of the analyses are presented and discussed in Section 5.0. Figure 5-8 summarizes the results obtained from the analyses.

4.5 SENSITIVITY OF FLIGHT REQUIREMENTS AND AVAILABILITY TO MAXIMUM FAILURE RESPONSE TIME

During the previous analysis investigating the level of replaceable redundancy, the maximum response time to deliver a mandatory payload was varied to obtain different levels of satellite availability. A similar investigation was conducted using the geosynchronous mission model rather than the synthesized model used in the response time investigation. The time was varied from 30 to 90 days in the analysis with all other parameters being held constant.

Results of the analyses are presented and discussed in Section 5.0. Figure 5-9 summarizes the results obtained from the analyses.

4.6 SENSIVITIY OF FLIGHT REQUIREMENTS AND AVAILABILITY TO NUMBER OF SPARE MODULES PERMITTED IN QUEUE

During some of the previous analyses, it appeared that more spare modules were being replaced than needed to be, resulting in unnecessary hardware procurement. Some of the satellite designs contained elements having a large number of spare, redundant modules, all of which might not need to be replaced to maintain satellite availability. To investigate the situation, a series of runs were made limiting the number of spare modules that could be enetered into the waiting queue. The number permitted to enter was changed for successive runs to obtain the necessary data.

Results of the analyses are presented and discussed in Section 5.0. Figure 5-10 summarizes the results obtained from the analyses.

5. RESULTS/CONCLUSIONS

The primary conclusions to be drawn from the analyses that were performed are:

1. The capability of the LOVES computer code to analyze space programs in general and space serviceable programs, in particular, was successfully demonstrated. The flexibility built into the program in the form of various options enhances its basic capability to conduct a variety of parametric and sensitivity studies.
2. The concept of space servicing of automated payloads, especially those stationed in synchronous equatorial orbit, is not only feasible but also appears economically rewarding when compared to the continued use of the expendable satellite replacement concept.

The results of a tradeoff between the space servicing concept and the expendable satellite replacement concept indicated a clear advantage for the space servicing concept. The number of flights required and the number of equivalent payloads procured were both significantly reduced as can be seen in Figures 5-1 and 5-2. The number of upper stages required in the space servicing case was greater, however, because of the intial deployment of the heavier satellites that required expending a vehicle. Using the simplified cost analysis previously discussed, the space serviced concept appeared to offer considerable cost benefit over the expendable satellite concept. This cost comparison, however, did not include the DDT&E costs required to develop the space servicing technology.

In all cases having the same upper stage and operational constraints, the concept of space servicing proved to be more economical than its expendable satellite counterpart, in spite of the heavier satellite weights associated with the space serviceable design. In only one case was the use of expendable satellites shown to be superior to the use of space serviced satellites. In that case, the FCT was utilized as the upper stage in the expendable case and the Centaur in the space serviceable case. The space service option in question was the most demanding of the three options, i.e., the one that requires the

GEOSTATIONARY ORBIT (1980 - 1990)

PAYLOAD POLICY	OPTIONS			RESULTS			APPROX BENEFITS		
	STAGE	EXP	REC	FLTS	PL PROC	EXP STG	Δ FLTS	Δ P/L	Δ COST $M
REFERENCE (EXPEND PLS)	TRANS/KICK		✓	83	96	19	-	-	0
EXPEND PLS	TRANSTAGE	✓		47	96	47	36	0	228
EXPEND PLS	CENTAUR		✓	78	96	0	5	0	153
EXPEND PLS	FULL CAP TUG		✓	50	96	0	33	0	433

REFERENCE COSTS:

EXPENDED CENTAUR STAGE	$ 8M	EXPENDED KICK	$0.1M
EXPENDED FULL CAP TUG	$10M	SHUTTLE /UPPER STAGE FLIGHT	$10M
EXPENDED TRANSTAGE	$ 5M	AVERAGE SATELLITE COST	$10M

NO COST ASSIGNED TO SERVICE UNIT

Figure 5-1. Upper Stage Comparisons - Expendable Satellite Service Concepts

	PAYLOAD POLICY	OPTIONS			RESULTS			APPROX BENEFITS		
		STAGE	EXP	REC	FLTS	PL PROC	EXP STG	FLTS	PL	COST $M
	REFERENCE EXPENDABLE	TRANS/KICK			83	96	19	-	-	0
1	SPACE SERVICE	CENTAUR			87	65	10	-4	31	293
	0 RETRIEVE SRUs	FULL CAP TUG			59	65	2	24	31	633
	0 RETRIEVE SU	FULL CAP TUG/SEPS			47/39	65	0	36	31	773
2	SPACE SERVICE 0 EXPEND SRUs 0 RETRIEVE SU	CENTAUR			73	65	10	10	31	433
		FULL CAP TUG			53	65	2	30	31	693
3	SPACE SERVICE 0 EXPEND SRUs 0 EXPEND SU	CENTAUR			66	65	10	17	31	503
		FULL CAP TUG			50	65	2	33	31	723

SUMMARY: INITIAL BENEFIT FCT ---------------- $433M
SPACE SERVICING △ 1 -------------- 200M
2 -------------- 260M
3 -------------- 290M
SP SERV/SEPS △ 1 -------------- 340M

Figure 5-2. Upper Stage Comparisons - Space Service Concept return of the service unit and the spent replaceable modules. For that option with its very demanding performance requirements, the high boiloff rate of the Centaur proved to be too much of a handicap for the space service operation to overcome.

For comparable conditions, the space service concept shows cost benefits over its expendable satellite counterpart ranging from $150M to $350M over an 11-year period. The exact figure depends on the upper stage employed and on the space service option selected. If one takes into consideration the additional programs not included in the synchronous orbit sample and the fact that the space program will be an ongoing effort, the savings will be much greater than the figures indicated. These amounts of money would more than compensate for the added expenditure required to develop a space service capability.

If operations employing the Transtage/Kick combination are taken as the reference operation for the expendable satellite concept, a space service concept utilizing the Centaur upper stage could provide savings rangings from $290M to $500M. The FCT could provide benefits ranging from $635M to $725M. The limited SEPS analysis indicates that utilizing the SEPS in conjunction with the FCT could provide at least another $150M in benefits that would result in a minimum savings of approximately $785M.

The lower figures quoted in the above spreads apply to the space service option requiring the return of the service unit and the spent modules. The higher values refer to the option that returns nothing from orbit. If the space replaceable modules and the service unit could be left in orbit, substantial additional savings could be realized. Additional savings of $90M to $210M are indicated depending on the upper stage being utilized.

The savings that can be realized using the space servicing concept is also dependent upon when the concept is implemented. Another Aerospace study investigating the feasibility of a pilot program to demonstrate the space servicing technique (Ref 6) found that considerable savings would be sacrificed if the implementation of space servicing were delayed for any appreciable time. With the new starts shown in Figure 5-3, the ability to "capture" them for the space

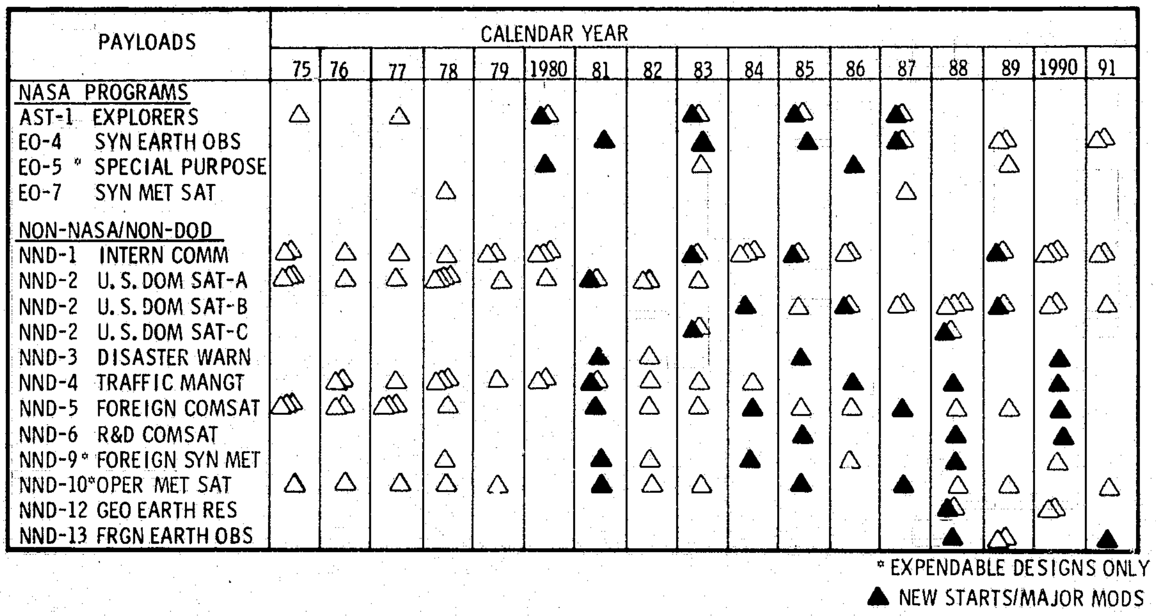


Figure 5-3. NASA and Domestic Geostationary Programs

servicing concept depends on when the capability is developed. Figure 5-4, which integrates the new starts with time, indicates that if space servicing becomes operational with the introduction of the FCT in the 1984 time frame, over 70 percent of the new starts could be captured. As time progresses, more and more of the new starts are "lost" with the subsequent cost penalties shown in Figure 5-5.

Given the space servicing concept, the effect of service unit weight on the flight requirements is shown in Figure 5-6. For the service option where both the service unit and the replaced modules are expended, the effect appears quite linear in the range of weights investigated. At first glance, the penalty incurred because of the increased service unit weight appears to be quite low. The penalty is on the order of one additional flight for every 114 kg (250 lb) of additional weight. However, with the cost of a shuttle/tug flight reported to be in the neighborhood of $10M, the penalty in terms of dollars per unit weight turns out to be approximately $88K per kg (2.2 lb). This is not an insignificant penalty. In the two options requiring payload return, the penalty is much worse. Instead of being linear, the curves exhibit an exponential characteristic and have a relatively high initial slope. In the region between

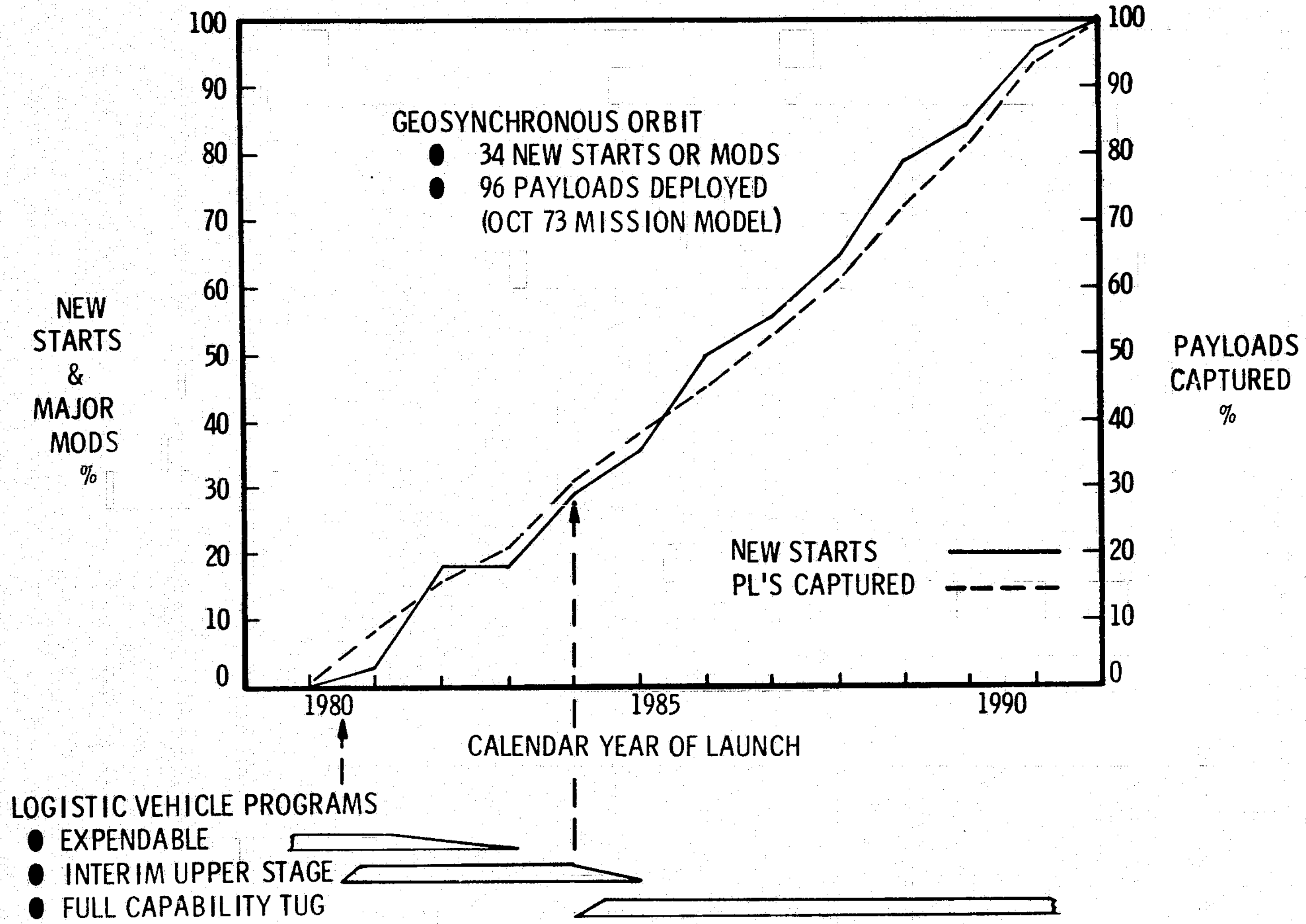


Figure 5-4. Major Payload Program Events

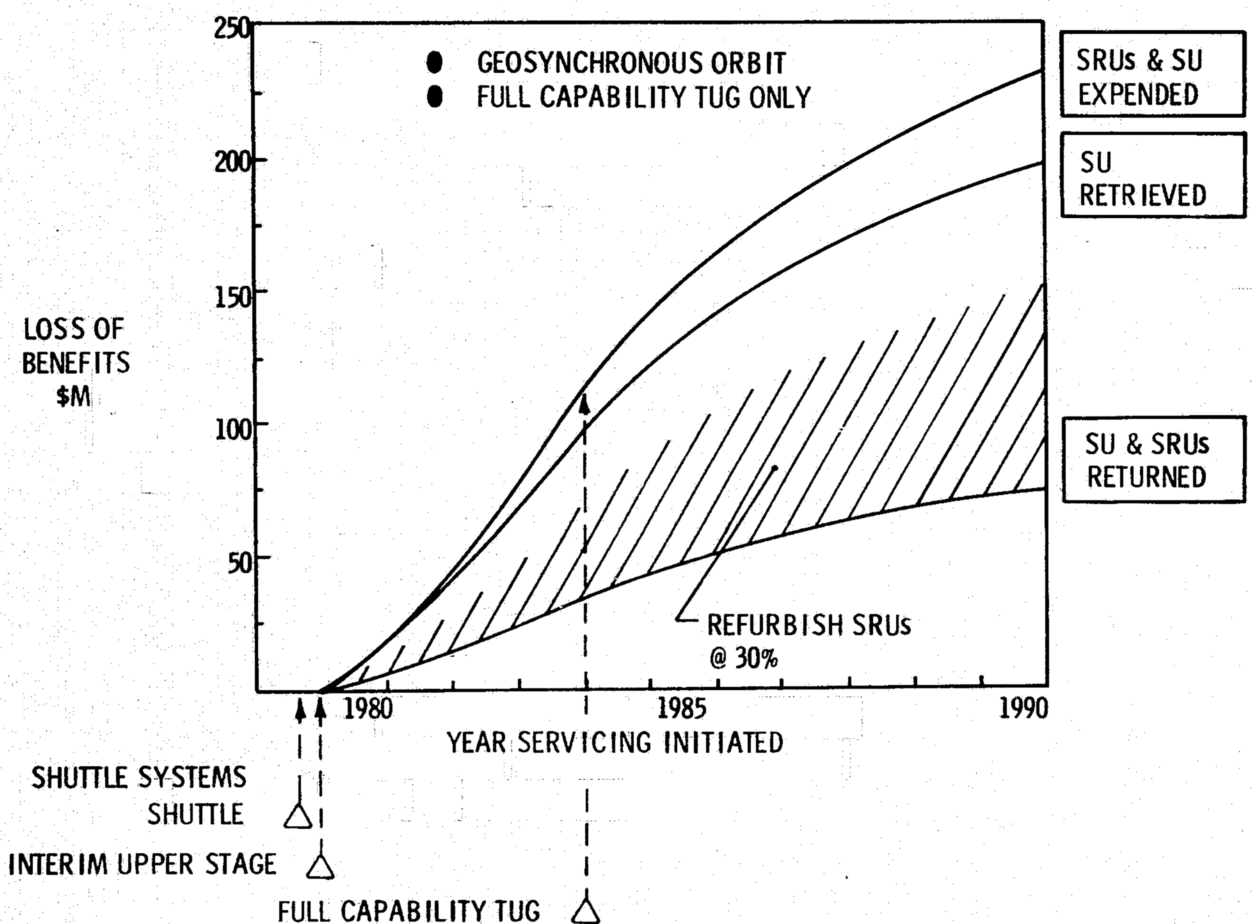
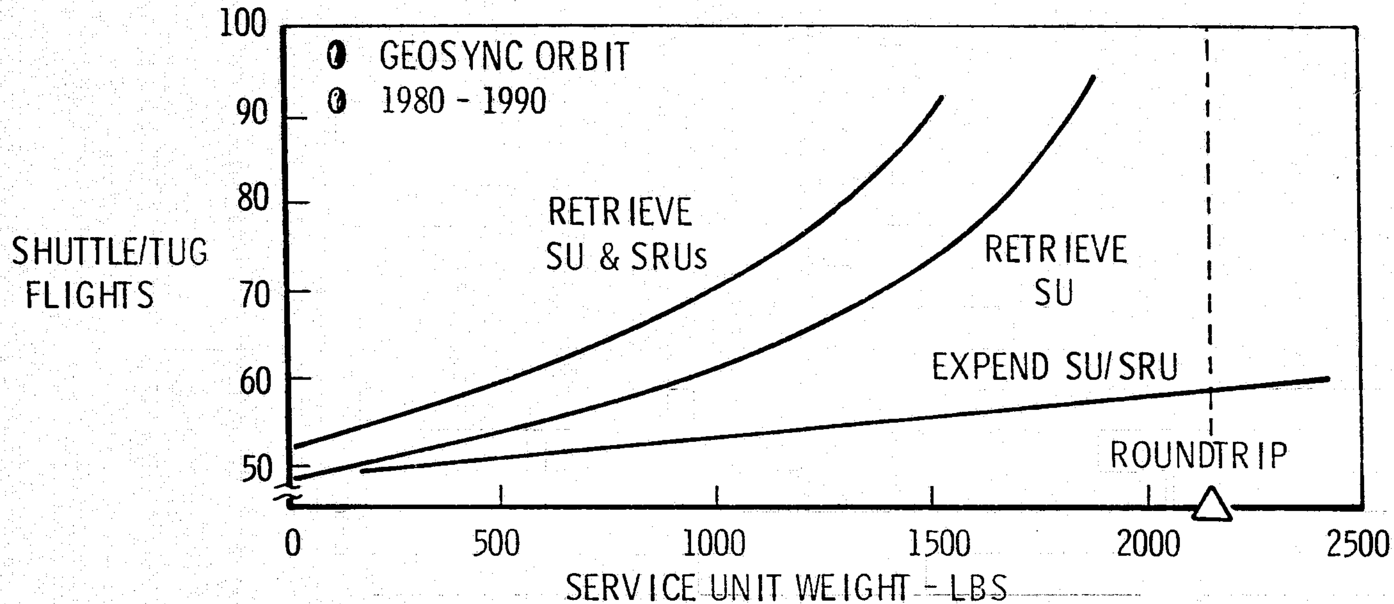


Figure 5-5. Cost Impact of Delaying Space Servicing



SERVICE UNIT WT. (LBS)	INCREMENTAL COST IMPACT ($M)		
	EXPEND SU/SRUs	RETRIEVE SU	RETRIEVE SU/SRUs
200			
	$8M	$13M	$25M
400			
	$8M	$17M	$30M
600			
PARTIAL	$40K/LB	$85K/LB	$150K/LB

Figure 5-6. Impact of Service Unit Weight on Flight Requirements

91 kg (200 lb) and 273 kg (600 lb) the curvature is small and a reasonably accurate linear approximation can be obtained. Beyond 273 kg (600 lb) the curvature increases too rapidly to make accurate linear approximations over any reasonable range of weights. In the 182 kg (400 lb) range discussed above, the average penalty in terms of dollars per unit weight, for the service option returning over the service unit, is approximately $187K per kg (2.2 lb). For the service option returning both the service unit and the failed modules, the average penalty is almost double that figure, approximately $330K per kg (2.2 lb). Beyond the 273 kg (600 lb) range, the cost per unit weight for the two service options requiring payload return increases very rapidly and soon becomes too extravagent to be practical.

In a space serviced program utilizing the FCT as the upper stage vehicle, it appears that the target weight of the service unit should be kept within the range of 91 kg (200 lb) to 273 kg (600 lb).

The sensitivity of the Centaur flight requirements to the duration of the mission and to the space service option being employed is presented in Figure 5-7. For the service option that expends the service unit and the SRUs in orbit, the effects of varying the mission duration appears to be minimal (a matter of only a flight or two). However, for the service options requiring the return of a payload the effects are more dramatic. For the option where both the service unit and the SRUs must be returned, the maximum variation in the number of flights required was 17. In terms of dollars, this is approximately $170M. The variation for the other service option requiring the service units return was not as significant, but still approached $100M. As interesting as the indicated variation in cost was, the indication of a minimum value (optimum mission duration) for each option was even more so. The minimum apparently exists because of the tradeoff between the increased velocity requirements associated with short mission durations and the excessively high boiloff resulting from long mission durations. The optimum mission duration for the extended version of the Centaur appears to vary between four and five days depending on the service option with the more demanding options indicating shorter times.

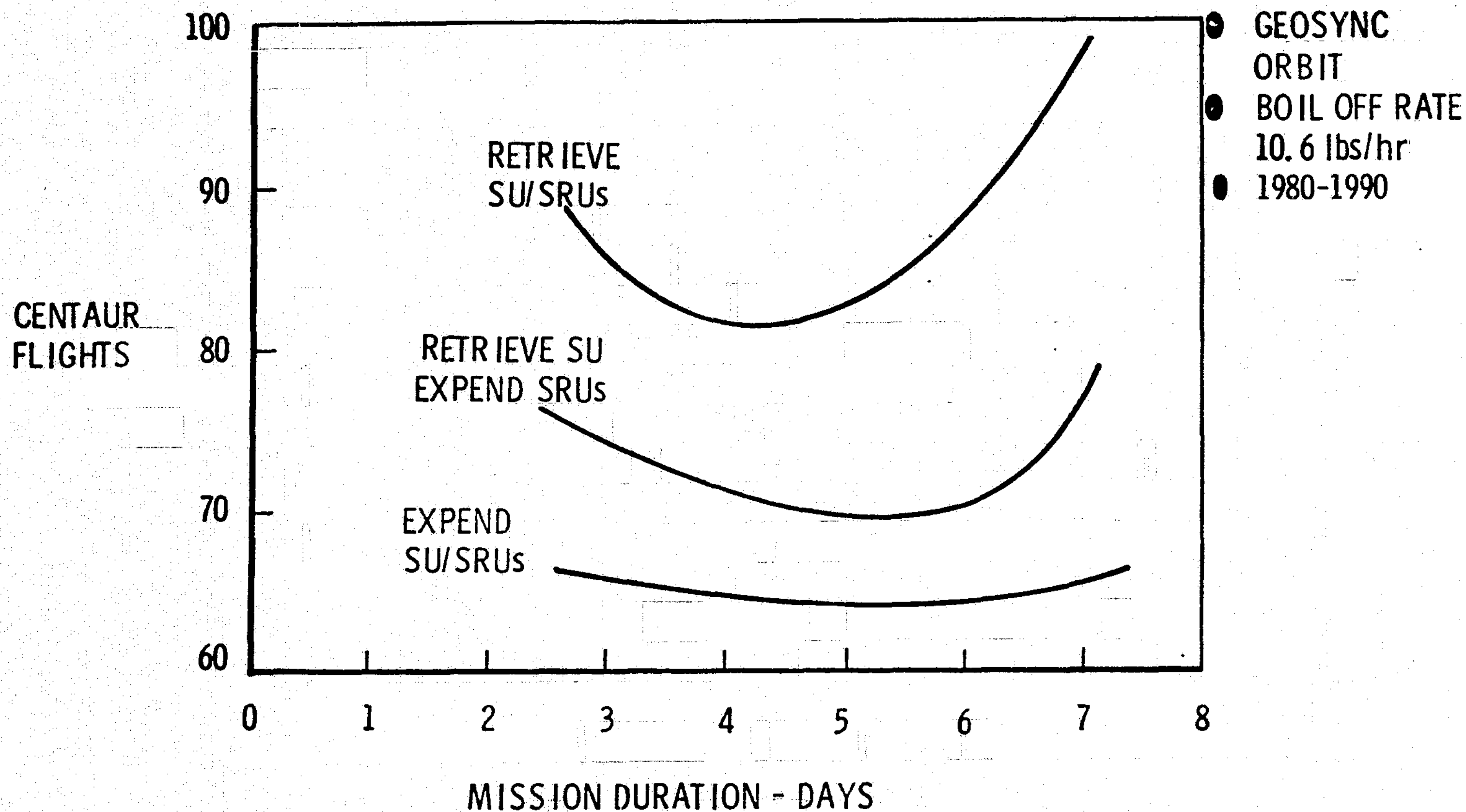


Figure 5-7. Impact of Mission Duration on Flight Requirements

The results of the investigation on the degree of module replaceability for functional elements with redundant circuits is shown in Figure 5-8. The figure presents the effects of replaceability on the most important cost bearing logistics element, the number of vehicle flights required. The configuration containing four replaceable modules per functional element required the least number of flights and the least number of satellites to support the mission model. The greater flexibility for remedial action provided by the increased number of replaceable modules makes the four module configuration the preferred configuration where high availabilities are required. It is also evident that the lower the availability requirement becomes, the less important is the manner in which the redundancy is configured.

Results of the analysis investigating the effects of varying the maximum response time to a failure requiring a mandatory replacement module are presented in Figure 5-9. As can be seen from the figure, both the satellite availabilities and the flight rate are strong functions of the response time. If the module failure requiring a mandatory replacement also causes the satellite to fail, the only recourse for maintaining a respectable system availability with the long response times would be to have a spare satellite in the system. The low availabilities of the individual satellites comprising the system would preclude obtaining a reasonable system availability without the spare. If, however, a mandatory replacement is required when the last (or next to last) spare module fails, reasonably good availabilities can be obtained without a spare satellite in the system even with the longer response times applied.

Figure 5-10 presents the results of the analysis conducted to assess the effect of limiting the number of spare module replacements that are put in the waiting queue. The results show that after two spare modules, very little effect, if any, is discernable. In fact, after the first spare module, very little is gained by replacing additional spare modules. This does not mean that the additional spare modules contribute nothing to the reliability of the satellite (and therefore its availability). It does imply, however, that once their initial contribution to satellite availability has been realized, it is not necessary to replace them upon failure.

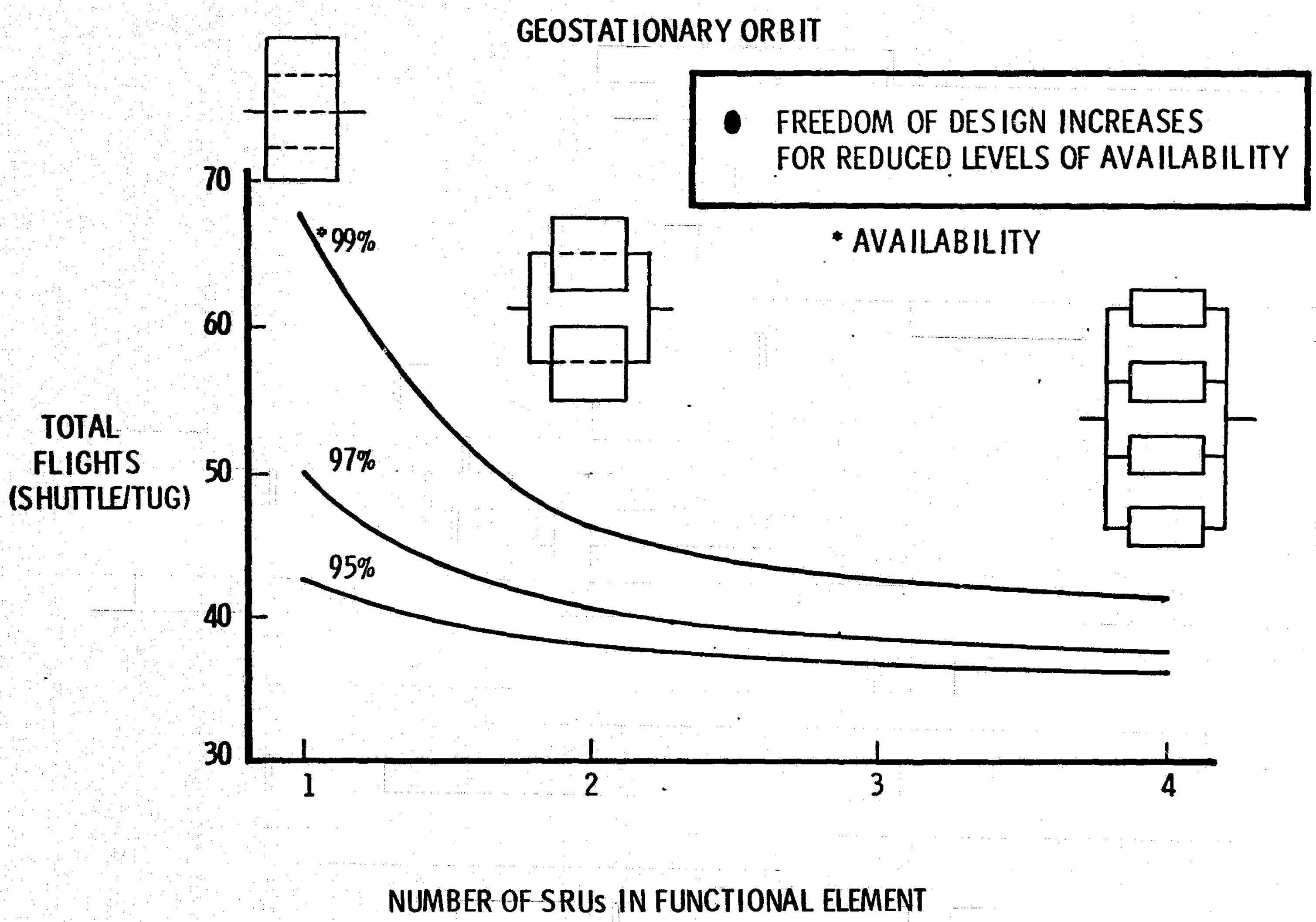


Figure 5-8. Impact of Level of Replaceability in Element Containing Redundancy

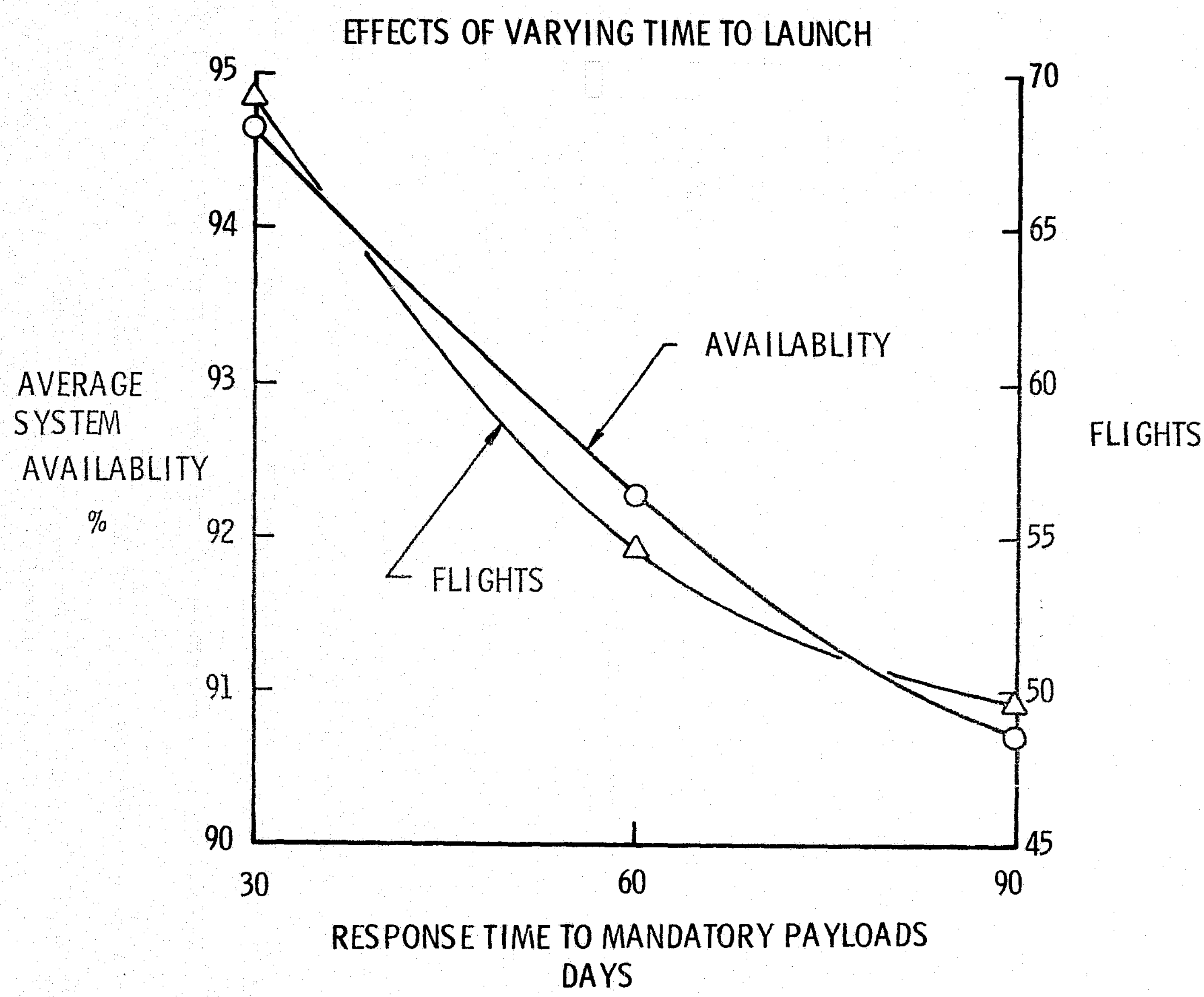


Figure 5-9. Impact of Response Time to a Failure

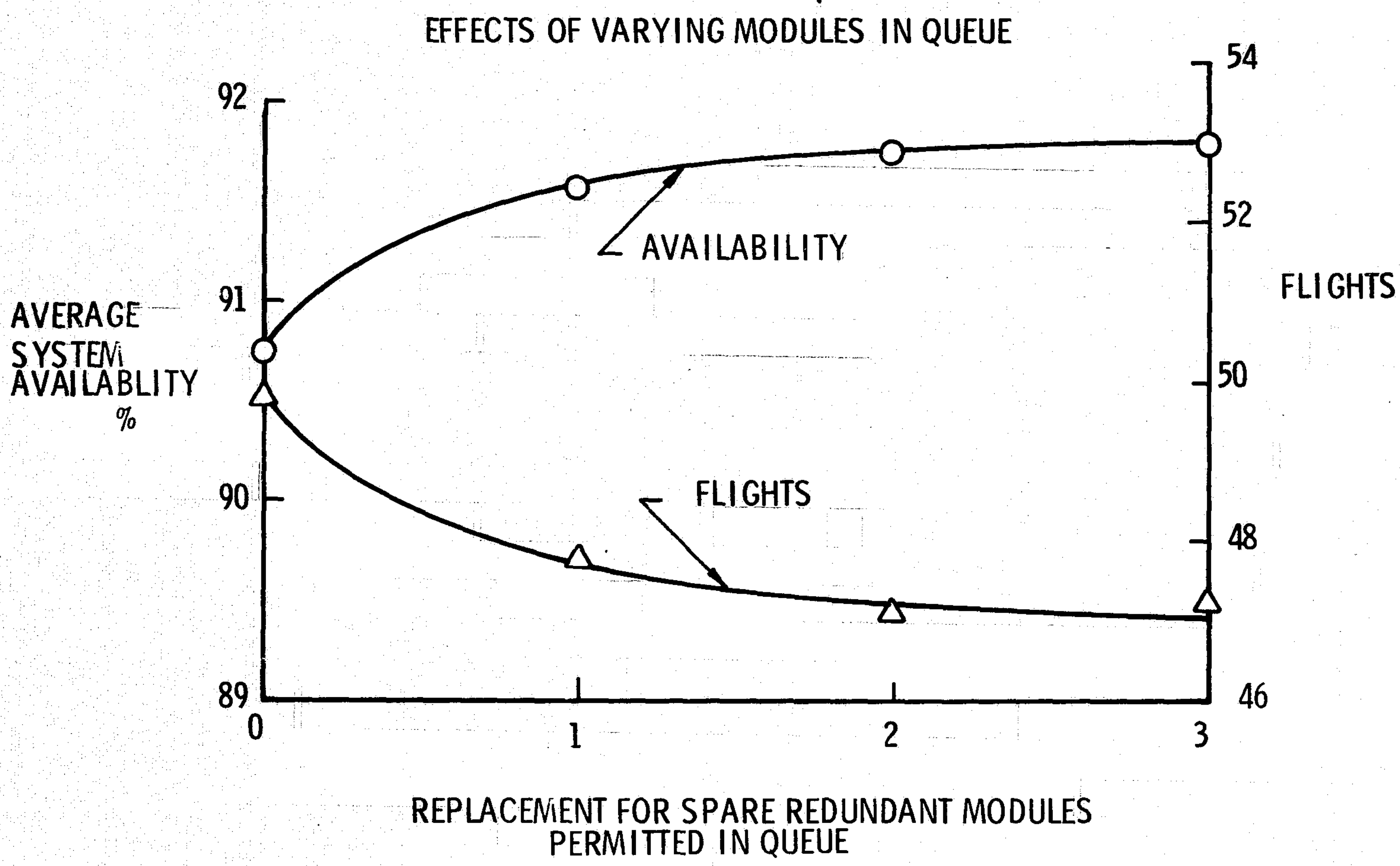


Figure 5-10. Impact of Excluding Spare Module Replacements from Queue

The concept of space servicing automated satellites with replaceable equipment modules appears to provide several advantages over the present practice of replacing entire satellites when failures occur. The most obvious is the cost savings that is accrued, especially when the space program is projected to continue indefinitely in the future. Another benefit derived from the space servicing concept is the increase in satellite availabilities obtained because the satellites are serviced prior to failure, in most cases. The opportunity to fly replacements for spare modules as payloads of opportunity provide this benefit.

The sensitivity studies of the space service concept showed, in general, that the greatest factor affecting satellite availability was the time required to respond to a failure. As would be expected, the time also greatly influenced the flight requirements for the model. They also showed that the effect of the spare modules included in a redundant set of modules on the satellite availabilities dropped off drastically after one or two were incorporated in the design. The degree of replaceability of the redundant elements, however, was of major importance if high satellite availabilities were to be maintained.

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APPENDIX A

OCTOBER 1973 NASA AUTOMATED PAYLOAD MODEL

This appendix contains the detailed information on the NASA's 1973 Automated Payload Model that is a part of the larger overall 1973 Payload Model (Ref. 1).

Table A-1 contains data on the various satellites' expected lifetimes, on-orbit schedules, and orbital characteristics. The on-orbit schedules were used to establish initial launch dates for both the expendable and the space serviceable satellites. If the on-orbit schedule indicated continuous coverage for any period of time by the satellite or satellite system, the required number of satellites were launched to the deterministic schedule only to initiate the system. Subsequent launches were made only to service the random failures as they occurred in the simulation. If the coverage was discontinuous with breaks between finite periods of continuous service, deterministic launches were scheduled on the dates initiating each of the service periods. Random failures again determined the launch schedule during the period of continuous service.

Table A-2 is an example of the data supplied in Reference 3 for the expendable versions of the satellites contained in Table A-1. The primary use of this data was to obtain the launch weights of the expendable versions of the satellites. The satellite described in Table A-2 is one of the geosynchronous satellites contained in the abbreviated model used in the analyses that were performed.

Table A-1. 1973 NASA Automated Payload Model Astronomy (AST)

Revised: 25 January 1974

P/L CODE: MISS MODEL	P/L CODE: SSPDA	PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC: ALTITUDE (km)	ORBIT CHARACTERISTIC: INCLINATION (deg)	P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	80	81	82	83	84	85	86	87	88	89	90	91
		EXPLORER																				
AST-1A	AS-02	Extra Coronal Lyman Alpha	EXP	(1)	297 ± 18	28.5 ± 0.5	(1)	1	3	14	1/2	2	1/1	1	1/1	1	1/1	1	1/1	1	1/1	1
-1B	AS-03	Cosmic Background	EXP	ANY	400 ± 100	ANY	ANY	1	2	14	0/1	1/0	1	1/0	1	1/0	1	1/0	1	1/0	1	1/0
-1C	AS-05	Adv. Radio Astronomy (2)	EXP	ANY	SYNC ± 37	0 ± 5	0 ± 5	2	3	10	2/0	2	2			2/0	2	2				
-1D	AS-05	Adv. Radio Astronomy (2)	EXP	ANY	SYNC ± 37	0 ± 5	80W ± 20	2	3	10				2/0	2	2		2/0	2	2		
AST-3	SO-03	SOLAR PHYSICS MISSION	GS	ANY	$500 \pm {}^{0}_{130}$	$30 \pm {}^{2}_{30}$	ANY	1	2	13	R		R		R			R	R			R
AST-4	HE-09	HIGH ENERGY ASTR. OBS-MagSpec.	OS/GS	ANY	370 ± 19	$28.5 \pm {}^{6.5}_{5.5}$	ANY	1	1	5												
AST-5A	HE-03	HIGH ENERGY ASTR.- Ext. X-Ray	OS/GS	ANY	370 ± 19	$28.5 \pm {}^{1.5}_{0.5}$	ANY	1	1	5												
		REVISITS												1	1	1						
-5B	HE-08	HIGH ENERGY ASTR. Gamma Ray	OS/GS	ANY	370 ± 19	$28.5 \pm {}^{0}_{13.5}$	ANY	1	1	5												
		REVISITS																1	1	1		
-5C	HE-10	HIGH ENERGY ASTR. Nuclear Cal.	OS/GS	ANY	370 ± 19	28.5 ± 0.5	ANY	1	1	5								R				
		REVISITS																	1	1	1	
-5D	HE-05	HIGH ENERGY ASTR. Cosmic Ray	OS/GS	ANY	370 ± 19	28.5 ± 0.5	ANY	1	1	5												R
AST-6	AS-01	LARGE SPACE TELESCOPE	MT/GS	ANY	612 ± 19	$28.5 \pm {}^{6.5}_{0}$	ANY	1	1	15				R					R			
		REVISITS										1	1		1	1	1	1		1	1	1
AST-7	SO-02	LARGE SOLAR OBSERVATORY	MT/GS	ANY	$350 \pm {}^{50}_{30}$	$30 \pm {}^{2}_{30}$	ANY	1	1	15												
		REVISITS															1	1	1	1	1	1
AST-8	AS-16	LARGE RADIO OBSERVATORY	OS	ANY	71600 ± 1000	28.5 ± 0.5	ANY	1	2	6												
		REVISITS																1		1		1
AST-9A	HE-11	FOCUSING X-RAY TEL.-1.2M	OS/GS	ANY	500 ± 19	$15.0 \pm {}^{13.5}_{15.0}$	ANY	1	1	10												R
		REVISITS													1	1						
-9B	HE-01	FOCUSING X-RAY TEL.- 3.0M	OS/GS	ANY	500 ± 19	$15.0 \pm {}^{13.5}_{15.0}$	ANY	1	2	10												
		REVISITS																	1		1	

Note:

(1) 21 Sept 1980 @ 1800 LAUNCH DATE INTO PARKING ORBIT FOR FINAL HELIOCENTIC ORBIT

(2) TWO SATELLITES DEPLOYED 20KM to 200 KM APART TO FORM INTERFEROMETER BASELINE

Table A-1. 1973 NASA Automated Payload Model Space Physics (PHY)

Revised: 25 January 1974

P/L CODE MISS MODEL	P/L CODE SSPDA NO.	PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC ALTITUDE (km)	ORBIT CHARACTERISTIC INCLINATION (deg)	P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	80	81	82	83	84	85	86	87	88	89	90	91
		EXPLORER																				
PHY-1A	HE-07	Small High Energy Observatory	EXP	ANY	371 ± 19	$28.5^{+0}_{-13.5}$	ANY	1	1	8		1/0			1/0				1/0	1/0	1/0	1/0
-1B	AP-01	Upper Atmosphere	EXP	ANY	259/3510 (1)	90 ± 20	ANY	1	1	13		1/0				1/0				1/0		
-1C	AP-02	Medium Altitude	EXP	ANY	1852/37038 (2)	$28.5^{+61.5}_{-28.5}$	ANY	1	1	13				1/0				1/0				1/0
-1D	AP-03	High Altitude	EXP	ANY	1 A.U.	Ecliptic	ANY	1	1	13	1/0		1/0		1/0		1/0		1/0		1/0	
		GRAVITY AND RELIABILITY SAT.																				
PHY-2A	AP-04	Earth Orbit	EXP	ANY	938 ± 62	90 ± 0.04	ANY	1	1	5	1/0			1/0								
-2B	AP-06	Solar	EXP	ANY	0.3/1.0 A.U.	Helio	ANY	1	1	7							1/0					1/0
		ENVIRONMENTAL PERTURB. SAT.																				
PHY-3A	AP-05	Satellite A	EXP	ANY	12,778 ± 926	55 ± 30	ANY	1	3	6		1/0	1	1	1/0	1	1					
-3B	AP-07	Satellite B	EXP	ANY	12,778 ± 926	55^{+35}_{-0}	ANY	1	3	6								1/0	1	1	1/0	1
PHY-4	AP-08	HELIO & INTERSTELLAR S/C	EXP	(3)	Escape (3)	(3)	ANY	1	7	7									1/0	1	1	1
PHY-5	HE-12	COSMIC-RAY LABORATORY	MT	ANY	371 ± 19	$28.5^{+26.5}_{-0.5}$	ANY	1	1	10								1 →	→	→	→	→
		REVISITS																	1	1	1	1

Note:

(1) ± 74 and ± 180 km

(2) ± 370 and ± 9200 km

(3) ESCAPE TRAJECTORY (OUT OF ECLIPTIC TO NEAREST STAR)

A-3

Table A-1. 1973 NASA Automated Payload Model Planetary Exploration (PL)

Revised: 25 January 1974

P/L CODE MISS. MODEL	P/L CODE SSPDA NO.	PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC ALTITUDE (km)	ORBIT CHARACTERISTIC INCLINATION (deg)	INITIAL LAUNCH DATE (2)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	PAYLOAD LAUNCH SCHEDULE (4) 80	81	82	83	84	85	86	87	88	89	90	91
PL-7	PL-01	Surface Sample Return	EXP(1)	720	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{27.5}$	Febr 1984	1	4	4					2							
PL-8	PL-02	Satellite Sample Return (1)	EXP(1)	720	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{14.5}$	Aug 1990	1	4	4											1	1
PL-10	PL-03	Inner Pl. Follow-On	EXP	2160	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{4.5}$	Apr 15, 1980	1	1	1	1	2		1			1					
PL-11	PL-07	Venus Radar Mapper	EXP	480	$436 \pm ^{164}_{251}$	$32 \pm ^{58.0}_{3.5}$	July 1983	1	1	1				2								
PL-12	PL-08	Venus Buoyant Station	EXP	480	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{24.5}$	Febr 1985	1	1	1						2						
PL-13	PL-09	Mercury Orbiter (3)	EXP	480	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{10.5}$	Nov 1987	1	1	1								2				
PL-14	PL-10	Venus Large Lander	EXP	480	$436 \pm ^{164}_{251}$	$32 \pm ^{58.0}_{3.5}$	Ncv 1989	1	1	1										2		
PL-17	PL-22	Pioneer Saturn Probe	EXP	480	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{10.5}$	Dec 15, 1980	1	7	7	1											
PL-18	PL-11	Pioneer Sat/Uranus Flyby	EXP	480	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{10.5}$	Dec 15, 1981	1	>7	>7		1										
PL-19	PL-12	Mariner Jupiter Orbiter	EXP	720	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{27.5}$	Dec 16, 1981	1	3	3		2										
PL-20	PL-13	Pioneer Jupiter Probe	EXP	480	$436 \pm ^{164}_{251}$	$31 \pm ^{59.0}_{2.5}$	Mar 1984	1	3	3					2							
PL-21	PL-14	Mariner Saturn Orbiter	EXP	480	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{26.5}$	Jan 1985	1	7	7						2						
PL-22	PL-15	Mariner Uranus/Nep. Flyby	EXP	480	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{8.5}$	Jan 1986	1	>10	>10							2					
PL-23	PL-16	Jupiter Sat. Orbiter/Lander	EXP	480	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{8.5}$	Oct 1990	1	7	7											1	1
PL-26	PL-18	Encke Rendezvous	EXP	480	$436 \pm ^{164}_{251}$	$45 \pm ^{15}_{2}$	Feb 6, 1981	1	3	3		2										
PL-27	PL-19	Halley Flyby	EXP	480	$436 \pm ^{164}_{251}$	$34 \pm ^{56}_{3}$	June 1985	1	~3	~3						1						
PL-28	PL-20	Asteroid Rendezvous	EXP	480	$436 \pm ^{164}_{251}$	$28.5 \pm ^{61.5}_{18.5}$	June 1986	1	~3	~3							2					

Note:

(1) ORBITER RECOVERS EARTH RETURN VEHICLE
(2) MONTHS CAN BE ± ONE MONTH (DEPENDING ON S/C INSERTION/RETRO CAPABILITY)
(3) DIRECT FLIGHT TO MERCURY (NO VENUS SWINGBY)
(4) PAYLOADS CAN BE LAUNCHED ON SAME FLIGHT

Table A-1. 1973 NASA Automated Payload Model Lunar Exploration (LUN), Life Sciences (LS), Space Technology (ST)

Revised: 25 January 1974

P/L CODE: MISS MODEL	P/L CODE: SSPDA NO.	PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC: ALTITUDE (km)	ORBIT CHARACTERISTIC: INCLINATION (deg)	P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	80	81	82	83	84	85	86	87	88	89	90	91
		LUNAR																				
LUN-2	LU-01	Lunar Orbiter	EXP	2	$436 \pm ^{164(2)}_{251}$	$28.5 \pm ^{61.5}_{0}$	Trans. Lun	1	1	4					1/0		1/0					
LUN-3	LU-02	Lunar Rover	EXP	2	$436 \pm ^{164(2)}_{251}$	$28.5 \pm ^{61.5}_{0}$	Trans. Lun	1	1	1								1/0	1/0			
LUN-4	LU-03	Lunar Halo	EXP	2	$436 \pm ^{164(2)}_{251}$	$28.5 \pm ^{61.5}_{0}$	Trans. Lun	1	5	5										1/0	1	1
LUN-5	LU-04	Lunar Sample Return (1)	EXP	2	$436 \pm ^{164(2)}_{251}$	$28.5 \pm ^{61.5}_{0}$	Trans. Lun	1	1	1											1/0	1/0
		LIFE SCIENCES																				
LS-1	LS-02	Life Science Research Mod.	GS	ANY	500 ± 100	$28.5 \pm ^{10}_{0}$	ANY	1	0.5	12	2/0	2/0	2/0	2/0	2/0	2/0	2/0	2/0	2/0	2/0	2/0	2/0
		SPACE TECHNOLOGY																				
ST-1	ST-01	Long Duration Exposure Mode.	GS	ANY	500 ± 50	$28.5 \pm ^{26.5}_{28.5}$	ANY	1	3 (3)	11			R		R		R		R		R	

Note:

(1) ORBITER RECOVER EARTH RETURN SAMPLE

(2) TRANSFER ORBIT

(3) 6 to 9 MONTHS DURATION ON-ORBIT

Table A-1. 1973 NASA Automated Payload Model Earth Observations (EO)

Revised: 25 January 1974

P/L CODE: MISS MODEL	P/L CODE: SSPDA	PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC: ALTITUDE (km)	ORBIT CHARACTERISTIC: INCLINATION (deg)	P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	80	81	82	83	84	85	86	87	88	89	90	91
EO-3A	EO-8	Earth Observatory Satellite-Res[1]	OS	0.25	914 ± 9	99.15 ± 0.10	9:00 [2]	2[3]	2	12				R						R		
		Revisit														1						1
-3B		Earth Observatory Satellite-Met[1]	OS	0.25	914 ± 9	99.15 ± 0.10	12:00 [2]	2[3]	2	12						R						R
		Revisit										1						1				
-3C		Earth Observatory Satellite-All Wea[1]	OS	0.25	914 ± 9	99.15 ± 0.10	3:00 [2]	2[3]	2	12		R						R				
		Revisit												1						1		
-3D		Earth Observatory Satellite-Test[1]	OS	0.25	914 ± 9	99.15 ± 0.10	9:00 [2]	1	2	2	1/0	1										
EO-4A	EO-9	Sync. Earth Obs. Satellite - RD	EXP	ANY	SYNC ± 46	0 ± 0.20	96 W	1	2	11		1/0	1	1/0	1	1/0	1					
-4B		Sync. Earth Obs. Satellite - OPER	EXP	ANY	SYNC ± 46	0 ± 0.20	96 W	1	2	11								2/0	2	2/0	2	2/0
EO-5A	EO-10	Special Purpose Satellite - Sync.	EXP	ANY	SYNC ± 46	0 ± 0.60	80 to 120 W	1	2	10	1/0	1					1/0	1				
-5B		Special Purpose Sat. - Polar 3000	EXP	0.50	5500 ± 30	150 ± 0.50	9:00[2]	1	2	5	1/1	1										
-5C		Special Purpose Sat. - Polar 280	EXP	0.25	500 ± 10	97.8 ± 0.10	15:00[2]	1	2	13		2/0	2		1/0	1		1/0	1		1/0	1
-5D		Special Purpose Sat. - Polar 400	EXP	0.25	750 ± 10	98.8 ± 0.10	9:00[2]	1	2	13			1/0	1		1/0	1		1/0	1		1/0
-5E		Special Purpose Sat. - Sync.	EXP	ANY	SYNC ± 46	0 ± 0.60	80 to 120 W	1	2	11				1/0	1					1/0	1	
EO-6	EO-12	TIROS	EXP	0.33	1460 ± 40	102 ± 0.06	9:00[2]	1	2	2			1/0	1				1/0	1			
EO-7	EO-7	Sync. Meteorological Satellite	OS	ANY	SYNC ± 46	0 ± 0.10	96 W	1	5	5	0/1	1	1					1/0	1	1	1	1

(1) Schedule revised from October Space Opportunities and SSPDA documents

(2) Assumed Nodal Crossing Times

(3) Number of payloads operating simultaneously with other Nodal Crossing Time EOS's

Table A-1. 1973 NASA Automated Payload Model Earth and Ocean Physics Applications (EOP)

Revised: 25 January 1974

P/L CODE MISS MODEL	P/L CODE SSPDA NO.	PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC ALTITUDE (km)	ORBIT CHARACTERISTIC INCLINATION (deg)	P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	80	81	82	83	84	85	86	87	88	89	90	91
EOP-3	OP-07	SEASAT-B	EXP.	Any	600 $\pm$ 100	90 $\pm$ 0.10	Any	1	5	10	0/1	1	1/0	1	1	1	1					
EOP-4	OP-01	Geopause	EXP.	0.16	30,000 $\pm$ 46	90 $\pm$ 0.10	(1)	2	3	6	0/2	2	2/0	2	2							
EOP-5	OP-02	Grav. Gradiometer	EXP.	1.10	200 $\pm$ 10	90 $\pm$ 0.10	Any	1	1	1	1/0											
EOP-6A	OP-03	Mini-Laser Geodynamic Satellite	EXP.	Any	650^{+350}_{-250}	90 $\pm$ (0.10)	(2)	2	5	10	2/0	2	2	2	2	2/0	2	2	2	2		
-6B		Mini-Laser Geodynamic Satellite	EXP.	Any	650^{+350}_{-250}	55 $\pm$ (0.10)	(2)	2	5	10	2/0	2	2	2	2	2/0	2	2	2	2		
-6C		Mini-Laser Geodynamic Satellite	EXP.	Any	650^{+350}_{-250}	28.5 $\pm$ (0.10)	(2)	2	5	10	2/0	2	2	2	2	2/0	2	2	2	2		
EOP-7	OP-04	GRAVSAT	EXP.	Any	200 $\pm$ 5	90 $\pm$ 0.10	Any	2	2	2	0/2											
EOP-8	OP-05	Vector Magnetometer Satellite	EXP.	Any	400 $\pm$ 10	90 $\pm$ 5.0	(3)	3	0.5	10		3/0					3/0				3/0	
EOP-9	OP-06	Magnetic Monitor Satellite	EXP.	Any	1500 $\pm$ 500	28^{+62}_{-28}	Any	1	1	10		1/0					1/0				1/0	

(1) Orbit plane normal to ecliptic plane $\pm$ 0.5° and both can be launched on same launch.

(2) Each pair of satellites to be deployed from same launch with 1 ft/sec delta velocity imparted to one satellite with respect to the other.

(3) Each satellite is phased 4 hours apart in local time (60 deg.) (to be verified in NASA review cycle).

Table A-1. 1973 NASA Automated Payload Model Non-NASA/Non-DoD Payloads (NND)

Revised: 25 January 1974

P/L CODE: MISS MODEL	P/L CODE: SSPDA NO.	PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC: ALTITUDE (km)	ORBIT CHARACTERISTIC: INCLINATION (deg)	P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	80	81	82	83	84	85	86	87	88	89	90	91
		COMM/NAV																				
NND-1A	CN-51	International Comm. (1) (3)	EXP.	Any	Sync ± 46	0 ± 0.1	40 W	≥ 1	10	12	2/2	4	4	△1/4	2/5	△2/7	1/9	9	9	△1/8	2/7	2/9
-1B		International Comm. (1)	EXP.	Any	Sync ± 46	0 ± 0.1	180 W	≥ 1	10	12	1/2	3	3	1/3	1/4	5	1/5	6	5	1/4	1/4	5
NND-2A	CN-52	U. S. Domestic - A (3)	EXP.	Any	Sync ± 46	0 ± 0.1	88 to 135W	≥ 1	7	11	1/6	△2/7	2/9	1/11	11	7	6	5	3	1		
-2B	CN-53	U. S. Domestic - B (ADV)	EXP.	Any	ync ± 46	0 ± 0.1	88 to 135W	≥ 1	10	10					①/0	1/1	△2/2	2/4	3/6	△2/9	2/11	1/13
-2C	CN-58	U. S. Domestic - C (TDRS) (2) (3)	EXP.	Any	Sync ± 46	2.5 ± 0.1	11 W	1	5	10				△1/0	1	1	1	1	△1/0	1	1	1
-2D		U. S. Domestic - C (TDRS)	EXP.	Any	Sync ± 46	2.5 ± 0.1	141 W	1	5	10				1/0	1	1	1	1	1/0	1	1	1
NND-3A	CN-54	Disaster Warning (3)	EXP.	Any	Sync ± 46	0 ± 0.6	94 W	1	5	14		△1/0	1	1	1	△1/1	1	1	1	1	△1/0	1
-3B		Disaster Warning	EXP.	Any	Sync ± 46	0 ± 0.6	124 W	1	5	14			1/0	1	1	1	1					
NND-4A	CN-55	Traffic Management (3)	EXP.	Any	Sync ± 19	2.15 ± 0.31	29 W	1	5	16	2	△1/2	2	1/2	2	2	1	1	△1/0	1	1	1
-4B		Traffic Management	EXP.	Any	Sync ± 19	2.15 ± 0.31	52 W	1	5	16	1/2	1/3	4	3	1/2	2	1	1	1		△1/0	1
-4C		Traffic Management	EXP.	Any	Sync ± 19	2.15 ± 0.31	162 W	1	5	16	1/1	2	1/2	2	2	1	△1/1	1	1	1	1	
NND-5A	CN-56	Foreign Communication (3)	EXP.	Any	Sync ± 46	0 ± 0.1	60 W	1	7	17	3	△1/3	4	1/4	3	1/2	3	△1/3	3	1/3	3	1/3
-5B		Foreign Communication	EXP.	Any	Sync ± 46	0 ± 0.1	96 W	1	7	17	1	1	1/1	2	△1/1	2	1/2	3	1/3	3	△1/3	3
NND-6	CN-59	Communication R&D/Proto.	EXP.	Any	Sync ± 46	0 ± 0.2	115 to 140W	1	5	10						①/0	1	1	①/1	2	①/1	2

(1) Launches based on expected traffic between Atlantic and Pacific of 2 to 1 (67% over Atlantic and 33 % over Pacific).

(2) One required in the system but two planned for high availability by providing one on-orbit spare satellite in the nominal model.

(3) NASA developed satellite

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Table A-1. 1973 NASA Automated Payload Model Non-NASA/Non-DoD Payloads (NND)

Revised: 25 January 1974

P/L CODE MISS MODEL	P/L CODE SSPDA NO.	PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC ALTITUDE (km)	ORBIT CHARACTERISTIC INCLINATION (deg)	P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	80	81	82	83	84	85	86	87	88	89	90	91
		EARTH OBSERVATIONS																				
NND-8	EO-56	Environmental Monitoring Satellite (1)	EXP.	0.33	1685 ± 46	102.97 ± 0.04	1144.5 GMT (2)	1	2	13	△1/0	1/1	1/1	1		△1/0	1/1	1/1	1/1	1	△1/0	1/1
NND-9A	EO-57	Foreign Sync. Met. Satellite (1)	EXP.	Any	Sync ± 46	0 ± 0.6	140 E	1	5	14	0/1	1	1/1	1	1	1	1/1	1	1	1	1/1	1
-9B		Foreign Sync. Met. Satellite	EXP.	Any	Sync ± 46	0 ± 0.6	60 W	1	5	14		△1/0	1	1	△1/1	2	1	1	△1/1	1	1	1
NND-10A	EO-58	Geosync. Oper. Envir. Satellite (1)	EXP.	Any	Sync ± 46	0 ± 0.6	80 W	1	5	16	2	2	1/1	2	1	△1/1	2	1	1/1	2	1	△1/1
-10B		Geosync. Oper. Envir. Satellite	EXP.	Any	Sync ± 46	0 ± 0.6	120 W	1	5	16	1	△1/1	2	1/1	2	2	1	△1/1	1	1/1	2	2
NND-11A	EO-61	Earth Resource - LEO (1)	EXP.	(0.33)	907.7 ± 23	99.098 ± 0.10	9:00 (3)	1	2	14	0/1	1/0	1	1/0	1	1/0	1	1/0	1	△1/0	1	1/0
-11B		Earth Resource - LEO	EXP.	(0.33)	907.7 ± 23	99.098 ± 0.10	15:00 (3)	1	2	14	△1/0	1	1/0	1	1/0	1	△1/0	1	1/0	1	1/0	1
NND-12A	EO-59	Earth Resource - Geosync. (1)	EXP.	Any	Sync ± 46	0 ± 0.2	80 W	1	2	14									△1/0	1	1/0	1
-12B		Earth Resource - Geosync.	EXP.	Any	Sync ± 46	0 ± 0.2	120 W	1	2	14									1/0	1	1/0	1
NND-13A	EO-62	Earth Resource - Foreign (1)	EXP.	Any	Sync ± 46	0 ± 0.2	60 W	1	2	14									△1/0	2/1	2	△1/0
		EARTH & OCEAN PHYSICS																				
NND-14	OP-08	Global Earth & Ocean Monit. (1)	EXP.	Any	371 ± 46	98 ± 0.1	Any (4)	3	2	10							△3/0	3	3/0	3	△3/0	3

(1) NASA Developed Payload
(2) WTR Launch Time
(3) Assumed Nodal Crossing Times
(4) To Provide Global Coverage Each Satellite Should be Deployed Nominally 60° Longitude Apart

Table A-2. Expendable Satellite Characteristics

TRACKING AND DATA RELAY SATELL NND-2C PAGE 67 02/07/74

MISS. OBJ. DEVELOP AND DEMONSTRATE A WORLD-WIDE TRACKING AND DATA ACQUISITION SATELLITE TO SUPPORT LOW EARTH ORBITING SPACE MISSIONS

PROGRAM	PAYLOAD	AGENCY	NO SATS	CHAR VELO	DELTA V100	DELTA V160	DELTA V200	EQ CIR OR
COMM/NAVIG	CURR EXP	NASA/OS	3.	39663.	14096.	13882.	13743.	19323.
NOM INCLIN	NOM APOG	NOM PERIG	NOM ECCENT	MAX APOG	MIN APOG	MAX PERIG	MIN PERIG	MAX INCLIN
.00	.1932E+05	.1932E+05	.000000	.1934E+05	.1930E+05	.1935E+05	.1935E+05	3.000
MIN INCLIN	LCH WINDOW	LCH SITE	SYS LF	SC MMD	ME LF	RETRIEVE	MAX PLD VS	LCH VOL
-3.000	...NONE...	ETR	5.	5.000	5.000	YES	.0	427.0
LCH LENGTH	LCH DIA	ADPT LEN	T LCH LEN	SENSOR 1	SENSOR 2	SENSOR 3	SENSOR 4	SENSOR 5
13.70	5.300	1.830	15.53	HF/VHF/UHF	LASER DTEC	SOLID S DE	SATL LINKS	GRND LINKS
POINT ACC	AV E PWR	ST W	ENV CONT W	STAB W	PROP W	PROP P W	PROP DR W	SEP WT
0.1 DEG	430.0	91.00	24.00	19.00	.0	.0	.0	.0
SEP PROP W	SEP DR W	A C W	A C P W	A C O W	TTC W	ELEC W	MIS E W	LANDER W
.0	.0	125.0	70.00	56.00	.0	154.0	273.0	.0
RESID W	CREW EQ W	PALLET W	TOTAL DW	TOTAL W W	ADAPTER W	LAUNCH W	TYPE ST	TYPE PROPS
.0	.0	.0	617.0	687.0	57.80	744.8	EXO	LIQUID
TYPE PROPL	TYPE A C	TYPE E PWR	TYPE ME	TYPE SD	TYPE MD	RD FUND YR	BAS AGE C	ST CLASS
HYDRAZINE	3-AXIS	SOLAR/BATT	LOW.COMPLX	*NO ENTRY*	*NO ENTRY*	3.000	1.000	SEO
ECLS CLASS	STAB CLASS	PROP CLASS	A C CLASS	TTC CLASS	ELEC CLASS	MIS E CLAS		
SEO	RL	RL	COM	RL	SEO	COM		

APPENDIX B

EXAMPLES OF REDESIGN OF EXPENDABLE PAYLOADS FOR SPACE SERVICING

This appendix contains an example of the configuration and characteristics of a typical satellite redesigned for space servicing. The example chosen is the same one. TDRS, that was used in Appendix A so that a one-for-one comparison between the expendable and space serviceable versions can be seen.

Table B-1 presents an example of the basic mission model of Appendix A revised to present the space serviceable characteristics of the various satellites, including the TDRS satellite. The most significant change that occurred as a result of the redesigns was in the physical properties of the satellites. In general, the weights and the diameters of the satellites increased while the lengths of the satellites decreased.

Table B-2 presents a breakdown of the space replaceable modules comprising the example TDRS satellite. Included are data on the number of modules required and the weight of the various modules comprising the satellite.

Table B-3 presents a breakdown of a typical module (AVCS-7) contained in the TDRS satellite. Included in the data are the number of components required, the weight of each component, the weight and reliability (Weibull) parameters of the module, and the functional arrangement of the components within the module.

Figure B-1 presents a functional layout of the space replaceable modules contained in the TDRS satellite, and information on the number of modules (in a redundant set of modules) that must remain functional for the satellite to remain operational. The reliability characteristics of the satellite are also given.

Table B-1. NASA Mission Model Revised for Space Servicing Program Characteristics - Non-NASA/Non-DoD (NND)

Payload Code		Payload Name	Payload Category	Design Parameters				Mission Param		
Mission Model	SSPDA			Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Ini La D
		Comm/Navigation								
NND-1A	CN-51	International Comm.(1)	RAS	2.50	4.35	0.70	2685	2	2	7
-1B		International Comm.(1)	RAS	2.50	4.35	0.70	2685	2	2	7
NND-2A	CN-52	U.S. Domestic - A	RAS	2.50	4.35	0.40	986	2	2	7
-2B	CN-53	U.S. Domestic - B	RAS	2.50	4.35	0.70	2685	2	2	8
-2C		U.S. Domestic - B	RAS	2.50	4.35	0.70	2685	2	2	8
-2D	CN-58	U.S. Domestic - C (TDRS) (2)	RAS	0.76	4.35	0.38	1325	1	1	8
NND-3A	CN-54	Disater Warning	RAS	2.00	4.35	0.50	1349	1	1	8
-3B		Disater Warning	RAS	2.00	4.35	0.50	1349	1	1	8
NND-4A	CN-55	Traffic Management	RAS	3.00	4.35	0.50	1136	1	1	7
-4B		Traffic Management	RAS	3.00	4.35	0.50	1136	1	1	7
-4C		Traffic Management	RAS	3.00	4.35	0.50	1136	1	1	7
NND-5A	CN-56	Foreign Communication	RAS	2.80	4.35	0.50	987	1	1	7
-5B		Foreign Communication	RAS	2.80	4.35	0.50	987	1	1	[illegible]
NND-6	CN-59	Communication R&D/Proto.	RAS	3.40	4.35	0.80	3148	1	1	[illegible]

NOTES:
(1) Launches based on expected traffic between Atlantic and Pacific of 2 to 1 (67% over Atlantic and 33% over Pacific).
(2) One is spare since only two are required in the system.

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Mission Parameters				Orbital Parameters				Lifetime Parameters			
Total Number in System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
2	2	78	Any	Sync ±46	0 ±0.1	40W	11,700	12	10		0.49
2	2	79	Any	Sync ±46	0 ±0.1	180W	11,700	12	10		0.49
2	2	78	Any	Sync ±46	0 ±0.1	110W	11,700	11	10		0.69
2	2	84	Any	Sync ±46	0 ±0.1	90W	11,700	10	10		0.49
2	2	83	Any	Sync ±46	0 ±0.1	120W	11,700	10	10		0.49
1	1	83	Any	Sync ±46	2.5 ±0.1	11W	11,700	10	7		0.37
1	1	81	Any	Sync ±46	2.5 ±0.1	141W	11,700	10	7		0.52
1	1	82	Any	Sync ±46	0 ±0.6	124W	11,700	14	7		0.52
1	1	77	Any	Sync ±19	2.15 ±0.31	29W	11,700	16	7		0.60
1	1	78	Any	Sync ±19	2.15 ±0.31	52W	11,700	16	7		0.60
1	1	79	Any	Sync ±19	2.15 ±0.31	162W	11,700	16	7		0.60
1	1	77	Any	Sync ±46	0 ±0.1	0	11,700	17	10		0.49
1	1	78	Any	Sync ±46	0 ±0.1	96W	11,700	17	10		0.49
1	1	85	Any	Sync ±46	0 ±0.1	160W	11,700	10	3		0.23

over Pacific).

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Table B-2. Satellite Mod

Satellite			Non-Replaceable Units			Spa						
						Standardi						
Payload Code						Attitude & Velocity Control			Guidance & Navigation			Te
Mission Model	SSPDA	Payload Code	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	It
EO-7	EO-07	Synchronous Meteorological Satellite	NEO-7	1	364.0	AVCS-3 AVCS-6 AVCS-7	1 1 4	109.2 72.5 202.4 384.1	GN-2	1	42.6	TT
EOP-3	OP-07	Seasat B	NEOP-3	1	380.0	AVCS-2 AVCS-5A AVCS-7 AVCS-9	1 1 2 1	42.6 55.9 101.2 51.3 251.0	N/A			TT
EOP-4	OP-01	Geopause	NEOP-4	1	338.0	AVCS-2 AVCS-5 AVCS-7	1 1 2	42.6 55.9 101.2 199.7	N/A			TT
EOP-7	OP-04	Gravsat	NEOP-7	1	320.0	AVCS-3 AVCS-5A AVCS-8 AVCS-9	1 1 4 1	42.6 55.9 230.4 51.3 380.2	N/A			TT
NND-1	CN-51	International Communication Satellite	NNND-1	1	829.0	AVCS-3 AVCS-5 AVCS-7	1 1 4	109.2 55.9 202.4 367.5	GN-2	1	42.6	TT
NND-2A	CN-52	U.S. Domestic Satellite - A	NNND-2A	1	318.0	AVCS-1 AVCS-5 AVCS-7	1 1 4	38.9 55.9 202.4 297.2	N/A			TT
NND-2B	CN-53	U.S. Domestic Satellite - B	NNND-2B	1	829.0	AVCS-3 AVCS-5 AVCS-7	1 1 4	109.2 55.9 202.4 367.5	GN-2	1	42.6	TT
NND-2D	CN-58	U.S. Domestic Satellite - C	NNND-2D	1	345.0	AVCS-1 AVCS-5 AVCS-7	2 1 4	77.8 55.9 202.4 336.1	N/A			TT

-2. Satellite Module Assignment

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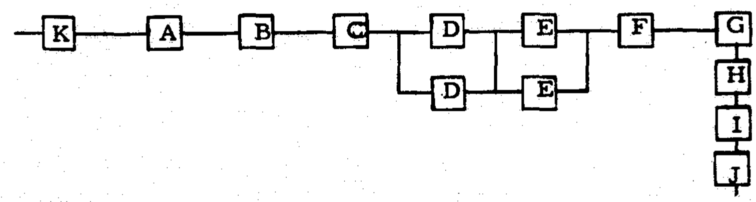
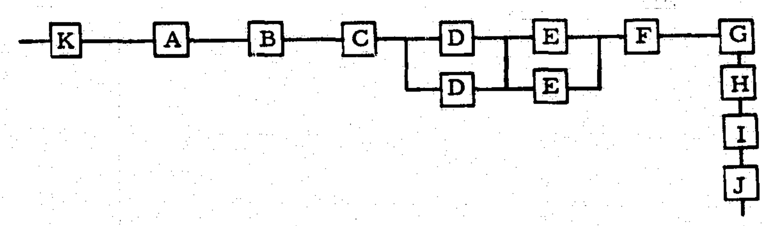
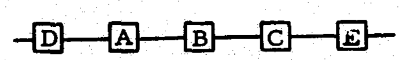
Space Replaceable Units															Satellite Weight (kg)		
Standardized												Non-Standardized					
Guidance & Navigation			Telemetry, Tracking & Command			Data Processing			Electrical Power			Mission Equipment					
Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	DRY	WP	WET
GN-2	1	42.6	TTC-6	1	54.0	DP-1M	1	46.8	EPS-2 EPS-3	1 2	131.0 102.0 234.0	EO-7-1 -2 -3 -4 -5	1 1 1 1 1	99.8 88.8 64.0 59.3 82.0 393.9	1519	62	1581
N/A			TTC-9A	1	65.0	DP-1N	1	46.8	EPS-1B EPS-2 EPS-3	1 1 2	78.0 131.0 102.0 311.0	EOP-3-1 -2 -3 -4 -5	1 1 1 1 1	99.8 145.8 97.8 127.8 102.8 574.0	1628	50	1678
N/A			TTC-5A	1	61.0	DP-1O	1	46.8	EPS-1C	1	105.0	EOP-4-1 -2	1 1	145.4 84.8 230.2	981	57	1038
N/A			TTC-10	1	63.0	DP-1P	1	46.8	EPS-1D	1	134.0	EOP-7-1 -2	1 1	280.8 145.4 426.2	1370	255	1625
GN-2	1	42.6	TTC-1	1	51.0	DP-1Q	1	46.8	EPS-2 EPS-3	6 2	786.0 102.0 888.0	NND-1-1 -2	1 1	171.0 248.7 419.7	2645	40	2685
N/A			TTC-1	1	51.0	DP-1R	1	46.8	EPS-1D	1	134.0	NND-2A-1	1	88.8	936	50	986
GN-2	1	42.6	TTC-1	1	51.0	DP-1S	1	46.8	EPS-2 EPS-3	6 2	786.0 102.0 888.0	NND-2B-1 -2	1 1	171.0 248.7 419.7	2645	40	2685
N/A			TTC-2	1	51.0	DP-1T	1	46.8	EPS-1D EPS-3	1 1	134.0 51.0 185.0	NND-2D-1 -2 -3	1 1 1	80.1 131.1 68.9 280.1	1244	81	1325

Table B-3. Standardized Subsystem Modules - Attitude and Velocity Control System

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODU DESIG LIFE (yrs)
AVCS-7	Hot Gas Propulsion (N_2H_4) Small Tank	A	Nitrogen Tank (7. 5-in OD)	1	2. 3	2. 3	1500	7
		B	Start Valve	1	0. 5	0. 5	100	
		C	Regulator Valve	1	1. 8	1. 8	100	
		D	Temperature Tranducer	2	0. 05	0. 1	2000	
		E	Pressure Transducer	2	0. 05	0. 1	2000	
		F	Hydrazine Tank (15-in OD)	1	4. 0	4. 0	1500	
		G	Latching Valves	2	0. 5	1. 0	200	
		H	Thruster (0. 1 lb)	4	0. 9	3. 0	1000	
		I	Thruster (5. 0 lb)	3	1. 4	4. 2	2000	
		J	Remote Terminal	1	2. 0	2. 0	500	
		K	Power Conditioning	1	2. 0	2. 0	500	
			Cabling	AR	5. 0	5. 0		
			Connectors	AR	2. 0	2. 0		
			Environmental Protection	AR	5. 0	5. 0		
			Structure	AR	17. 0	17. 0		
			TOTAL			50. 6		
AVCS-8	Hot Gas Propulsion (N_2H_4) Small Tank	A	Nitrogen Tank (7. 5-in OD)	1	2. 3	2. 3	1500	7
		B	Start Valve	1	0. 5	0. 5	100	
		C	Regulator Valve	1	1. 8	1. 8	100	
		D	Temperature Tranducer	2	0. 05	0. 1	2000	
		E	Pressure Transducer	2	0. 05	0. 1	2000	
		F	Hydrazine Tank (24-in OD)	1	11. 0	11. 0	1500	
		G	Latching Valves	2	0. 5	1. 0	200	
		H	Thruster (0. 1 lb)	4	0. 9	3. 6	1000	
		I	Thruster (5. 0 lb)	3	1. 4	4. 2	2000	
		J	Remote Terminal	1	2. 0	2. 0	500	
		K	Power Conditioning	1	2. 0	2. 0	500	
			Cabling	AR	5. 0	5. 0		
			Connectors	AR	2. 0	2. 0		
			Environmental Protection	AR	5. 0	5. 0		
			Structure	AR	17. 0	17. 0		
			TOTAL			57. 6		
AVCS-9	Magnetic Torquer	A	Magnetometer (3 Axis)	1	3. 2	3. 2	200	7
		B	Amplifier	1	1. 4	1. 4	1600	
		C	Coil	3	4. 6	13. 7	200	
		D	Power Conditioning	1	2. 0	2. 0	500	
		E	Remote Terminal	1	2. 0	2. 0		
			Cabling	AR	5. 0	5. 0		
			Connectors	AR	2. 0	2. 0		
			Environmental Protection	AR	5. 0	5. 0		
			Structure	AR	17. 0	17. 0		
			TOTAL			51. 3		

s - Attitude and

FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
1500 100 100 2000 2000 1500 200 1000 2000 500 500	7	.496	14.35	1.021	
1500 100 100 2000 2000 1500 200 1000 2000 500 500	7	.496	14.35	1.021	
200 1600 200 500	7	.832	38.05	1.0	

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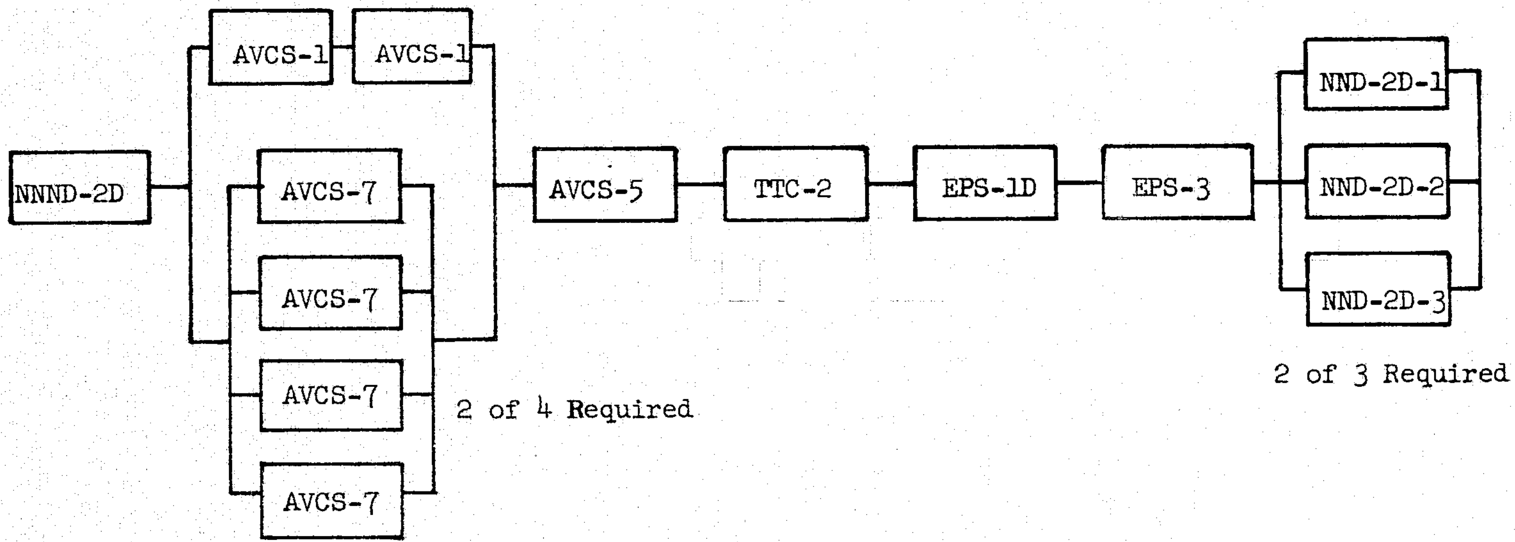
NND-2D U. S. Domestic Satellite-C (TDRS)



PAYLOAD CHARACTERISTICS	
NUMBER OF SRUs	13
DESIGN LIFE (YRS)	7
RELIABILITY @ D. L.	0.148
WEIGHT - DRY (Kg)	1233.7
PROP (Kg)	81
TOTAL (Kg)	1314.7

Figure B-1. Space Serviceable Payload Descriptions

APPENDIX C

STAGE CHARACTERISTICS OF CANDIDATE UPPER STAGE VEHICLES

This appendix contains the basic data on the various upper stages that were utilized in the analyses. The reference source of data for the Transtage and Centaur vehicles used in the analyses was an internal memorandum generated for the USAF/SAMSO upper stage assessment of November 1973. The orbital lifetimes of the basic Transtage and Centaur designs were approximately one and two days respectively. The source of data for the FCT was obtained from NASA-MSFC documentation. The orbital lifetime of the tug was seven days. Data for the SEPS was obtained from Rockwell International final SEPS documentation.

Tables C-1 and C-2 present the characteristics of the basic Transtage and Centaur upper stage vehicles. Table C-3 presents the characteristics of the Kick stage used in conjunction with the Transtage for synchronous equational missions. Table C-4 presents the stage characteristics of the MSFC designed FCT. Table C-5 presents the characteristics of the SEPS.

Seven-day versions of the Transtage and Centaur vehicles were synthesized for the analyses primarily from data internally generated for that purpose. In addition, estimates on the reduction of main propellant boiloff that could be expected on the Centaur with minor modifications was obtained from General Dynamics Corporation. Even with the reduced boiloff rate, the total boiloff was still too high to use the LOVES program performance routines.

The upper stage performance routines in the LOVES computer code employs an effective specific impulse (IEF) in the computations. This IEF is a product of the vacuum Isp and the ratio of the propellant available for impulsive maneuvers to the total propellant initially available for us.

Table C-1. Reusable Transtage Characteristics

Characteristic	Units	Value	
Dry Weight	Kg (lb)	2000	(4400)
Non Usable Propellant	Kg (lb)	69	(153)
Burnout Weight	Kg (lb)	2050	(4553)
Non Impulsive Propellant	Kg (lb)	25	(54)
Attitude Control Propellant	Kg (lb)	x 103	(237)
Impulsive Propellant	Kg (lb)	14550	(32000)
First Ignition Weight (Max)	Kg (lb)	16750	(36844)
Orbiter Interface Accommodations	Kg (lb)	1360	(3000)
Nominal ISP	Sec	310	
Flight Velocity Reserve	%	1	

Table C-2. Reusable 28-ft Large Tank Centaur Characteristics

Characteristic	Units	Value	
Dry Weight	Kg (lb)	2275	(4959)
Non Usable Propellant	Kg (lb)	324	(717)
Burnout Weight	Kg (lb)	2580	(5676)
Non Impulsive Propellant	Kg (lb)	367	(809)
Attitude Control Propellant	Kg (lb)	214	(472)
Impulsive Propellant	Kg (lb)	20600	(45313)
First Ignition Weight (Max)	Kg (lb)	23750	(52270)
Orbiter Interface Accommodations	Kg (lb)	1095	(2411)
Nominal ISP	Sec	439.2	
Flight Velocity Reserve	%	1	

Table C-3. Expendable Geosynchronous Kick Stage Characteristics

Characteristic	Units	Value
Dry Weight	Kg (lb)	356 (784)
Non Usable Propellant	Kg (lb)	6 (14)
Burnout Weight	Kg (lb)	363 (798)
Non Impulsive Propellant	Kg (lb)	15 (33)
Attitude Control Propellant	Kg (lb)	58 (128)
Impulsive Propellant	Kg (lb)	1805 (3967)
First Ignition Weight (Max)	Kg (lb)	2240 (4926)
Stage Interface Accommodations	Kg (lb)	85 (188)
Nominal ISP	Sec	288
Flight Velocity Reserve	%	NA

Table C-4. Full Capability Tug (FCT) Characteristics

Characteristic	Units	Value	
Dry Weight	Kg (lb)	2340	(5150)
Non Usable Propellant	Kg (lb)	275	(605)
Burnout Weight	Kg (lb)	2660	(5755)
Non Impulsive Propellant	Kg (lb)	248	(547)
Attitude Control Propellant	Kg (lb)	131	(288)
Impulsive Propellant	Kg (lb)	22650	(49889)
First Ignition Weight (Max)	Kg (lb)	25800	(56779)
Orbiter Interface Accommodations	Kg (lb)	860	(1900)
Nominal ISP	Sec	456	
Flight Velocity Reserve	%	1	

Table C-5. Envelope of SEPS Design Characteristics

Item	Characteristic
Maximum powered flight (days)	1080 (Encke Rendezvous)
Maximum subsystem lifetime (days)	1650 days (Earth Orbital)
Maximum/minimum solar distance (AU)	0.32/3.7
Maximum communication distance (AU)	4.7 (outer planets)
Maximum dry weight/mercury propellant weight kg (lb)	1230/1500 (2700/3300)
Maximum solar array/TSS power (kW)	25/21
Specific Impulse (Sec)	3000
Maximum installed dimensions m (ft)	3 x 4.25 dia (10 x 14 dia)
TSS efficiency (mission dependent)	0.61 to 0.66
Stabilization (deg)	$\pm$ 1.0
Science pointing accuracy (deg)	$\pm$ 0.5

As long as the ratio remains within a few percentage points of unity, the approximation gives reasonably accurate results. However, when the total propellant loss due to boiloff becomes very large, the accuracy of the computation diminishes rapidly. As a matter of fact, use of the approximate method gives extremely optimistic (nonconservative) results under those conditions. In order to obtain meaningful results when using the Centaur upper stage in the analysis, provisions were made to modify the LOVES performance routines to account discretely for the propellant boiloff in cases where it becomes excessive. This was implemented as an input option so that the performance routines could be used in either mode. This alternate performance computation was used only for the Centaur vehicle since the Tug was designed for a seven-day mission and the Transtage had no boiloff problems, being a storable propellant vehicle.