Volume II

**NUCLEAR INTEGRATED MULTIMISSION
SPACECRAFT (NIMS) DESIGN DEFINITION**

**MASTER
DYNAMIC ISOTOPE POWER SYSTEM (DIPS)
APPLICATIONS STUDY**

FINAL REPORT

November 1979

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REPORT ORGANIZATION

This report is presented in three volumes:

- Volume I — Summary**
- Volume II — NIMS Design Definition**
- Volume III — NIMS Survivability Assessment**

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SECTION 1

INTRODUCTION

The purpose of the Dynamic Isotope Power System (DIPS) Application Study was to show the utility and advantages of using the DIPS as the power source for a variety of Air Force Space Missions as compared to solar powered spacecraft. The DIPS has a primary advantage in its inherent hardness as compared to solar array systems and offers, as will be shown, potential mission improvements because of the relaxation or elimination of many constraints associated with large, oriented solar arrays.

The DIPS, shown schematically in Figure 1-1, is being developed by Sundstrand Energy Systems organization for the Department of Energy (DOE) who also sponsored this applications study. A brief description of the DIPS operation follows. Further details of the DIPS and the related DOE program are presented in Appendix A of this volume.

The heart of the system is the turbine-alternator-pump mounted on a common shaft. This single unit is supported by tilt-pad thrust and journal bearings lubricated by the Dowtherm A working fluid. There is no contact between the rotating shaft and the bearings during DIPS operation, except for a few seconds during startup. The system has two major flow paths. Starting at the pump discharge "A" (see Figure 1-1) the fluid is liquid. Approximately 10 percent of the fluid moves through the inside of the tubes in the recuperator, accepting heat from, and cooling the turbine discharge vapor. The fluid then passes through the spiral tubes surrounding the isotope heat source where the Dowtherm A is vaporized and heated to its peak temperature of 6750°F. This vapor is then expanded through the turbine where energy is removed and converted to shaft power to drive the alternator and pump. After leaving the turbine, the vapor passes

Figure 1-1. DIPS Schematic

over the external surfaces of the recuperator tubes where it preheats the pump discharge flow, enhancing system efficiency. The vapor is then condensed on the subcooled radiator flow in the jet condenser and proceeds to the accumulator, from which it is returned to the pump, completing the power conversion cycle.

Returning this discussion to the pump discharge, approximately 90 percent of the flow passes through the radiator as a single phase fluid. There is no condensing (two-phase flow) in the radiator. This liquid is cooled in the radiator, rejecting the waste heat picked up from the vapor in the jet condenser. This fluid exists from the radiator to a "shower-head" type of fixture in the jet condenser.

These multiple fluid streams generated in the shower head provide the surface area and heat capacity to condense the vapor exiting from the power conversion system recuperator. This jet condenser is fully effective in zero gravity, and will operate at accelerations up to 30 g's through launch and orbit injection.

Ancillary temperature regulation valves and electrical controls (not shown) maintain the proper relationship of working fluid temperatures and provide control of the electrical output, respectively.

SECTION 2

OBJECTIVES AND GUIDELINES

2.1 OBJECTIVES

The study objectives are discussed.

2.1.1 DIPS ATTRIBUTES ASSESSMENT

The baseline objective is to assess the attributes of the Dynamic Isotope Power System (DIPS) when it is thermally integrated with a spacecraft bus. This assessment addresses all identified DOD missions for which the DIPS may have potential application during approximately the next ten years. Its scope includes operational advantages and disadvantages relative to solar array/battery powered spacecraft.

The scope of the economic assessment covers the development (through qualification) and unit costs of the spacecraft bus. No payload costs are considered. Rough-Order of Magnitude (ROM) costs are delineated and compared for

- 1) The Nuclear Integrated Multimission Spacecraft (NIMS) Bus which has enhanced survivability and extended mission life in natural and hazardous man-made space environments.
- 2) A similar but non-enhanced solar array/battery powered spacecraft bus.

No rigorous comparison with a solar array/battery powered bus is possible at this time because comparable data is not currently available.

2.1.2 NIMS BUS REQUIREMENT

The definition of a Nuclear Integrated Multimission spacecraft (NIMS) Bus is necessary to support the baseline objective. The NIMS Bus must be compatible with any of the payloads required by the candidate missions assessed. This includes communications, surveillance, navigation and meteorological missions.

2.1.3 UTILIZATION OF PREVIOUS WORK

Earlier related work is used to the greatest possible extent. The DIPS thermal integration approach is based on an earlier study which investigated the Global Positioning Satellite (GPS) mission at the stage of its Phase I development*. In addition, developed or evolving hardened spacecraft hardware and design techniques are utilized. Key subsystem designs are based on flight hardware developed at GE Space Division for missions such as NIMBUS, LANDSAT, BSE and DSCS III. Data on subsystems such as attitude control, tracking telemetry and control, propulsion, thermal control, and structure are utilized in the definition of the NIMS Bus and its costs, as well as the Solar Array/Battery (SAB) Bus costs. The Dynamic Isotope Power System (DIPS) costs are provided by the DOE from their contractor, Sundstrand Energy Systems.

2.2 GUIDELINES

This paragraph presents the key guidelines employed in the definition of the NIMS Bus. These are listed on Table 2.2-1.

The definition of a nuclear-powered spacecraft bus was required to permit the attributes assessment of the DIPS. This bus must be fully capable of meeting all requirements of the six Air Force missions considered. Accomplishment of this design in the most cost-effective way requires maximum standardization of the multi-mission bus and all of its support subsystems. These include the Tracking, Telemetry and Command (TT&C) subsystem, the Attitude Control System (ACS), the Thermal Control System (TCS), and the

*Nuclear-Powered GPS Spacecraft Design Study, Fairchild Space & Electronics Co., ERDA NRA 3063 10 May 1977

TABLE 2.2-1 - GUIDELINES

- Define spacecraft with multi-mission payload capability
- Standardize:
 - Configuration
 - Tracking, Telemetry and Command (TT&C)
 - Attitude Control System (ACS)
 - Thermal Control System (TCS)
 - Propulsion
 - Perigee and Apogee Functions
 - Electrical Power and Distribution (EP&D)
- Provide for payload thermal control
with DIPS radiator
- Design for survivability
- Provide for separate fabrication, assembly and test
of both NIMS and the mission payload.
- Assure compatibility with STS launch
- On-board propulsion selection must consider both
 - Initial deployment mode
 - Replacement mode

propulsion system; the perigee and apogee functions. The Electrical Power System is also standardized in that the DIPS is used together with standard harness and electronic payload switchgear.

The bus is to enhance survivability against specified nuclear and laser environments. To this end, the bus and payload thermal control system is integrated with the DIPS radiator to maximize survivability against the laser threat. The approach to enhance survivability from the JCS threat maximizes use of design techniques and procedures being employed on other spacecraft programs being conducted for DOD.

The requirement that the spacecraft has to be of a modular design was established to simplify fabrication and test. This approach permits the NIMS bus development with only limited dependence on the mission payloads, and the parallel assembly and test of the NIMS bus and the payloads.

Since the missions considered are all in the mid-80's to mid-90's time frame, the requirement of STS compatibility is a must. The on-board propulsion system is required to place the spacecraft into its final orbit starting from the low earth orbit of the STS (nominally 160 NM circular). The selection of this on-board propulsion system must consider both the initial deployment and the replacement modes. For example, although multiple spacecraft launch with the IUS on-board propulsion system may be cost effective for mission initial deployment, an alternate propulsion system may be more attractive when the eventual satellite replacement requirement is also considered.

2.2.1 MULTIMISSION SPACECRAFT DEFINITION

Through the conceptual design of a nuclear-powered spacecraft with the capability of handling a wide range of payloads with differing mission requirement, the versatility of space nuclear power can be shown. In addition to the mission flexibility, the inherent survivability of the DIPS compared to a solar array powered system can significantly improve mission success.

Because of the wide range of mission requirements to be satisfied, caution was exercised in the over-all design to minimize the potential problem of overdesigning some of the support subsystems with respect to the individual missions' requirements. An example of this was in the Attitude Control Subsystem (ACS) area where four of the six missions evaluated required a pointing accuracy of about 0.08° in pitch and roll and 0.8° in yaw and the other two missions had a pointing requirement of 0.01° in all three axes. Obviously, using a standardized ACS with a pointing accuracy of 0.01° in all three axis would penalize four of the missions. Therefore, the baseline design was selected to meet the majority of the mission's needs with the modifications to the ACS to meet the other two mission needs identified.

2.2.2 SUPPORT SUBSYSTEM STANDARDIZATION

The preliminary design definition effort on the support subsystems listed in Table 2.2-1 was driven by the self imposed requirement for standardization of these subsystems as far as possible. In the configuration/structure, TT&C, Thermal Control, and Propulsion areas, this was fairly straightforward. The ACS, as mentioned earlier, and the Electrical Power and Distribution (EP&D) areas require mission-unique changes. In the EP&D subsystem, these changes are to the harness and load control and switching units.

2.2.3 INTEGRATED THERMAL CONTROL

In the earlier study on a nuclear powered spacecraft for the GPS mission*, the DIPS active cooling loop and the spacecraft electronics thermal control were integrated to assure thermal uniformity and minimum temperature variation in orbit. This approach was to be used in this multimission spacecraft design.

2.2.4 SURVIVABILITY

Survivability against both nuclear and laser threats must be considered because of the Air force missions evaluated. Only the spacecraft bus was considered in this assessment; not the payload components. Hardening of these components are being done under existing survivability programs.

*OP. CT.

The survivability aspects of the NIMS Bus are fully covered in Volume III, NIMS Survivability Assessment, of this report.

2.2.5 SPACECRAFT/PAYLOAD FABRICATION, INTEGRATION AND TEST

Another self-imposed requirement was to make the NIMS Bus completely modular in design. This greatly simplifies the fabrication, integration and test flow procedures. Figure 2.2-1, NIMS exploded view, shows the various modules which are built and tested separately and how the vehicle is assembled. As shown, the payload interface is through payload equipment panels on the top to which the payload is mounted and tested independently of the rest of the vehicle. Externally mounted sensors and/or antennas are attached after the equipment panels are installed.

2.2.6 STS LAUNCH COMPATIBILITY

STS compatibility is a definite requirement, since all of the missions considered are in the mid-1980's and beyond time frame. Even though STS launch costs are projected to be significantly lower than expendable launch vehicles, each spacecraft launched via the STS will be charged for launch costs as a function of either its gross weight or the amount of cargo bay length used. To minimize launch costs, the weight and cargo bay lengths usage is minimized. For five of the six missions considered, this was achieved by vertical mounting of the spacecraft in the cargo bay. This was made possible through the use of a combined solid rocket motor (SRM) and bi-propellant propulsion system to minimize the overall height of the vehicle. The sixth mission, DSSS, uses the Inertial Upper Stage (IUS) to place it into the final orbit because of its size and weight (210 ft³, 1786 lbs. See Section 5 for further details). Figure 2.2-2 NIMS-STS Vertical Installation, illustrates the vertical installation concept.

The vehicle structure was designed to meet the worst case STS launch loads imposed on the system. Some of the preliminary structural analyses are described in Paragraph 6.2 Structure, of this volume.

Figure 2.2-1. NIMS Exploded View

Figure 2.2-2. NIMS-STS Vertical Installation

2.2.7 ON-BOARD PROPULSION

Since the STS delivers the spacecraft payloads to a low earth orbit (nominally 160 nautical miles, circular), a means of transporting the spacecraft to the final operational orbits must be provided. Upper stage systems are under development now for this purpose. These include the PAM-A, PAM-D, and IUS systems. The spacecraft is mounted to the top of these systems and with the exception of smaller spacecraft using the PAM-D, are installed horizontally within the STS cargo bay. This approach is more costly, since it takes up more of the cargo bay than does a vertical installation.

An alternative approach to the use of these upper stage systems is to integrate both perigee and apogee transfer functions within the spacecraft. This approach was selected for five of the six missions considered for its packaging efficiency, lower weight, and therefore, lower cost. The approach is to use a large solid rocket motor (SRM) for the perigee function (placing the spacecraft into an elliptical orbit with apogee at the desired operational altitude) and a bi-propellant system to circularize the orbit and remove inclination, if required. The NIMS is 3-axis stabilized during PKM and AKM burn, using the bi-propellant system for thrust vector control.

SECTION 3

NIMS DESCRIPTION

The geometry of the Nuclear Integrated Multimission Spacecraft (NIMS) Bus shown in Figure 3-1 is dominated by the 238 square foot cylindrical radiator required for heat rejection. Approximately 50 percent of this thermally and mechanically integrated structure is required to reject the 5900 Watts of thermal waste heat from the DIPS. The remainder provides an 80° temperature heat sink for up to 1000 Watts thermal of housekeeping and payload electronics cooling.

Figure 3-1. Nuclear Integrated Multimission Spacecraft (NIMS)

The NIMS Bus is compatible with the six DOD missions assessed, providing multimission capability from low earth orbit to beyond synchronous altitudes for a wide range of payloads. It is fully compatible with the STS and IUS.

Enhanced survivability against both nuclear and laser threats is provided. The NIMS integrated thermal control for all subsystems uniquely provides the ability to manage thermal energy during threat encounter.

The DIPS provides 1200 Watts of electrical power, full time, over the 10 year design life of the NIMS. The DIPS can be readily adapted to meet any mission power requirement from 500 to 2500 Watts. The DIPS rotating unit's momentum is used by the 3-axis ACS to meet the precision pointing requirements of the candidate missions.

3.1 SPACECRAFT SYSTEMS APPROACH TO SPACE NUCLEAR POWER

The systems approach employed in the NIMS Bus shown in Figure 3.1-2 consists of three basic elements. This concept provides the ability to manage thermal energy and combines this with the capability for separate payload and NIMS development, fabrication and test.

3.1.1 PUMPED COOLANT LOOP UTILIZATION

The DIPS Rankine cycle pumped coolant is used for electronics and other subsystem cooling for both normal operations and during threat encounter. This is accomplished by oversizing the radiator, above that required to reject waste heat from the DIPS, to provide for electronic cooling. Specifically, the NIMS radiator is sized to reject about 6900 Watts(t); 1000 Watts(t) from the NIMS housekeeping and/or mission dependent payload electronics and 5900 Watts(t) from the DIPS. The DIPS working fluid is cooled to 80°F (27°C), 45°F below the 135°F (57.6°C) minimum nominally required by the DIPS Rankine cycle, in order to provide an 80°F heat sink for electronics cooling. In addition, the coolant tubes in the NIMS tube-and-fin radiator are spiraled circumferentially around the NIMS

Figure 3.1-2. Spacecraft Systems Approach to Space Nuclear Power

cylindrical surface. Each of the four (4) tubes makes 1.25 complete circuits while transversing the 6.3 feet from the hot-to-cold end of the cylinder. Thus, some of the thermal energy absorbed from the threat on the illuminated side of the NIMS is transported to the non-illuminated side of the NIMS, and re-radiated to space.

3.1.2 PAYLOAD/ELECTRONICS INTEGRATION

The payload and NIMS housekeeping electronics are mounted on an annular panel located inside the NIMS radiator envelope. As shown schematically, low temperature heat pipes are employed to collect the waste heat from the electronics and dump this thermal load into the DIPS working fluid at the cold (80°F) NIMS radiator exit header. The panel(s) (cold plate) are bolted or otherwise mechanically attached to the radiator exit header.

3.1.3 MULTIMISSION CAPABILITY

The payload(s) can be developed, fabricated and tested independent from the NIMS Bus, guided by suitable interface specifications. Shortly before launch, the payload is integrated with the NIMS Bus. A detailed flow chart substantiating this approach may be found in Paragraph 6.9 of this volume. A single NIMS Bus may be developed which is compatible with the six (6) mission payloads assessed.

3.2 NIMS FEATURES

The design features of the NIMS shown in Figure 3.2-1 fully satisfy all of the requirements placed on the vehicle. Payload accommodations are provided on the large earth-viewing face with no fields-of-view obstructions from any of the bus subsystems. The TT&C subsystem, based on the DSCS III design, is fully compatible with the Air Force's SGLS system and incorporates both security and A/J protection. The Propulsion Subsystem uses a large solid rocket motor to provide the energy necessary to place the spacecraft into a transfer orbit with perigee at the STS altitude and apogee at the desired altitude (assumed to be, 10,900 NM for GPS; 19,323 NM for DSSS, DSCS IV, GPSC; 110,375 NM for NFCS). Orbit circularization

Figure 3.2-1. NIMS Features

and inclination removal is accomplished by a bi-propellant system. The bi-propellant system can also be used to provide on-orbit agility for those missions which are compatible with this mode of survivability enhancement. The structure is designed to handle the worst case weights/loads. It provides for modularity and is closed off to provide a "Faraday Cage" to minimize EMP/SGEMP effects. The DIPS provides all required power for the mission life. The attitude control subsystem is a 3-axis bias momentum type. It can use the DIPS momentum as the stabilizing element or alternately cancel it out. Thermal control of the bus and payload components is provided by the DIPS active cooling loop through radial heat pipes from the spacecraft's equipment mounting surface.

All electronic equipments used are either from DSCS III or derived from that program.

SECTION 4

MISSIONS

The missions to be used for this study were primarily selected from systems which are now and have been operational Air Force space missions or from advanced system concepts with a strong operational requirement. To prevent the study results becoming oriented to a single mission family, the selection of at least one mission from each of four major mission categories was required.

4.1 MISSION SELECTION

The factors used to select the candidate missions applicable to the NIMS concept include:

- The mission can be identified with the general DOD requirements either as a subsequent phase of an on-going flight program or as a potential new start within the next five-to-ten years. No consideration is given to retro-fitting DIPS to existing spacecraft.
- The payload and mission requirements definition is adequate to assist in establishing the NIMS multimission requirements.
- The schedule for the mission of interest is compatible with the introduction of the NIMS/DIPS developing technology.
- The mission power requirements are consistent with the DIPS capability.

Application of these factors has resulted in the selection of the six (6)

missions shown in Table 4.1-1. Note that at least one mission is identified in each of the four major mission categories; communications, surveillance, navigation and meteorological.

4.1.1 DSCS IV MISSION

The DSCS IV spacecraft system will be the fourth generation of strategic communications satellite systems. The DSCS III spacecraft currently under development and soon to be flown has a six channel SHF (7.5 GHz - 8.5 GHz) transponder, a 61 beam receive Multi Beam Antenna (MBA), and two 19 beam transmit MBA's. For growth to the DSCS IV system, it was assumed that the growth would be in the number of transponder channels rather than major changes in the new MBA system. For payload sizing, a 50 percent increase in the number of channels was assumed - nine channels instead of the present six channels.

The spacecraft flies in a geostationary orbit (19,323 NM altitude; 0° inclination) and has a 0.08° pointing accuracy requirement in pitch and roll, and 0.8° in yaw. This is required to accurately locate the transmit and receive beams.

4.1.2 NUCLEAR FORCES COMMUNICATIONS SATELLITE (NFCS)

The NFCS (or SSS) is a highly survivable communications satellite system which provides communications to tactical forces. It operates in three frequency bands from UHF to V-band. The satellites are in a nominal 110,000 NM polar orbit. At least four operating satellites will be required for full earth coverage. The pointing accuracy requirement has been assumed to be the same as for the DSCS IV spacecraft.

4.1.3 GENERAL PURPOSE SYNCHRONOUS COMMUNICATIONS SATELLITE (GPSCS)

This spacecraft system is the next generation experimental tactical satellite system. It will carry a variety of communications transponders ranging from UHF up through Q-Band along with multibeam and phased array antenna

TABLE 4.1-1. MISSION SELECTION

MISSION CATEGORY	MISSION NAME	COMMENT
COMMUNICATIONS	DSCS IV	STRATEGIC COMMUNICATIONS
	NFCS (SSS)	SAC COMMUNICATIONS
	GPSCS	ADVANCED TACTICAL COMSAT
SURVEILLANCE	DSSS	EARLY WARNING SYSTEM
NAVIGATION	GPS	GLOBAL NAVIGATION SYSTEM
METEOROLOGICAL	DMSP BLOCK 6	NEXT GENERATION METSAT

systems. This satellite will also be in a geostationary orbit. The pointing requirements are assumed to be the same as for the DSCS IV spacecraft.

4.1.4 DEEP SPACE SURVEILLANCE SYSTEM (DSSS)

This system is the next generation space surveillance and early warning system. The primary payload is an IR sensor presently under definition. The SIRE sensor was used as typical in terms of size, weight, power and pointing requirements to the sensor to be used operationally in the DSSS satellite. The Sensor data will be sent to the ground via an S-Band data communications link. This satellite is assumed to be geostationary with a pointing requirement of 0.01° in all three axes. Additionally, there may be a spacecraft slewing requirement unique to this mission.

4.1.5 GLOBAL POSITIONING SYSTEM (GPS)

The GPS will provide extremely accurate position location capability to all military users. The satellites must be maintained precisely in orbit so that their orbital element changes can be modeled accurately in order that the transmitted "navigation" message accurately describes the satellite position. The satellite system consists of 24 operational satellites (three orbit planes with eight satellites in each orbit plane). The operational orbit is at 10,900 NM with an inclination between 45° and 63° .

4.1.6 DEFENSE METEOROLOGICAL SATELLITE PROGRAM (DMSP) BLOCK 6

The DMSP Block 6 spacecraft will be the shuttle-era version of this operational military satellite system. In addition to a longer life time requirement, the need for survivability in the presence of both nuclear and laser threats is extremely important.

The spacecraft is a nominal 450 NM circular orbit with an inclination of about 98° . Pointing accuracy requirements are about 0.01° in all three axes. The payload complement consists of earth-looking sensors which operate in the visible band microwave, and in several IR bands.

4.2 MISSION REQUIREMENTS

To establish the overall spacecraft bus requirements, each of the selected mission (DSCS IV, NFCS, GPSCS, DSSS, GPS, and DMSP) and their payload complements were defined to the latest available program information. From this, the critical requirements (i.e., pointing accuracy, rate stability, payload power/weight/volume) for each of the missions were determined. This step then led to defining an overall set of bus requirements which would satisfy any of the mission requirements.

4.2.1 MISSION PAYLOAD PRELIMINARY DEFINITION

The mission payload complements shown in Table 4.2-1 are representative of the payload elements for each of the programs considered. For example, the GPS Ø III payload listing is what is currently planned for that system. The DMSP payloads shown are typical of past vehicle payloads with the addition of a Microwave Imager with a 4 foot parabolic antenna. The DSCS IV payload was assumed to be similar to that of DSCS III, with the primary change being an increase in transponder channels from four 10 Watt and two 40 Watt channels to six 10 Watt and three 40 Watt channels. The antenna payloads were assumed identical. The GPSCS and NFCS (formerly SSS) payloads are based on preliminary customer definitions. The SIRE sensor was used for the DSSS payload as being typical in size, weight and power of the to-be defined DSSS payload.

4.2.2 MISSION REQUIREMENTS SUMMARY

Table 4.2-2 summarizes the spacecraft requirements for each of the missions evaluated. As shown, the requirements vary widely among the missions in all areas considered, with the order-of-magnitude difference in the pointing accuracy requirement probably being the most demanding.

4.2.3 NIMS MULTIMISSION REQUIREMENTS

The most demanding requirements from Table 4.2-2 were used to define the NIMS multimission spacecraft requirements shown in Table 4.2-3 to insure

TABLE 4.2-1 MISSION PAYLOAD PRELIMINARY DEFINITION

DMSP	GPS III	DSCS IV	GPSCS	DSSS	NFCS (SSS)
<u>SENSORS</u>	<u>NAVIGATION S/S</u>	<u>COMMUNICATIONS S/S</u>	<u>COMMUNICATIONS S/S</u>	<u>SENSOR</u>	<u>COMMUNICATION S/S</u>
OPERATIONAL LINE SCANNER (2)	SHF RECEIVE ANTENNA	9 CHANNEL TRANSPONDER	SIGNAL PROCESSOR	SIRE	PROCESSORS
MICROWAVE IMAGER	PRN RECEIVER	6 10W. CHANNELS	UHF TRANSPONDER		REPORTBACK
4 FT. ANTENNA	PRNSA	3 40W. CHANNELS	SHF TRANSPONDER		DIGITAL (2)
IR SOUNDER	L-BAND TRANSMITTERS (2)	61 BEAM MBA	(2 CHANNEL)		CONFERENCE
	L-BAND TRANSMIT HELICAL	RECEIVE	K _a -BAND TRANSPONDER		CROSSLINK
	ARRAY ANTENNA	19 BEAM MBA	ANTENNA SYSTEMS	DATA TRANSMITTER	RECEIVERS
		2	UHF RECEIVE	(S-BAND)	UHF
	<u>SCT</u>	TRANSMIT	SHF RECEIVE	S-BAND ANTENNA	X-BAND
MICROWAVE SOUNDER	SHF/UHF UPLINK	GIMBALLED DISH ANTENNA	K _a -BAND RECEIVE		Q-BAND (2)
IONOSPHERIC SENSOR	UHF DOWNLINK	EARTH COVERAGE HORN	Q-BAND RECEIVE		V-BAND (2)
	SIGNAL PROCESSOR	ANTENNAS	Q-BAND PHASED ARRAY		TRANSMITTERS
TOPSIDE IONOSPHERIC		TRANSMIT (2)	(RECEIVE)		UHF
SOUNDER		RECEIVE (2)	SHF 61 BEAM		X-BAND (2)
	<u>IONDS</u>	<u>SCT</u>	MBA RECEIVE		V-BAND (2)
LOW ENERGY GAMMA	VISIBLE SENSOR	SHF/UHF UPLINK	UHF TRANSMIT		ANTENNAS
DETECTOR	X-RAY SENSOR	UHF DOWNLINK	K _a -BAND PHASED		RECEIVE: UHF
TDRSS LINK ANTENNA	L-BAND DOWNLINK	SIGNAL PROCESSOR	ARRAY (TRANSMIT)		X-BAND
	DATA PROCESSOR		K _a -BAND TRANSMIT		Q-BAND (2)
			SHF 19 BEAM		V-BAND (2)
			MBA TRANSMIT		TRANSMIT: UHF
			SHF TRANSMIT		X-BAND (2)
<u>COMMUNICATIONS S/S</u>			<u>SCT</u>		V-BAND (2)
S-BAND DATA TRANS-			SIGNAL PROCESSOR		
MITTERS (4)			UHF RECEIVE		
			ANTENNA/RECEIVER		
			UHF TRANSMITTER		
			& ANTENNA		

TABLE 4.2-2. MISSION REQUIREMENTS SUMMARY

ITEM	SPACECRAFT SYSTEMS					
	GPS ØIII	DSCS IV	SSS	GPSCS	DSSS	DMSP
POINTING ACCURACY						
PITCH	0.1 ⁰	0.08 ⁰	0.08 ⁰	0.08 ⁰	0.01 ⁰	0.01 ⁰
ROLL	0.1 ⁰	0.08 ⁰	0.08 ⁰	0.08 ⁰	0.01 ⁰	0.01 ⁰
YAW	1.0 ⁰	0.8 ⁰	0.8 ⁰	0.8 ⁰	0.01 ⁰	0.01 ⁰
POINTING STABILITY	N/A	N/A	N/A	N/A	0.005 ⁰ /SEC	0.002 ⁰ /SEC.
PAYLOAD POWER, WATTS	605	760	890	860	850*	652
PAYLOAD WEIGHT, LBS.	400	700	640	700	1786	1034
SPACECRAFT POWER, W	905	1060	1190	1160	1150*	952
SPACECRAFT WEIGHT, LBS.	3000	3830	3579	3830	5000	3400
PAYLOAD REQUIREMENTS						
SENSOR VOLUME-FT ³	133	215	168	275	167	203
SENSOR AREA-FT ²	40	50	84	65	33	72
EQUIPMENT VOLUME-FT ³	20	24	25	23	210**	34
ORBIT						
ALTITUDE, N.M.	10,900	19,323	96,615	19,323	19,323	400-500
INCLINATION, DEG.	63	0	90	0	0	99
LAUNCH SYSTEM	STS	STS	STS	STS	STS/IUS	STS
LAUNCH PROPULSION	← PKM/BI-PROP AKM →				NONE	BI-PROP

*DOES NOT INCLUDE REFRIGERATOR POWER REQUIREMENT.

**VOLUME INCLUDES 7 FT³ FOR COMPONENTS + 198 FT³ FOR CMG MODULE + 5 FT³ FOR REFRIGERATOR HEAT SOURCE/EXCHANGER.

TABLE 4.2-3. NIMS MULTIMISSION REQUIREMENTS

ITEM	REQUIREMENTS	COMMENTS
1. POINTING ACCURACY -PITCH/ROLL -YAW	0/08°/0.01° 0.8°/0.01°	• DMSP/DS ³ MISSIONS DRIVE REQUIREMENTS
2. RATE STABILITY	0.002°/SEC.	• REQUIRED FOR DMSP/DS ³
3. PAYLOAD POWER*, WATTS (MAX.)	900	• DS ³ REFRIGERATOR REQUIRES ADDITIONAL 2400 WATTS
4. PAYLOAD WEIGHT*, LBS (MAX.)	1786	• DS ³ MISSION • AVERAGE OF 700 LBS FOR OTHER MISSIONS
5. TOTAL SPACECRAFT POWER, WATTS	1190	
6. TOTAL SPACECRAFT WEIGHT, LBS (IN-ORBIT)	3000 - 5000	• DS ³ • OTHERS 3800 LBS.
7. SURVIVABILITY NUCLEAR LASER	JCS STANDARDS SMATH IV	
8. LAUNCH SYSTEM	STS	• STS/IUS FOR D ³ AND AS BACKUP FOR S ³ (2 AT A TIME)
9. PROPULSION	SOLID ROCKET MOTOR (PKM) WITH BI-PROPELLANT S/S (AKM)	• PROVIDES MEANS TO TRANSFER FROM STS ORBIT TO FINAL ORBIT.

*INCLUDES POWER AND WEIGHT FOR ALL MISSION-UNIQUE EQUIPMENT.

that all the missions considered could be satisfied by one spacecraft bus design. For example, the total spacecraft power requirement is driven by the SSS mission while the maximum weight the structure must be capable of supporting is established by the DSSS mission payload. The attitude control subsystem requirements are driven by the DMSP mission. Since four of the six missions can be satisfied by a pointing accuracy almost an order of magnitude less than that required for DMSP, it was decided to use the reduced accuracy system as the baseline with the addition of star sensors and an IMU to meet the higher accuracy requirements.

SECTION 5

MISSION PAYLOADS

The payload complements shown in Table 4.2-1, Paragraph 4.2.1, Mission Payload Preliminary Definition, were used to determine the payload mounting area/volume, power, thermal, vehicle stability, and fields-of-view requirements and/or constraints.

5.1 SPACECRAFT DESIGN CONSTRAINTS

The primary driver to the overall vehicle design was to satisfy the fields-of-view requirements for the wide variety of payloads considered. A secondary driver was the payload equipment mounting area and volume requirements. Both of these were major influences in the configuration design activity.

5.1.1 FIELDS-OF-VIEW SATISFACTION

Figure 5.1-1, Typical Fields-Of-View (FOV) Studies, shows how through careful arrangement of sensors and/or antennas, all the FOV requirements were met. Shown on the figure is the DSCS IV, DMSP, and NFCS mission payloads FOV's. DSCS IV was selected as typical of the communications-type missions because it had the tightest antenna packaging of these classes of missions.

5.1.2 PAYLOAD MOUNTING AREA/VOLUME REQUIREMENTS

The physical size of both the externally mounted (sensors antennas) and internal panel mounted payload components for the missions considered was determined to set the area and volume requirements.

Table 5.1-1 summarizes the results for each mission. The requirements were

Figure 5.1-1. Typical Fields-of-View (FOV) Studies - NFCS

5-3

Figure 5.1-1. Typical Fields-of-View (FOV) Studies - GPSCS

Figure 5.1-1. Typical Fields-of-View (FOV) Studies - DMSP

Figure 5.1-1. Typical Fields-of-View (FOV) Studies - DSCS IV

TABLE 5.1-1 PAYLOAD MOUNTING AREA/VOLUME REQUIREMENTS

MISSIONS

PAYLOAD REQUIREMENTS	GPS ØIII	DSCS IV	SSS	GPSCS	DSSS	DMSP
SENSOR VOLUME, FT ³	133	215	168	275	167	203
SENSOR AREA, FT ²	40	50	84	65	33	72
EQUIPMENT VOLUME, FT ³	20	24	25	23	210*	34

* Volume includes 7 Ft³ for components + 198 Ft³ for CMG module + 5 Ft³ for refrigerator heat source/exchanger. CMG module is based on presumed slewing requirement. This is a SIRE requirement, but may not be an operational requirement.

determined by defining the payload configurations to satisfy the overall mission requirements. In the case of the payload sensors and antennas, this was based on customer information. The payload electronics (transmitters, receiver, etc.) definition represents customer-provided information and previous in-house activity on these missions. Examples of the latter are shown in Figure 5.1-2 through 5.1-5 which are preliminary block diagrams of the communications subsystems for GPS ØIII, DSCS IV, SSS, and GPSCS. These block diagrams were used to define the various required functions and, from this, determine the physical size of each major component to do this function.

5.2 MISSION PAYLOADS DESCRIPTION AND NIMS INTEGRATION

5.2.1 MISSION PAYLOAD DESCRIPTIONS

Payload characteristics for the alternate NIMS missions have been derived from preliminary program definitions or have been extrapolated from similar existing programs such as DSCS III. The six mission considered cover a wide range of operating orbits, sensors and payload components, and were selected to demonstrate the flexibility of the NIMS system to accommodate varying payload requirements.

5.2.1.1 DMSP (Defense Meteorological Satellilte Program)

The payload, shown on Figure 5.2-1, consists of multiple fixed and deployable sensors with associated sensor electronics and data processing components. The sensor complement selected for the NIMS application is typical of those specified for the advanced DMSP missions, and would vary with each DMSP flight. The primary sensors are the redundant Optical Line Scanners (OLS) which require highly accurate spacecraft pointing and stability, dictating use of a thermally isolated precision mounting platform for the sensors. Secondary sensors include both fixed and deployable instruments also mounted on the precision platform. In addition to the sensors and electronics, a deployable tracking antenna system is required for communication with the TDRSS satellite throughout the mission. The DMSP payload has the most stringent pointing accuracy, control stability

Figure 5.1-2. GPS Navigation Subsystem/TTC/Secondary Payloads
(Block Diagram)

1. EARTH COVERAGE HORN	6. MULTI BEAM ANTENNA (MGA)
2. MULTI BEAM ANTENNA (MBA)	7. MULTI BEAM ANTENNA (MBA)
3. CROSSED UHF BOWTIE ANTENNA	8. CROSSED DIPOLES WITH UHF DIRECTOR
4. EARTH COVERAGE HORN	9. GIMBALLED DISH ANTENNA
5. EARTH COVERAGE HORN	10. EARTH COVERAGE HORN

Figure 5.1-3. DSCS IV Communications Subsystem Block Diagram

Figure 5.1-4. GPSCS Communications Subsystem Block Diagram

* EC — EARTH COVERAGE

Figure 5.1-5. SSS Communications Subsystem Block Diagram

Figure 5.2-1. DMSP Payload Characteristics

and sensor field of view requirements of the alternate payloads evaluated.

5.2.1.2 GPS ØIII (Global Positioning System)

This payload is shown on Figure 5.2-2. This is the smallest and lightest of the alternate NIMS mission payloads. Although overall component heat loads are low, the Cesium clocks require close temperature control and will require thermal isolation from other heat generating components. This mission also requires high precision orbit maintenance and close control of orbital center of gravity movement.

5.2.1.3 DSCS IV (Advanced Defense Satellite Communications System)

Figure 5.2-3 presents a geosynchronous comsat having a large array of multiple fixed antennas. Requirements for DSCS IV have been extrapolated from the current DSCS III system under development by GE Space Division. Overall component heat loads and heat intensities are highest for this type of payload, and will govern design of the NIMS component panels and thermal control/radiator interfaces.

5.2.1.4 NFCS (Nuclear Forces Communications Satellite) or SSS Survivable Strategy Satellite

This is an advanced military comsat having a 24 ft. diameter deployable antenna as shown on Figure 5.2-4. This satellite has a five times synchronous orbit and the highest total heat dissipation of all missions.

5.2.1.5 GPSCS (General Purpose Satellite Communications System) is a next generation tactical comsat having a fixed frequency band average. The weight and installation requirements for this geosynchronous payload, shown on Figure 5.2-5, are similar to DSCS IV.

5.2.1.6 DSSS (Deep Space Surveillance System)

As shown on Figure 5.2-6, this is the largest and heaviest NIMS payload.

Figure 5.2-2. GPS ØIII Payload Characteristics

TOTAL PAYLOAD WEIGHT

ANTENNAS	310.3#
COMPONENTS	373.7#
MISCELLANEOUS	16.0#
TOTAL P/L	700.0#

PANEL MOUNTED COMPONENTS

SCT ELECTRONICS	53.5 # - 115.3W
SHF TRANSPONDER	320.2 # - 654.7W
TOTAL COMPONENTS	373.7# - 770.0W

37" DIA. x 20" L
61 MBA
BASE ENVELOPE

Figure 5.2-3. DSCS IV Payload Characteristics

Figure 5.2-4. NFCS (SSS) Payload Characteristics

TOTAL PAYLOAD WEIGHT

ANTENNAS	319.2#
COMPONENTS	360.9#
MISCELLANEOUS	20.0#
TOTAL	700.1#

PANEL MOUNTED COMPONENTS

SHF TRANSPONDER	64.8# - 126.5W
UHF TRANSPONDER	70.8# - 38.6W
K BAND ELECTRONICS	33.4# - 126.6W
SCT ELECTRONICS	53.5# - 115.3W
SIGNAL PROCESSOR	57.5# - 126.0W
LOCAL OSCILLATORS	80.9# - 42.0W
TOTAL COMPONENTS	360.9# - 575.0W

Figure 5.2-5. GPSCS Payload Characteristics

Figure 5.2-6. DSSS Payload Characteristics

The DSSS is a geosynchronous satellite with the SIRE sensor selected as a typical payload. The assumed high slew maneuver requirements for this mission has dictated addition of a control moment gyro system (CMG's) A separate isotope heat source (IHS) assembly has been included to provide energy for SIRE sensor and the payload radiator provides heat rejection for the refrigerator and CMG module. Payload heat dissipation requirements for the payload panels is 170 Watts full time.

5.2.1.7 Payload Summary

NIMS payloads weight, heat dissipation and volume characteristics as summarized in Table 5.2-1, have been used in developing the NIMS mission configurations and in evaluating the mechanical and thermal designs for payload accommodation. Note that the characteristics include the primary mission sensors, antennas and components plus any mission unique equipment required over and above that provided by the basic NIMS BUS subsystems.

5.2.2 NIMS PAYLOAD COMPONENT ACCOMMODATION

The selected NIMS configuration has an eight section upper bulkhead for mounting payload and housekeeping heat dissipating components. Each section or bay is coupled to the radiator exit header by panel mounted heat pipes for heat rejection via the oversized external DIPS radiator. Each panel section has a gross area of 11.3 FT² and a component mounting area of 10 FT² as shown on Figure 5.2-7.

For conventional spacecraft, components are mounted directly to outboard facing panels and reject heat through passive radiation or through active shutter thermal control systems. The geosynchronous DSCS III and BSE satellites use passive North and South radiator panels housing the components installation parameters summarized in Table 5.2-2.

Values shown in Table 5.2-2 are for mounting on one side of the equipment panel only. For NIMS installation components can be mounted on both sides depending on total panel heat loads and heat pipe arrangements. Average equipment panel heat loads on DSCS III are 500 Watts for the North payload panel and 217 Watts for the South housekeeping panel, indicating thermal requirements are significantly lower for the housekeeping

TABLE 5.2-1 NIMS PAYLOADS CHARACTERISTICS SUMMARY

MISSION	PANEL MOUNTED COMPONENTS *		FRONTAL AREA (FT ²)	SENSORS AND ENVELOPE VOLUME (FT ³)	ANTENNAS HEIGHT (IN.)	WEIGHT (LBS)	TOTAL PAYLOAD WEIGHT (LBS)	UNIQUE REQUIREMENTS
	WEIGHT (LBS)	HEAT DISS. (WATTS)						
DMSP	306.5	404.0	72.0	229.0	78.0	467.5	1,034	<ul style="list-style-type: none"> • INTERNALLY MOUNTED TDRSS ANTENNA • PRECISION MOUNTING PLATFORM • DEPLOYABLE SENSORS
GPS ØIII	330.5	515.5	40.0	133.0	40.0	33.6	389.6	<ul style="list-style-type: none"> • FIXED SENSORS AND ANTENNAS • CESIUM CLOCKS • PRECISION ORBIT
DSCS IV	373.7	770.0	50.0	215.0	68.0	310.3	700.0	<ul style="list-style-type: none"> • LARGE FIXED MULTI-BEAM ANTENNAS • HIGH HEAT LOAD • GEOSYNCHRONOUS
NFCS (SSS)	410.6	779.2	88	254.0	66.0	191.4	638.0	<ul style="list-style-type: none"> • LARGE DEPLOYABLE ANTENNA & FEED • HIGH HEAT LOAD • 5X SYNC. ORBIT
GPSCS	360.9	575.0	65.0	275.0	68.0	319.2	700.1	<ul style="list-style-type: none"> • LARGE FIXED ANTENNAS • SHF & UHF • GEOSYNCHRONOUS
DSSS	63.0	170.0	113.0	521.0	90.0	1195.0	1438.0	<ul style="list-style-type: none"> • CMG CONTROL MODULE • REFRIGERATOR • HEAVIEST PAYLOAD

* CHARACTERISTICS FOR PRIMARY MISSION SENSORS AND COMPONENTS PLUS MISSION UNIQUE EQUIPMENT

TABLE 5.2-2 EXAMPLE CONVENTIONAL SPACECRAFT COMPONENT
MOUNTING CHARACTERISTICS

PANEL	COMPONENT WEIGHT LBS.	MOUNTING AREA	LBS/FT ²	ENCLOSED VOL. FT ³	LBS/FT ³
DSCS NORTH	259.0	24.0	10.8	15.0	17.3
DSCS SOUTH (LESS BATTERIES)	240.0	20.0	12.0	13.0	18.0
BSE NORTH	136.4	11.6	11.8	7.7	7.7
BSE SOUTH (LESS BATTERIES)	87.0	8.0	10.9	5.3	16.4
AVERAGE MOUNTING DENSITY			11.4		17.4

TYPICAL PANEL GEOMETRY
(EIGHT PANELS PER NIMS BUS)

Figure 5.2-7. NIMS Component Panel

equipment panel. The thermal analysis presented in Section 6.3 used the DSCS IV payload components divided into six of the eight equipment bays. The DSCS IV transponder, which uses 40 Watt TWTA's, is representative of the higher heat dissipation payloads elements (97 Watts thermal). Utilizing these TWT units, an example payload with a maximum heat load of 262 Watts was identified for the two bay segment analyzed. Results of the thermal analysis indicated the payload component temperature limits could be maintained with this arrangement. The section analyzed is one quarter of the total bulkhead area. Therefore, four times this thermal load, 1048 Watts, is representative of the total component heat dissipation capacity. This provides design guidelines of 11.6 Watts/FT² for the total panel area and 13.1 Watts/FT² based on component mounting area.

A comparison of the NIMS Thermal analysis results (See Paragraph 6.3) to the DSCS III North and South panels thermal performance is shown in Table 5.2-3. For NIMS Payload panels the allowable packing density is based on the North panel, typically 7.5 panels per square foot. For the NIMS housekeeping panels corresponding to the DSCS III South panel, the typical packing density is 19.3 pounds per square foot based on thermal considerations. For allocation of the NIMS equipment panels, design guideline values of 7.5 pounds per square foot have been used for payload components and 15.0 pounds per square foot for housekeeping components, with payload components mounted predominately on one side of the panel and housekeeping components mounted on both sides. The design point identified for heat dissipation is 131 Watts/panel or 13.1 Watts/FT² based on mounting area.

Internally mounted component total available volume is approximately 100 cubic feet with additional volume available above the panels for externally mounted components. An average installed panel density of 7.5 pounds per cubic foot for payload components and 15.0 pounds per cubic foot for housekeeping components has been used to assess internal volume requirements for NIMS. These values are based on an average internal clearance of 12 inches for all panels.

TABLE 5.2-3 COMPARISON OF NIMS AND DSCS III
PAYLOAD PANEL THERMAL PERFORMANCE

CONFIGURATION	TOTAL PANEL AREA-FT ²	TOTAL HEAT LOAD-WATTS	DISSIPATION WATTS/FT ²
DSCS III NORTH PAYLOAD PANEL	30.0	500	16.7
NIMS PAYLOAD PANEL	22.6	262	11.6
DSCS III SOUTH HOUSEKEEPING PANEL	30	217	7.2

The values of allowable component mounting density (lbs/ft^2) and installed panel density (lbs/ft^2) established for sizing the NIMS spacecraft are lower than those shown in Table 5.2-2 for the conventional DSCS III and BSE spacecraft and compensate for the unique NIMS thermal coupling of the equipment panels to the radiator header by the extensive use of heat pipes on the panels.

NIMS Housekeeping Attitude Control Power Control and TT and C Subsystem components weigh 245 pounds and have a total heat dissipation of 173 Watts as shown in Table 5.2-4.

Based on the identified mounting density of 15 lb/ft^2 for housekeeping components, the required mounting area is 16.4 square feet upper bulkhead panel sections.

The remaining six panel sections provide a total of 60 square feet at 7.5 lb/ft^2 for mounting payload components and a total heat dissipation capacity of 786 Watts minimum at 13.1 Watts/ft^2 . In addition to these 786 Watts, additional capacity of $262 - 173 = 89$ Watts is available from the housekeeping panels providing a total payload heat dissipation capacity of 875 Watts. NIMS alternate mission payload panel mounted components weights, heat dissipations and panel allocations are summarized in Table 5.2-5.

As shown in Table 5.2-5, the selected NIMS preliminary equipment bulkhead design has the capability to accommodate the payload components meeting both mounting area and thermal requirements. This preliminary assessment has been based on the thermal analysis of a typical layout of a high heat dissipation payload having higher heat flux density components, and neither the component arrangement or panel/header inter-faces have been

TABLE 5.2-4 NIMS HOUSEKEEPING SUBSYSTEM PANEL MOUNTED COMPONENTS

SUBSYSTEM	COMPONENT WEIGHT POUNDS	HEAT DISSIPATION WATTS
ATTITUDE CONTROL	75.6	44.5
POWER CONTROL	54.4	10.0
TT & C	115.2	118.0
TOTAL	245.2	172.5

TABLE 5.2-5 NIMS ALTERNATE MISSION PAYLOADS
PANEL MOUNTED COMPONENTS

MISSION	COMPONENT WT. LBS.	DISSIPATION AREA @ 7.5 LB/FT ²	MOUNTING AREA @ 7.5 WT/FT ²	DISSIPATION AREA @ 13.1
DMSP	306.5	404.0	40.9	30.8
GPS ØIII	330.0	515.5	44.0	39.4
DSCS IV	373.7	770.0	49.8	58.8
SSS	410.6	779.2	54.7	59.5
GPSCS	360.9	575.0	48.1	43.9
DSSS	63.0	170	8.7	13.0

optimized for maximum thermal performance. Further optimization and more detailed analysis of each specific payload should result in increased thermal capacity for the system.

In summary, the equipment bulkhead has a total component mounting area of 80 square feet divided into eight bays. Total internal volume is 100 cubic feet with additional volume available above the panels. The total bulkhead thermal dissipation capacity is 1048 Watts minimum. Of these eight panels, a total of six panels or 60 square feet is available for mission payload components and total payload heat dissipation capacity is 875 Watts.

5.2.3 NIMS PAYLOAD SENSOR ACCOMMODATION

The available NIMS sensor volume and frontal area is shown on Figure 5.2-8. For the vertical spacecraft launch installation shown, the maximum allowable payload height is 81 inches measured from the upper equipment mounting bulkhead. For a horizontal spacecraft mounting, the allowable height and volume may be significantly increased. The required sensor accommodations for the candidate NIMS missions are summarized in Table 5.2-6. Note that the areas and volumes tabulated include the payload sensors plus any payload unique equipment required over and above the basic NIMS Bus provisions.

The NIMS Bus, vertically oriented, readily accommodates the sensor and antenna envelopes for all missions except DSSS, necessitating horizontal orientation in the STS cargo bay for this payload. DSSS is the highest weight payload and is launched by the IUS which also requires horizontal payload orientation. The remaining missions are all launched from STS by self-contained bi-propellant or solid rocket propulsion subsystems.

In addition to the launch volume and size requirements, each sensor payload is configured to provide the required fields of view for each sensor or antenna. The field of view capability for NIMS is greatly increased over a conventional solar powered satellite since no solar array is present.

Figure 5.2-8. Availability - NIMS Sensor Area and Volume (Vertical STS Installation)

TABLE 5.2-6 NIMS ALTERNATE MISSION PAYLOAD SENSOR
ENVELOPE REQUIREMENTS

MISSION	SENSOR AREA FT ²	SENSOR VOLUME FT ³	MAXIMUM HEIGHT INCHES	COMMENTS
DMSP	72	229	78	FIXED AND DEPLOYABLE SENSORS & INTERNALLY STOWED TDRSS ANTENNA
GPS ØIII	40	133	40	FIXED ANTENNAS
DSCS IV	50	215	68	FIXED ANTENNAS
SSS	88	254	66	DEPLOYABLE ANTENNAS
GPSCS	65	275	68	FIXED ANTENNAS
DSS	113	521	90	REQUIRES HORIZONTAL LAUNCH BY IUS
AVAILABLE	145	575	81	VERTICAL STS INSTALLATION

5.2.4 NIMS MISSION CONFIGURATIONS

Launch and orbital configurations were developed and are presented along with weight, power and payload summaries for each NIMS mission. Orbital configurations are shown identifying major payload elements. A general arrangement layout for each mission shows launch configuration, sensor and antenna fields of view, and overall configuration dimensions.

NIMS spacecraft are launched on the STS and are deployed at the STS loiter orbit for final orbit insertion by the IUS or the Star 62 Pengie Kick Motor (PKM) and/or the on-board bi-propellant propulsion subsystem. For those missions requiring a PKM, the burned-out PKM motor is ejected by a vee-band clamp and separation springs prior to final insertion (Apogie Kick) provided by bi-propellant propulsion. The PKM is required for the GPS ØIII, DSCS IV, GPSCS and NFCS (SSS) missions, and the IUS is required to inject the DSSS into final orbit. DMSP with its lower orbit is injected using only the on-board bi-propellant propulsion. The bi-propellant subsystem is also utilized by all configurations for on-orbit reaction control.

5.2.4.1 DMSP-NIMS

These configurations are shown on Figures 5.2-9 and 5.2-10; mission weight and power are summarized in Table 5.2-7 and the payload in Table 5.2-8. DMSP is injected into final orbit from the STS by the on-board bi-propellant propulsion subsystem and a PKM is not required, permitting launch stowage of the TDRSS antenna assembly in the empty PKM well as shown on Figure 5.2-10.

5.2.4.2 GPS ØIII

These configurations are presented on Figures 5.2-11 and 5.2-12. Weight and power are presented in Table 5.2-9 and the payload is summarized in Table 5.2-10. GPS ØIII has the lightest weight payload and lowest orbital weight for the missions evaluated.

Figure 5.2-9. DMSP - NIMS Orbital Configuration

Figure 5.2-10. DMSP-NIMS General Arrangement

Figure 5.2-11. GPS III - NIMS Orbital Configuration

Figure 5.2-12. GPS Ø III - NIMS General Arrangement

TABLE 5.2-7 DMSP - NIMS WEIGHT AND POWER SUMMARY

ITEM	WEIGHT, POUNDS	POWER, WATTS
STRUCTURE	375.0	-
DIPS	450.0	-
ATTITUDE CONTROL	106.1	50.5
TELEMETRY, TRACKING AND COMMAND	121.0	120.0
POWER DISTRIBUTION AND CONTROL	182.9	26.0
THERMAL CONTROL	188.0	100.0
PROPULSION (ON-ORBIT)	565.0	2.0
BALLAST & ESD	25.0	-
PAYLOAD AND CONTINGENCY	1387.0	652.0
ON-ORBIT TOTAL	3400.0	950.5
ΔV FUEL (BI-PROPELLANT)	1322.0	-
TRANSFER ORBIT TOTAL	4722.0	-
PKM (NOT REQUIRED)	-	-
SPACECRAFT TOTAL (IN STS)	4722.0	-
AIRBORNE SUPPORT EQUIPMENT	3400.0	-
TOTAL SYSTEM, IN STS	8122.0	-

TABLE 5.2-8 DMSP-NIMS PAYLOAD SUMMARY

KEY ELEMENT/ COMPONENT	WEIGHT (LBS)	MAX. TEMP. LIMITS	THERMAL DATA
			THERMAL DISSIPATION (WATTS)
OPERATIONAL LINE SCANNER (OLS)			
-SENSORS (2)	130	-	10
-ELECTRONICS (REDUNDANT)	10	(1)	15
IR SOUNDER			
-SENSOR	40	-	5
-ELECTRONICS	20	(1)	35
MICROWAVE SOUNDER			
-SENSOR	60	-	45
-ELECTRONICS	20	(1)	15
IONOSPHERE SENSOR			
-SENSOR	5	-	-
-ELECTRONICS	10	(1)	4
TOPSIDE IONOSPHERE SOUNDER			
-SENSOR	23	-	-
-ELECTRONICS	20	(1)	20
MICROWAVE IMAGER			
-SENSOR	130	-	15
-ELECTRONICS	50	(1)	85
LOW ENERGY GAMMA DETECTOR	30	-	25
PLANE LIMB SCANNER			
-SENSOR	40	-	10
-ELECTRONICS	20	(1)	30
COMMUNICATIONS S/S			
-ANTENNAS	2.5	N/A	-
-ELECTRONICS	16.5	(1)	100
PAYLOAD MOUNTING PLATFORM	100	N/A	-
-HEATERS	-	N/A	35
TDRSS ANTENNA SYSTEM			
-ANTENNA SYSTEM	110	N/A	10
-ELECTRONICS	50	(1)	35
CELESTIAL SENSOR ASSEMBLY	7	N/A	< 1.0
SIGNAL PROCESSING SYSTEM	140	(1)	100
INERTIAL MEASUREMENT UNIT (IMU)	(2)	-	37
TOTAL PAYLOAD	1034.0		632.0

TABLE 5.2-8 (CONT.)

1) ALL TEMPERATURES AT COMPONENT MOUNTING INTERFACE

<u>TWT'S:</u>	OPERATIONAL	-1°C to 64°C
	OPERATING QUAL	-1°C to 85°C
	NON-OPERATING QUAL	-34°C to 85°C

<u>OTHER ELECTRONICS:</u>	OPERATIONAL	-13°C to 50°C
	OPERATIONAL QUAL	-34°C to 71°C

2) WEIGHT INCLUDED IN ACS SYSTEM.REQUIRES FULL-TIME USE.

TABLE 5.2-9 GPS ØIII - NIMS WEIGHT AND POWER SUMMARY

ITEM	WEIGHT, POUNDS	POWER, WATTS
STRUCTURE	375.0	-
DIPS	450.0	-
ATTITUDE CONTROL	106.1	50.5
TELEMETRY, TRACKING AND COMMAND	121.0	120.0
POWER DISTRIBUTION AND CONTROL	182.9	26.0
THERMAL CONTROL	188.0	100.0
PROPULSION (ON-ORBIT)	744.0	2.0
BALLAST AND ESD	25.0	-
PAYLOAD AND CONTINGENCY	808.0	605.0
ON-ORBIT TOTAL	3000.0	903.5
AKM FUEL	3532.0	-
TRANSFER ORBIT TOTAL	6532.0	-
PKM (TOTAL)	8445.0	-
SPACECRAFT TOTAL (IN STS)	14977.0	-
AIRBORNE SUPPORT EQUIPMENT	3300.0	-
TOTAL SYSTEM, IN STS	18277.0	-

TABLE 5.2-10 GPS ØIII - NIMS PAYLOAD SUMMARY

KEY ELEMENT/ COMPONENT	WEIGHT (LBS)	THERMAL DATA	
		MAX. TEMP. LIMITS	THERMAL DISSIPATION (WATTS)
● L-BAND HELIX ARRAY ANTENNA SYSTEM	15.0	N/A	10.0
● SHF HORN ANTENNA AND LNA	5.0	N/A	1.0
● SCT RECEIVER ANTENNA	2.0	N/A	0
● SCT TRANSMIT ANTENNA	4.0	N/A	13.0
● IONDS X-RAY DETECTOR	4.5	N/A	2.0
● IONDS VISIBLE DETECTOR	3.1	N/A	2.0
● NAVIGATION PAYLOAD ELECTRONICS	149.5	(1)	289.0
● SCT ELECTRONICS	53.5	(1)	115.3
● CESIUM CLOCKS (3)	72.0	20°C to 45°C	24.0
● IONDS ELECTRONICS	55.0	(1)	87.2
OTHER	26.0		0
TOTALS	389.6		543.5

1) ALL TEMPERATURES AT COMPONENT MOUNTING INTERFACE

TWT'S: OPERATIONAL -1°C to 64°C
 OPERATING QUAL -1°C to 85°C
 NON-OPERATING QUAL -34°C to 85°C

OTHER ELECTRONICS: OPERATIONAL -13°C to 50°C
 OPERATING QUAL -34°C to 71°C

5.2.4.3 DSCS IV

These configurations are shown on Figures 5.2-13 and 5.2-14. Weight, power and payload data are presented in Tables 5.2-11 & 5.2-12. The DSCS IV payload is an increased capacity version of the GE DSCS III payload and has been used as the NIMS baseline for component accommodation analyses.

5.2.4.4 NFCS(SSS)

These configurations are shown on Figures 5.2-15 and 5.2-16. Weight, power and payload data are presented in Tables 5.2-13 & 5.2-14. The NFCS configuration illustrates a significant advantage for NIMS is its capability to accommodate a large deployable antenna or sensor payload without impacting the design and weight of a solar array deployed beyond the antenna periphery. NFCS requires the largest PKM and has the highest STS weight of all missions except the IUS launched DSSS.

5.2.4.5 GPSCS

These configurations are presented on Figures 5.2-17 and 5.2-18. Weight, power and payload data is summarized in Tables 5.2-15 and 5.2-16. The GPSCS configurations, weight and power requirements are similar to DSCS IV.

5.2.4.6 DSSS

These configurations are shown on Figures 5-19 and 5-20, and weight, power and payload data in Tables 5.2-17 and 5.2-18. DSSS is the heaviest NIMS payload and requires launch from STS by the IUS. The assumed slew maneuver requirements have dictated use of a separate CMG on-orbit control module and the SIRE sensor refrigerator cooling system necessitates the addition of a separate isotope heat source for refrigerator power and a separate radiator to reject payload sensor and CMG waste heat. The DSSS configuration illustrates integration of a large complex payload with NIMS and a typical NIMS application with an IUS class payload.

MODULAR EQUIPMENT
PANELS (60 FT²)

- 9 CHANNEL SHF
TRANSPONDER
- SCT ELECTRONICS

GIMBAL DISH
ANTENNA

STC TRANSMIT
ANTENNA

ORBIT

SCT RECEIVE
ANTENNA

61 MBA
SHF RECEIVE
ANTENNA

EARTH

EARTH COVERAGE
TRANSMIT HORNS
(2)

NIMS

TT&C ANTENNA

19 MBA
SHF TRANSMIT
ANTENNA (2)

EARTH COVERAGE
RECEIVE HORNS (2)
AND LNA

Figure 5.2-13. DSCS IV - NIMS Orbital Configuration

DSCS IV / NIMS

Figure 5.2-14. DSCS IV - NIMS General Arrangement

MODULAR EQUIPMENT
PANELS (60 FT²)

- TRABSOIBDER
- IONDS ELECTRONICS

30 IN. DIA. CROSS LINE
2 AXIS GIMBALLED
ANTENNAS (2)

30 IN DIA K-BAND
EARTH LINK
2 AXIS GIMBALLED
ANTENNA

NIMS

30 IN DIA
SHF EARTH LINK
ANTENNA

ORBIT

SHF FEED
(2)

24.3 FT DIA UHF ANTENNA
(REFLECTOR & FEED
DEPLOYABLE)

UHF FEED

10 FT DIA SHF EARTH LINK
ANTENNA APERTURES (2)

EARTH

Figure 5.2-15. NFCS (SSS) - NIMS Orbital Configuration

Figure 5.2-16. NFCS (SSS) - NIMS
General Arrangement

Figure 5.2-17. GPSCS - NIMS Orbital Configuration

Figure 5.2-18. GPSCS - NIMS General Arrangement

Figure 5.2-19. DSSS - NIMS Orbital Configuration

Figure 5.2-20. DSSS - NIMS General Arrangement (with IUS)

TABLE 5.2-11 DSCS IV - NIMS WEIGHT AND POWER SUMMARY

ITEM	WEIGHT, POUNDS	POWER, WATTS
STRUCTURE	375.0	-
DIPS	450.0	-
ATTITUDE CONTROL	106.1	50.5
TELEMETRY, TRACKING AND COMAND	121.0	120.0
POWER DISTRIBUTION AND CONTROL	182.9	26.0
THERMAL CONTROL	188.0	100.0
PROPULSION (ON-ORBIT)	1391.0	2.0
BALLAST AND ESD	25.0	-
PAYLOAD AND CONTINGENCY	991.0	760.0
ON-ORBIT TOTAL	3830.0	1058.5
ΔV FUEL	3141.0	-
TRANSFER ORBIT TOTAL	6971.0	-
PKM (TOTAL)	11300.0	-
SPACECRAFT TOTAL (IN STS)	18271.0	-
AIRBORNE SUPPORT EQUIPMENT & CONTINGENCY	3300.0	-
TOTAL SYSTEM, IN STS	21571.0	-

TABLE 5.2-12 DSCS IV - NIMS PAYLOAD SUMMARY

KEY ELEMENT/ COMPONENT	WEIGHT (LBS)	THERMAL DATA	
		MAX. TEMP. LIMITS	THERMAL DISSIPATION (WATTS)
● SHF RECEIVE MBA	161.3	N/A	<1.0
● SHF TRANSMIT MBA (2)	102.8	N/A	<1.0
● EARTH COVERAGE ANTENNAS (4) (INCLUDES LNA & W(G))	16.0	N/A	-
● GIMBALLED ANTENNA	24.2	N/A	3.0
● SCT ANTENNAS	6.0	N/A	13.0 (TRANSMIT)
● SCT ELECTRONICS	53.5	(1)	115.3
● SHF TRANSPONDER (9 CHANNEL)	320.2	(1)	410.4
● MISCL.	16.0	-	-
TOTALS	700.0		543.7

1) ALL TEMPERATURES AT COMPONENT MOUNTING INTERFACE

TWT'S: OPERATIONAL -1°C to 64°C
 OPERATING QUAL -1°C to 85°C
 NON-OPERATING QUAL -34°C to 85°C

OTHER ELECTRONICS: OPERATIONAL -13°C to 50°C
 OPERATING QUAL -34°C to 71°C

TABLE 5.2-13 NFCS(SSS) NIMS WEIGHT & POWER SUMMARY

ITEM	WEIGHT, POUNDS	POWER, WATTS
STRUCTURE	375.0	-
DIPS	450.0	-
ATTITUDE CONTROL	106.1	50.5
TELEMETRY, TRACKING & COMMAND	121.0	120.0
POWER DISTRIBUTION AND CONTROL	182.9	26.0
THERMAL CONTROL	188.0	100.0
PROPULSION (ON-ORBIT)	1269.0	2.0
BALLAST AND ESD	25.0	-
PAYLOAD AND CONTINGENCY	862.0	890.0
ON-ORBIT TOTAL	3579.0	1188.5
ΔV FUEL	1989.0	-
TRANSFER ORBIT TOTAL	5568.0	-
PKM (TOTAL)	12,760.0	-
SPACECRAFT TOTAL (IN STS)	18,328.0	-
ASE & LAUNCH CONTINGENCY	3,400.0	-
TOTAL SYSTEM, IN STS	21,728.0	-

TABLE 5.2-14 NFCS(SSS) PAYLOAD SUMMARY

KEY ELEMENT/ COMPONENT	WEIGHT (LBS)	<u>THERMAL DATA</u>	
		MAX. TEMP. LIMITS	THERMAL DISSIPATION (WATTS)
24.3' DEPLOYABLE ANTENNA	101.9	N/A	~6
30" GIMBALLED ANTENNA (3)	74.5	N/A	<1.0 TOTAL
30" FIXED ANTENNA (1)	5.4	N/A	<1.0
5" EARTH COVERAGE (REC.)	2.0	(1)	-
TRANSPONDER	355.6	(1)	692
IONDS X-RAY DETECTOR	4.5	N/A	2.0
IONDS OPTICAL DETECTOR	3.1	N/A	2.0
IONDS ELECTRONICS	55.0	(1)	87.2
PAYLOAD SUPPORT	<u>36.0</u>	N/A	<u>-</u>
TOTALS	638.0		791.2

1) ALL TEMPERATURES AT COMPONENT MOUNTING INTERFACE

TWT'S: OPERATIONAL -10°C to 64°C

OPERATING QUAL -10°C to 85°C

NON-OPERATING QUAL -34°C to 85°C

OTHER ELECTRONICS: OPERATIONAL -13°C to 50°C

OPERATING QUAL -34°C to 71°C

TABLE 5.2-15 GPSCS - NIMS WEIGHT AND POWER SUMMARY

ITEM	WEIGHT, POUNDS	POWER, WATTS
STRUCTURE	375.0	-
DIPS	450.0	-
ATTITUDE CONTROL	106.1	50.5
TELEMETRY, TRACKING AND COMMAND	121.0	120.0
POWER DISTRIBUTION AND CONTROL	182.9	26.0
THERMAL CONTROL	188.0	100.0
PROPULSION (ON-ORBIT)	1391.0	2.0
BALLAST AND ESD	25.0	-
PAYLOAD AND CONTINGENCY	991.0	860.0
ON-ORBIT TOTAL	3830.0	1158.5
AKM FUEL	3141.0	-
TRANSFER ORBIT TOTAL	6971.0	-
PKM (TOTAL)	11300.0	-
SPACECRAFT TOTAL (IN STS)	18271.0	-
AIRBORNE SUPPORT EQUIPMENT	3300.0	-
TOTAL SYSTEM, IN STS	21571.0	-

TABLE 5.2-16 GPSCS - NIMS PAYLOAD SUMMARY

KEY ELEMENT/ COMPONENT	WEIGHT (LBS)	THERMAL DATA	
		MAX. TEMP. LIMITS	THERMAL DISSIPATION (WATTS)
SHF RECEIVE MBA	161.3	N/A	< 1.0
SHF TRANSMIT MBA	51.4	N/A	~ 2.0
SHF HORN ANTENNA (3)	7.0	N/A	-
SHF TRANSPONDER	64.8	(1)	126.5
UHF ARRAY ANTENNA	15.0	N/A	< 1.0
UHF TRANSPONDER	70.8	(1)	38.6
K-BAND PHASED ARRAY (2)	58.0	N/A	< 1.0
K-BAND HORN ANTENNA (2)	2.5	N/A	< 1.0
K-BAND GIMBALLED ANTENNA	24.0	N/A	< 1.0
K-BAND ELECTRONICS	33.4	(1)	126.6
SCT ELECTRONICS	53.5	(1)	115.3
SIGNAL PROCESSOR	57.5	(1)	126.0
LOCAL OSCILLATORS	80.9	(1)	42.0
OTHER	20.0		—
TOTALS	700.0		582.0

1) ALL TEMPERATURES AT COMPONENT MOUNTING INTERFACE

TWT'S: OPERATIONAL -1°C to 64°C
 OPERATING QUAL -1°C to 85°C
 NON-OPERATING QUAL -34°C to 85°C
OTHER ELECTRONICS: OPERATIONAL -13°C to 50°C
 OPERATING QUAL -34°C to 71°C

TABLE 5.2-17 DSSS - NIMS WEIGHT AND POWER SUMMARY

ITEM	WEIGHT, POUNDS	POWER, WATTS
STRUCTURE	375.0	-
DIPS	450.0	-
ATTITUDE CONTROL	106.1	50.5
TELEMETRY, TRACKING & COMMAND	121.0	120.0
POWER DISTRIBUTION AND CONTROL	182.9	26.0
THERMAL CONTROL	188.0	100.0
PROPULSION (ON-ORBIT)	1091.0	2.0
BALLAST AND ESD	25.0	-
PAYLOAD AND CONTINGENCY	1861.0*	850.0*
ON-ORBIT TOTAL	4400.0	1148.5
AKM FUEL	IUS WEIGHT = 32,500 LBS + 200 LBS + ADAPTER	-
TRANSFER ORBIT TOTAL		
PKM (TOTAL)		
SPACECRAFT TOTAL (IN STS)	37,100.00	-
AIRBORNE SUPPORT EQUIPMENT & CONTINGENCY	8,500.00	-
TOTAL SYSTEM, IN STS	45,600.00	-

* INCLUDE 500 LBS AND 300 WATTS FOR CMG SYSTEM

TABLE 5.2-18 DSSS PAYLOAD SUMMARY

KEY ELEMENT/ COMPONENT	WEIGHT (LBS)	THERMAL DATA	
		MAX. TEMP. LIMITS	THERMAL DISSIPATION (WATTS)
SIRE SENSOR/ELECTRONICS	510.0 (2)	(1)	130
REFRIGERATOR	-	-	70-2270
SENSOR MOUNT	94	-	
RADIATOR (4)	100	-	
REFRIGERATOR ISOTOPE HEAT UNIT (3)	150	-	
HORN ANTENNA	2.0	N/A	1.0
DATA TRANSMITTER	35.0	(1)	40
CONTROL MOMENT GYRO PACKAGE	500	-	300 (4)
CELESTRIAL SENSOR	7.0	N/A	1.0
OTHER	<u>40.0</u>	-	<u>-</u>
TOTALS	1438.0		242

- 1) ALL TEMPERATURES AT COMPONENT MOUNTING INTERFACES
TWT'S: OPERATIONAL -1°C to 64°C
 OPERATING QUAL -1°C to 85°C
 NON-OPERATING QUAL -34°C to 85°C
OTHER ELECTRONICS: OPERATIONAL -13°C to 50°C
 OPERATING QUAL -34°C to 50°C
- 2) INCLUDES REFRIGERATOR UNIT
- 3) USED TO SUPPLY THERMAL ENERGY TO REFRIGERATOR
- 4) HEAT REJECTED BY SEPARATE PAYLOAD RADIATOR

5.2.5 NIMS MISSION WEIGHTS AND MASS PROPERTIES

An overall weight summary for the alternate NIMS missions is shown on Table 5.2-19, with an expanded breakdown of mission weight in Table 5.2-20. A breakdown of orbital weights is presented in Table 5.2-21, and detail weights of bus subsystems common to all missions are listed in Table 5.2-22. Table 5.2-23 is a breakdown of the bi-propellant subsystem for each mission listing both hardware and fuel weights for each mission configuration. Table 5.2-24 presents a weight breakdown of all payloads sensors and components for the alternate missions. The detail weights for NIMS subsystems were derived from existing programs weight data, such as DSCS III, or estimated from the NIMS preliminary designs developed during the study program. Payload weight data has been taken primarily from current advanced mission requirements generated from preliminary customer inputs for each mission.

Preliminary mass properties were calculated for the NIMS configurations and are presented in Tables 5.2-25 and 5.2-26. The geosynchronous DSCS IV and GPSCS configurations mass properties, Table 5.2-25, show the variation in inertia throughout a typical launch sequence, and estimates of the orbital configuration mass properties for each mission is shown in Table 5.2-26.

TABLE 5.2-19 NIMS ALTERNATE MISSIONS WEIGHT SUMMARY

ITEM	GPS	DSCS IV	GPSCS	DMSP	SSS	DSSS
	WEIGHT IN POUNDS					
NIMS S/S	1,448	1,448	1,448	1,448	1,448	1,448
DRY BI-PROP.	644	644	644	365	511	191
PAYLOAD	400	700	700	1,034	640	1,438
CONTINGENCY	408	291	291	353	222	423
(%)	(16.4)	(10.4)	(10.4)	(12.4)	(8.5)	(13.8)
ORBIT FUEL	100	747	747	200	758	900
S/C ORBITAL WT.	3,000	3,830	3,830	3,400	3,579	4,400
ΔV BI-PROP FUEL	3,532	3,141	3,141	1,322	1,989	(IUS:)
PKM	8,345	1,200	11,200		12,660	(32,500)
LAUNCH ADAPTER	100	100	100		100	200
EJECTED WEIGHT	14,977	18,271	18,271	4,722	18,328	37,100
CRADLE	2,800	2,800	2,800	2,900	2,900	8,000
LAUNCH CONTINGENCY	500	500	500	500	500	500
STS P/L WEIGHT	18,277	21,571	21,571	8,122	21,728	45,600

TABLE 5.2-20 NIMS ALTERNATE MISSION EXPANDED WEIGHT BREAKDOWN

MISSION	GPS	DSCS IV	GPSCS	DMSP	S ³	DSSS
COMMON NIMS S/S	1448.0	1448.0	1448.0	1448.0	1448.0	1448.0
DRY BI-PROP	<u>644.0</u>	<u>644.0</u>	<u>644.0</u>	<u>365.0</u>	<u>511.0</u>	<u>191.0</u>
TOTAL DRY NIMS	2092.0	2092.0	2092.0	1813.0	1959.0	1639
P/L	400.0	700.0	700.0	1034.0	640.0	1438
TOTAL DRY S/L	2492	2792	2792	2847	2599	3077
CONTINGENCY	408	291	291	353	222	423
(% DRY S/C)	(16.4)	(10.4)	(10.4)	(12.4)	(8.5)	(13.8)
S/C & CONTINGENCY	2900	3083	3083	3200	2821	3500
ORBIT FUEL	<u>100</u>	<u>747</u>	<u>747</u>	<u>200</u>	<u>758</u>	<u>900</u>
BOM S/C ORBITAL WT.	3000	3830	3830	3400	3579	4400
V BI-PROP	3532	3141	3141	1322	1989	IUS
PKM PROP	7802	10,657	10,657	-	12,060	32,500
PKM CASE	543	543	543	-	600	
PKM ADAPTER	<u>100</u>	<u>100</u>	<u>100</u>	<u>-</u>	<u>100</u>	<u>200</u>
WT EJECTED	14,977	18,271	18,271	4,722	18,328	37,100
CRADLE	2,800	2,800	2,800	2,900	2,900	8,000
LAUNCH CONTINGENCY	<u>500</u>	<u>500</u>	<u>500</u>	<u>500</u>	<u>500</u>	<u>500</u>
STS P/L WEIGHT	18,277	21,571	21,571	8,122	21,728	45,600

TABLE 5.2-21 NIMS ORBITAL WEIGHT/MISSION

• ORBITAL WEIGHT/MISSION

	GPS	DSCSIV	GPSCS	S ³	DMSP	DSSS
COMMON S/S	1448	1448	1448	1448	1448	1448
DRY PROP.S/S	<u>644</u>	<u>644</u>	<u>644</u>	<u>511</u>	<u>365</u>	<u>191</u>
TOTAL DRY NIMS	2092	2092	2092	1959	1813	1639
BI-PROP ORBIT FUEL	100	747	747	758	200	900
P/L & CONTG.	<u>808</u>	<u>991</u>	<u>991</u>	<u>862</u>	<u>1387</u>	<u>1861</u>
BOM ORBITAL S/C	3000	3830	3830	3579	3400	4400
L/V		STS + STAR 62 PKM + BI-PROP AKM			STS + BI-PROP	STS + IUS

• COMMON BUS SUBSYSTEMS

STRUCTURE	375.0
ACS	106.1
DIPS	450.0 (incl. 98.0 LB RADIATOR*)
S/C RADIATOR	83.0*
BALLAST & ESD	25.0
POWER CON. & DISTR.	182.9
THERMAL CONTROL	105.0
TT&C	121.0
TOTAL COMMON S/S	1448LB

*TOTAL INTEGRATED RADIATOR WEIGHT = 181 LBS.

TABLE 5.2-22 COMMON NIMS BUS SUBSYSTEMS WEIGHT BREAKDOWN

<u>STRUCTURE</u>	
CYLINDER ASSEMBLY	60.0
SKIRT ASSEMBLY	22.0
UPPER BULKHEAD	105.0
LOWER BULKHEAD	85.0
KEELS (8)	64.0
FITTINGS, STRUTS, ATTS.	39.0
TOTAL STRUCTURE	375.0
<u>ACS</u>	
ACE BOX	52.0
IMU	22.4
RATE PACKAGE	1.2
MOMENTUM WHEELS (4)	21.0
EARTH SENSOR	5.5
SUN SENSOR (4)	4.0
TOTAL ACS	106.1
<u>DIPS</u>	
POWER CONVERSION SYSTEM	167.0
HEAT SOURCE ASSEMBLY	185.0
RADIATOR	98.0
TOTAL DIPS	450.0
<u>S/C RADIATOR</u> (INTEGRAL WITH DIPS RADIATOR)	83.0
<u>BALLAST & ESD</u>	25.0

TABLE 5.2-22 (CONT.)

<u>POWER CONTROL & DISTRIBUTION</u>	
POWER CONTROLLER (2)	51.6
ORD. CONTROLLER	2.8
PARASITIC RADIATOR	6.0
HARNESS	<u>122.5</u>
	182.9
<u>THERMAL CONTROL</u>	
HEAT PIPES	30.0
HEATERS & THERMOSTATS	5.0
INSULATION	40.0
THERMAL I/F FITTINGS	25.0
MISCL. H/W	<u>5.0</u>
	105.0
<u>TT&C</u>	
S BAND TRANSPONDER	10.8
SHB-SB DOWN CONV.	7.1
S/IF DOWN CONV.	11.5
BATSON/MURDEN (2)	40.0
DECRYPT/ENCRYPT	17.3
CMD TLM UNIT	21.4
REMOTE TLM UNIT	5.6
DIPLEXER (2)	<u>1.5</u>
TOTAL ELECTRONICS	115.2
ANTENNAS (4)	2.8
LNA	<u>3.0</u>
ANTENNAS & LNA	5.8
TOTAL TT&C	<u>121.0</u>

TABLE 5.2-23 NIMS BI-PROPELLANT PROPULSION SUBSYSTEM
WEIGHT BREAKDOWN

ITEM	WEIGHT/MISSION - LBS					
	GPS	DSCSIV	GPSCS	DMSP	SSS	DSSS
FUEL TANKS	193.0	193.0	193.0	90.0	135.0	45.0
PRESS. TANKS*	316.0	316.0	316.0	168.0	252.0	84.0
HELIUM	22.0	22.0	22.0	12.0	18.0	6.0
TRAPPED PROP.	34.0	34.0	34.0	18.0	27.0	9.0
ΔV THRUSTERS (4)	28.0	28.0	28.0	28.0	28.0	N/R
ACS THRUSTERS (16)	25.0	25.0	25.0	25.0	25.0	25.0
S/S HARDWARE	26.0	26.0	26.0	24.0	26.0	22.0
TOTAL DRY WEIGHT	644.0	644.0	644.0	365.0	511.0	191.0
PROPELLANT	3532	3141	3141	1322	1989	N/R
ORBIT PROPELLANT	100	747	747	200	758	900
TOTAL SUBSYSTEM	4276	4532	4532	1887	3258	1091

*PRESSURANT TANKS SIZED FOR BURST FACTOR OF 4.0

TABLE 5.2-24 NIMS ALTERNATE PAYLOAD WEIGHT BREAKDOWNS

MISSION	SENSORS-WT.LBS	ELECTRONICS WT.LBS
GPS-ØIII TOTAL PAYLOAD 389.6 LBS (400)	L-BAND HELIX ARRAY 15.0 SHF HORN ANTENNA 5.0 SCT Rx ANTENNA 2.0 SCT Tx ANTENNA 4.0 IONDS X-RAY DET. 4.5 IONDS VISIBLE DET. 3.1 TOTAL SENSORS (33.6)	NAV.P/L ELECTRONICS 149.5 SCT ELECTRONICS 53.5 CESIUM CLOCKS(3) 72.0 IONDS ELECTRONICS 55.0 TOTAL ELECTRONICS (330.0)
	MOUNTS & MISCL	(26.0)
DSCS IV (9 CHANNEL SHF TRANSPONDER) TOTAL PAYLOAD (700 LBS)	SHF Rx MBA 161.3 SHF TxMBA (2) 102.8 EC ANT(4)+LNA 16.0 GIMBALLED ANTENNA 24.2 SCT ANTENNAS 6.0 TOTAL SENSORS (310.3)	SCT ELECTRONICS 53.5 SHF TRANSPONDER 320.2 TOTAL ELECTRONICS (373.7)
		MOUNTS & MISCL (16.0)
GPSCS TOTAL PAYLOAD 700.1 LBS (700)	SHF Rx MBA 161.3 SHF Tx MBA 51.4 SHF HORN (3) 7.0 UHF ARRAY 15.0 K BAND Ø ARRAY (2) 58.0 K BAND HORN (2) 2.5 K BAND GIMBALLED ANT. 24.0 TOTAL SENSORS (319.2)	SHF TRANSPONDER 64.8 UHF TRANSPONDER 70.8 K BAND ELECTRONICS 33.4 SCT ELECTRONICS 53.5 SIGNAL PROCESSOR 57.5 LOCAL OSCILLATORS 80.9 TOTAL ELECTRONICS (360.9) MISCL. (20.0)
SSS(S ³) TOTAL PAYLOAD 638 LBS (640)	24.3'DEPLOYABLE ANT. 101.9 30"GIMBALL ANT.(3) 74.5 30" FIXED ANTENNA 5.4 5" EC Rx ANTENNA 2.0 IONDS X-RAY DET. 4.5 IONDS VISIBLE DET. 3.1 TOTAL SENSORS (191.4)	TRANSPONDER 355.6 IONDS ELECTRONICS 55.0 TOTAL ELECTRONICS (410.6)
	MOUNTS & MISCL	(36.0)

TABLE 5.2-24 (CONT.)

MISSION	SENSORS WT. LBS		ELECTRONICS WT. LB	
<u>PAYLOAD EQUIPMENT</u>				
DMSP-BLK6	OPTICAL LINE SCAN. (2)	130.0	OLS ELECTRONICS	10.0
	IR SOUNDER	40.0	IR SOUND. ELECT.	20.0
	MICROWAVE SOUNDER	60.0	M.W. SOUND. ELECT.	20.0
	IONOSPHERE SENSOR	5.0	ION. SEN. ELECT.	10.0
TOTAL	TOPSIDE IONOS. SOUND.	23.0	TOP. ION. SOUND.	
PAYLOAD (1034 LBS)			ELECT.	20.0
	MICROWAVE IMAGER	130.0	M/W.I. ELECT.	50.0
	LOW ENG. GAMMA DET.	30.0	(NONE)	
	PLANE LIMB SCAN.	40.0	P.K.S. Elect.	20.0
	TOTAL P/L SENSORS	(458.0)	SENSOR ELECT.	(150.0)
<u>MISSION UNIQUE EQUIPMENT</u>				
	COMM. S/S ANTENNAS	2.5	COMM. S/S ELECT.	16.5
	PAYLOAD MTG. PLATFORM	100.0	SIGNAL PROG. SYS.	140.0
	TDRSS ANTENNA ASSEM.	110.0	TDRSS ELECT.	50.0
	CELESTIAL SENSOR	7.0	IMU (22.4 INCL.	
			IN. BUS	
	TOTAL SUPPORT SENSORS	(219.5)	TOTAL SUPPORT	
			ELECTRONICS	(206.5)
<u>PAYLOAD EQUIPMENT</u>				
DSSS	SIRE SENSOR	482.0		
	SENSOR MOUNT	94.0	DIG. PROCESSING	
			UNIT	28.0
	HORN ANTENNA	2.0	DATA XMTR.	35.0
TOTAL	CELESTIAL SENSOR	7.0	PLUMBING/CABLES	30.0
PAYLOAD	DUST COVER	10.0		
	SENSOR RADIATOR	100.0		
	TOTAL SENSORS	(695.0)	TOTAL SENSOR	
			EQUIP.	(93.0)
(1973 LBS)	<u>MISSION UNIQUE EQUIPMENT</u>			
	CMG MODULE (500.0)	REFRIG. ISOTOPE HEAT UNIT		(150.0)

TABLE 5.2-25 NIMS MASS PROPERTIES FOR GEOSYNCHRONOUS MISSION

CONDITION	WEIGHT LBS.	$\bar{Z}$ CG IN.	MOMENT OF INERTIA, SLUG-FT ²		
			I_x	I_y	I_z
LAUNCH	18,271	18.0	7989	7119	5709
1/2 PKM BURN	12,261	22.3	6025	5159	5138
PKM B/O	7,614	29.3	4130	3260	4618
PKM EJECT (START AKM)	6,971	31.1	3862	2992	4538
AKM B/O (ORBIT BOM)	3,830	36.2	2327	2403	2705

TABLE 5.2-26 NIMS MASS PROPERTIES FOR ORBITAL
CONFIGURATIONS

MISSION CONFIGURATION	ORBITAL WEIGHT (LBS)	Z.C.G.* (INCHES)	INERTIAS, SLUG-FT ²		
			I _x (ROLL)	I _y (PITCH)	I _z (YAW)
DSCS IV GPSCS	3830	36.2	2327	2403	2705
GPS ØIII	3000	32.7	1820	1820	1720
DMSP	3400	31.6	2500	2430	2415
NFCS (SSS)	3579	36.8	2329	2400	2604
DSSS	4400	52.4	3600	3360	3110

* Z.C.G. MEASURED ABOVE LOWER BULKHEAD

SECTION 6

NIMS DEFINITION

This section summarizes the NIMS configuration development and provides design descriptions of the basic NIMS housekeeping subsystems.

6.1 CONFIGURATION OPTIONS AND SELECTION

The NIMS spacecraft has been designed using the DIPS nuclear power generation subsystem. An external radiator is used with DIPS and is oversized to dissipate spacecraft, waste heat for thermal control of payload and housekeeping subsystem components via heat pipe coupling. Total radiator area is 237.5 square feet, approximately half for DIPS and half for equipment cooling.

Initial estimates of propulsion requirements for alternate missions indicated internal space would be required to centrally house the 62-inch diameter STAR 62 solid rocket Perigee Kick Motor (PKM) and a radial clearance of 35 inches is required to accommodate bi-propellant subsystem tankage.

An initial assessment of payload and housekeeping internal equipment indicated a total panel area of 66 square feet should be allocated for components. Subsequent detail thermal analysis increased this total area allocation to 80 square feet versus.

Another requirement was to reduce the height of the NIMS spacecraft body to provide maximum clearance for payload sensor with the spacecraft mounted

vertically in the 15-foot diameter STS cargo bay.

A comparison of cylindrical, square and octagonal NIMS configurations against these requirements is shown in Table 6.1-1. A 144-inch (12 foot) diameter cylindrical configuration having a height of 76 inches met all requirements with ample margin. It was compared to equal height square and octagonal shapes. Of these three equal height configurations only the cylindrical and octagonal shapes met all parameters. A comparison of the radiator and heat pipe design for these two shapes showed the cylindrical design simplest for both the radiator and heat pipes, leading to its selection for NIMS. Not that the selected cylindrical configuration has the highest component bulkhead mounting area and highest packaging efficiency of all configurations. A comparison of the selected configuration to the selection requirements is shown in Table 6.1-2.

The general arrangement of the selected NIMS configuration for vertical orientation in the STS is shown on Figure 6.1-1, and an exploded view illustrating the major spacecraft bus subassemblies is presented on Figure 6.1-2. These figures show the location and volume allocations for payloads and subsystems. NIMS Bus characteristics for each subsystem are summarized on Figure 6.1-3, and a summary of housekeeping subsystem component weights, heat dissipations and temperature limits is shown in Table 6.1-3. These bus subsystems are described in detail in subsequent sections.

Table 6.1-4 is a total weight and power breakdown for a typical NIMS geosynchronous mission, and weight and power data for other missions is presented in Section 5.2.4. Figure 6.1-4 illustrates NIMS installed vertically in the STS and shows the overall dimensions and clearances including the launch support cradle. This vertical orientation is applicable for all missions evaluated except DSSS which requires launch from STS by the IUS after deployment from a horizontal orientation as previously shown on Figure 5.2-20. In the vertical configuration and spacecraft is separated by release of the cradle separation vee-band and ejected vertically by cradle attached separation springs.

NIMS orbital configurations previously described in Section 5.2.4 are

TABLE 6.1-1 NIMS CO MURATION COMPARISON

CONFIGURATION PARAMETER	CYLINDRICAL			SQUARE	OCTAGONAL

WIDTH - INCHES	110	136	144	113	136.4
HEIGHT - INCHES	98.5	80	76	76	76
VERTICAL CLEARANCE IN STS - INCHES	60.5	66.4	65.3	65.3	65.3
INTERNAL CLEARANCE INCHES (35 th REQ'D)	23	36	40	24.5	36.2
COMPONENT BULKHEAD MOUNTING AREA-FT ² (66 FT ² REQ'D)	38	69	80	56	74
INTERNAL VOLUME- FT ³	387	550	608	449	562
STRUCTURE & RADIATOR WEIGHT-POUNDS	485	535	555	500	540
VOLUME FT ³ WEIGHT-POUND	.80	1.03	1.09	.90	1.04
RADIATOR	CYLINDER WITH CONSTANT CURVATURE PIPES-SIMPLEST DESIGN			FLATS WITH MULTIPLE BENDS - MORE COMPLEX DESIGN	
HEAT PIPES	EQUAL LENGTH SIMPLEST DESIGN			VARIABLE LENGTHS MORE COMPLEX DESIGN	

TABLE 6.1-2 NIMS CONFIGURATION SELECTION

- CONFIGURATION REQUIREMENTS

- PROVIDE 237.5 FT² INTEGRATED RADIATOR AREA
- HOUSE 62 INCH DIAMETER SOLID ROCKET PERIGEE KICK MOTOR (PKM)
- PROVIDE 66 FT² (MINIMUM) UPPER BULKHEAD COMPONENT MOUNTING AREA (HEAT PIPE COUPLED TO RADIATOR)
- PROVIDE 35 INCH MINIMUM INTERNAL CLEARANCE
- PROVIDE MAXIMUM VERTICAL CLEARANCE IN STS

- CONFIGURATION SELECTED

- 144" (12 FT) DIAMETER RADIATOR - 76 INCHES HIGH
- CENTRAL 64" DIAMETER CYLINDER WITH 8 RADIAL KEELS - CONTAINS PKM AND PROVIDES PRIMARY LAUNCH LOAD PATH
- MODULAR UPPER BULKHEAD PROVIDES 80 FT² FOR COMPONENT MOUNTING
- 20 FT² (2 BAYS) USED FOR HOUSEKEEPING COMPONENTS - 60 FT² (6 BAYS) RESERVED FOR PAYLOAD COMPONENTS
- STS VERTICAL CLEARANCE 65.3 INCHES FROM RADIATOR RIM (80 INCHES FROM RECESSED UPPER BULKHEAD)
- HIGHEST VOLUMETRIC EFFICIENCY ($\frac{\text{FT}^3}{\text{LB}}$) AND SIMPLIEST RADIATOR AND HEAT PIPE DESIGNS

TABLE 6.1-3 NIMS BUS HOUSEKEEPING SUBSYSTEM
COMPONENTS SUMMARY

SUBSYSTEM	WEIGHT (LBS)	HEAT DISSIPATION (WATTS)
<u>ATTITUDE CONTROL</u>	(106.1)	(50.5)
Sensors (Earth, Sun)	9.5	4.0
*Electronics (ACE, IMU)	75.6	44.5
Reaction Wheels (4)	21.0	2.0 (Avg.)
<u>TT&C</u>	(121.0)	(120.0)
Antennas/LNA	5.8	2.0
*Electronics	115.2	118.0
<u>POWER DISTRIBUTION & CONTROL</u>	(182.9)	(26.0)
Harness	122.5	16.0 (Distr. Loss)
*Switching Electronics	54.4	10.0
Parasitic Radiator	6.0	-
<u>THERMAL CONTROL</u>	(105.0)	(100.0)
Heaters/Thermostats	5.0	100.0 (Max)
Insulation	40.0	-
Heat Pipes	30.0	-
Thermal I/F Fittings	25.0	-
Misc1.	5.0	-
TOTAL HOUSEKEEPING COMPONENTS	(515.0)	(296.5)

* ALLOWABLE TEMPERATURES AT COMPONENT MOUNTING INTERFACE:

OPERATIONAL: -13°C to 50°C
OPERATING QUAL: -34°C to 71°C

TABLE 6.1-4 NIMS WEIGHT AND POWER SUMMARY
(TYPICAL GEOSYNCHRONOUS MISSION)

ITEM	WEIGHT (LBS)	POWER (WATTS)
<u>NIMS BUS</u>		
Structure	375.0	-
DIPS (Including 9818 DIPS		
Radiator	450.0	-
Spacecraft Radiator	83.0	-
ACS (Attitude Control)	106.1	51
TT&C	121.0	120
Power Dist. & Control	182.9	26
Thermal Control	105.0	100
Dry Bi-Propellant Propulsion	644.0	-
Ballast & ESD	25.0	-
TOTAL DRY NIMS BUS	2,092.0	297
ON-ORBIT FUEL	747.0	-
PAYLOAD & CONTINGENCY	991.0	903
TOTAL ORBITAL SPACECRAFT	3,830.0	1200 ⁽¹⁾
BI-PROPELLANT AKM FUEL	3,141.0	-
STAR 62 PKM	11,200.0	-
PKM ADAPTER	100.0	-
TOTAL WEIGHT EJECTED FROM STS	18,271.0	-
LAUNCH CRADLE	2,800.0	-
LAUNCH CONTINGENCY ⁽²⁾	500.0	-
TOTAL STS PAYLOAD WEIGHT ⁽³⁾	21,571.0	1200

1) DIPS POWER AT 10 YEARS END OF MISSION

2) COOLING DURING LAUNCH HANDLED BY ORBITER ATCS. DIPS INTEGRATION INCLUDED IN LAUNCH CONTINGENCY.

3) STS P/L LENGTH = 15 FT MAX (VERTICAL S/C INSTL.)

Figure 6.1-1. NIMS Bus General Arrangement

Figure 6.1 NIMS Bus Major Sub-Assemblies

Figure 6.1-3. NIMS Bus Characteristics

shown on Figure 6.1-5, summarizing the flexibility of the basic NIMS Bus in accommodating a wide variety of mission payloads and orbits.

6.2 STRUCTURE

6.2.1 STRUCTURAL ARRANGEMENT AND REQUIREMENTS

The NIMS spacecraft structure, shown on Figure 6.2-1, has been designed to meet STS loads and environments installed in either a vertical or horizontal orientation. For vertical installation the structure is attached to an ASE cradle fitting by a vee-band separation clamp. The horizontal attitude is for the spacecraft attached to an IUS adapter using the same vee-band separation interface. The structure has been sized for the maximum spacecraft weights presented in Section 5.2.5.

The structural arrangement is a central primary structural cylinder designed to resist all body axial, shear and bending loads. Eight radial keels and upper and lower bulkheads are used to support internal equipment and redistribute their loads into the primary cylinder. The non-structural radiator closes the outer spacecraft shell and attachments at the keels and bulkheads are designed to permit both radial and axial expansion relative to the structure.

The structure is modular with separable equipment panels to permit parallel fabrication, assembly and testing of subsystems and payloads. The structure is constructed of conventional aluminum and magnesium alloys and uses high modulus graphite-epoxy materials for struts, keels, and the center cylinder shell. These materials are all state-of-the-art, and require no new development for the NIMS applications. All materials including the graphite-epoxy are currently in use on the DSCS-III spacecraft being constructed for the Air Force by GE.

6.2.2 STRUCTURAL LOADS AND STRESS ANALYSIS

NIMS structural load paths are illustrated on Figure 6.2-2, for spacecraft configurations with and without a Perigee Kick Motor (PKM), also shown

**DEFENSE SYSTEM
COMMUNICATIONS
SATELLITE-IV**

**DEEP SPACE
SURVEILLANCE SYSTEM**

**GLOBAL POSITION
SYSTEM - PHASE III**

**NUCLEAR FORCES
COMMUNICATIONS
SATELLITE**

**DEFENSE METEOROLOGICAL
SATELLITE PROGRAM**

**GENERAL PURPOSE
SYNCHRONOUS
COMMUNICATIONS
SATELLITE**

Figure 6.1-5. NIMS Orbital Configuration

- STRUCTURE DESIGNED TO MEET STS LOADS AND ENVIRONMENTS FOR VERTICAL OR HORIZONTAL LAUNCH ORIENTATION
- STRUCTURE SIZED FOR MAXIMUM WEIGHT SPACECRAFT
- CENTRAL CYLINDER PROVIDES EFFICIENT DIRECT LOAD PATH FOR SPACECRAFT AND PKM - LOW INDUCED RADIATOR LOADS
- MODULAR ARRANGEMENT FOR PARALLEL ASSEMBLY AND TEST OF SUB-SYSTEMS AND PAYLOAD
- STATE OF THE ART MATERIALS AND CONSTRUCTION

Figure 6.2-1. NIMS Spacecraft Structure

Figure 6.2-2. NIMS spacecraft Load Paths and Separation Interfaces

are maximum spacecraft weights for these two conditions. The maximum weight with a PKM is for the DSCS IV or GPSCS aposynchronous missions, and the maximum weight without a PKM is for IUS launch of the DSSS. The NIMS structural arrangement and load paths are similar to those used by GE on the geosynchronous BSE satellite and by Fairchild for their Nuclear GPS Organic Rankine configuration.

Design limit load factors used for preliminary structural analysis were obtained from the McDonnell-Douglas STS-PAM-D User's Requirements Document for vertical launch orientation and are shown in Table 6.2-1. Load factors for horizontal launch conditions, shown in Table 6.2-3, are from Boeing IUS data. The spacecraft limit load factors for both orientations include a preliminary assessment of spacecraft launch dynamics effects and are the most recent data available. For design ultimate loads an ultimate factor of safety has been used, and the resulting axial, shear and bending moments at the spacecraft separation interface are shown in Table 6.2-3.

Stresses were checked in the aluminum separatin interface ring and the graphite-epoxy cylinder, points (A) and (B) shown on Figure 6.2-3.

For point (A), the 0.15 inch thick 7075 aluminum interface ring, the running peak running load (W) from the axial load and bending moment is ± 729 lb/in. For the vertical condition and 694 lb/in. for the horizontal. This load also produces unit bending moment. (M) of 273.4 in.-lb/in. due to the 0.375 inch eccentricity. Allowables for the ring are $F_{tu} = 70$ KSI, and $FBU = 100$ KSI. for vertical load condition:

$$f_t = \frac{729}{0.15} = 4.86 \text{ KSi}, R_t = \frac{f_t}{F_{Tu}} = \frac{4.86}{70} = 0.07$$

$$f_b = \frac{6(273.4)}{(.15)^2} = 72.9 \text{ KSi}, R_B = \frac{72.9}{100} = 0.73$$

$$M.S. = \frac{1}{R_T + R_B} - 1 = \frac{1}{0.07+0.73} - 1 = +.25$$

Table 6.2-1. Spacecraft Design Load Factors (S/C Vertical)
(STS PAM-D)

Condition	Spacecraft Weight	Limit Load Factors (G's)	
		Lateral	Axial
Maximum Lateral	544-907 Kg (1200-2000 lb)	5.2	3.5
	>907 Kg (2000 lb)	5.1	3.3
Max Axial (Compression) See Note 6	544-907 Kg (1200-2000 lb)	3.6	5.4
	>907 Kg (2000 lb)	3.5	5.1
Max Axial (Tension)	544-907 Kg (1200-2000 lb)	5.2	-3.5
	>907 Kg (2000 lb)	5.1	-3.3

Values Given are in PAM/Spacecraft Axes

Notes

- 1 Subscripts p are PAM/spacecraft axes. Subscripts o are STS axes
- 2 Y Y axes are into plane of paper
- 3 Positive axial load factor results in compression
Negative axial load factor results in tension
- 4 Lateral load factors given are resultant of Z and Y and can act in any lateral direction on spacecraft.
- 5 Use factor of 1.5 for ultimate loads

MDAC G6626A, May 1978

Table 6.2-2. IUS Design Limit Load Factors for Spacecraft Design (S/C Horizontal)

Axis	Steady state load factors	Spacecraft dynamic load factors- spacecraft weight and center of gravity		
		1,950 lb at 67 in	6,000 lb at 130 in 5,000 lb at 162 in	5,000 lb at 70 in
X	-1.5	±3.5	±3.5	±3.0
Y	-	±2.4	±2.0	±2.4
Z	+1.0	±4.0	±2.0	±3.5

1 Lateral load factors (Y and Z) are applicable to spacecraft with cantilever frequencies of 7 Hz or greater. Spacecraft with lower frequencies may have significantly higher lateral loads.

2 Spacecraft center of gravity location shown in sketch (D)

3 Lift off condition

4 Landing condition

Boeing Data, Dec. 1978

TABLE 6.2-3 NIMS STRUT ANALYSIS SUMMARY

LAUNCH CONFIGURATION	LIMIT LOAD FACTORS (S/C AXES)	ULTIMATE LOADS* AT S/C SEPR. I/F	STRESS ANALYSIS	
			I/F FITTING	SHELL
STS Launch Vertical S/C Orientation WT S/C=6,971 LBS C.G. 33.6 IN. ABOVE S/C SEPARATION I/F	$N_x(\text{AXIAL}) = \pm 3.3g$ $N_{y-z}(\text{LATERAL}) = \pm 5.1g$ MAX. LATERAL LOADS FOR PAM-D-REF (1)	$P_{axial} = \pm 34,506 \text{ LBS}$ $M = 1,791,826 \text{ IN LBS}$ $SHEAR = \pm 53.328 \text{ LBS}$	$W = \pm 729 \#/\text{IN}$ $M.S. = \pm .25$	$W = \pm 643 \#/\text{IN}$ $M.S. = \pm .19$
STS-IUS Launch Horizontal Orientation WT S/C = 4400 LBS C.G. 605 IN. FWD of S/C Separation I/F	$N_x(\text{AXIAL}) = \pm 3.5g$ $N_{y-z}(\text{LATERAL}) = \pm 4.2g$ IUS Dynamic Load Factors-REF(2)	$P_{axial} = \pm 25,200 \text{ LBS}$ $M = 1,829,520$ $SHEAR = \pm 30,240 \text{ LBS}$	$W = 694 \#/\text{IN}$ $M.S. = \pm .31$	$W = \pm 646 \#/\text{IN}$ $M.S. = \pm .18$

* DESIGN ULTIMATE FACTOR OF SAFETY = 1.5, LOADS AT 64" DIA. SPACECRAFT SEPARATION INTERFACE.

REF. (1) PAM-D USER'S REQUIREMENTS DOCUMENT, MDAC G6626A, MAY 1978.

REF. (2) BOEING IUS PERFORMANCE AND CONFIGURATION DATA - DEC. 1978

Figure 6.2-3. NIMS Spacecraft Cylinder and Separation Joint Structure

At point (A) for the horizontal condition:

$$f_t = \frac{694}{0.15} = 4.63 \text{ KSi}, R_T = 0.066$$

$$f_b = 69.4 \text{ KSi}, R_B = 0.694$$

$$M.S. = \frac{1}{0.066+0.694} - 1 = + .31$$

At point (B) the 0.075 inch thick graphite epoxy cylinder, the running loads (W) are 643 lb/in. for the vertical condition and 646 lb/in. for the horizontal. The compression allowable for an $R/t = 32/.075 = 427$ is 10.2 KSi. The 10.2 allowable is for straight compression and is conservative for cylinders in bending where $F_b \approx 1.3 F_c$. The allowable running load for the cylinder is $W_c = (10,200) 0.075 = 765 \text{ lb/in.}$
For Vertical Condition (W = 643 LB/IN.)

$$M.S. = \frac{765}{643} - 1 = + .19$$

For Horizontal Condition (W = 646 LB/IN.)

$$M.S. = \frac{765}{646} - 1 = + 0.18$$

Based on these preliminary loads and stress analyses the primary structure as sized for the NIMS spacecraft, shows positive margins of safety for both vertical and horizontal orientations and at maximum spacecraft weights for both cases.

6.2.3 STRUCTURAL WEIGHTS

Based on the preliminary loads and stress analyses and geometry of the NIMS structure, weight estimates were made for the structural elements and the weight breakdown is presented in Table 6.2-4. Also tabulated are the materials and construction for each element.

TABLE 6.2-4 NIMS BUS STRUCTURE WEIGHT BREAKDOWN

<u>ELEMENT</u>	<u>MATERIAL/CONSTRUCTION</u>	<u>WEIGHT(POUNDS)</u>
• CYLINDER ASSEMBLY		(60.0)
- SHELL	MONOCOQUE GRAPHITE-EPOXY	40.0
- RINGS	MACHINED ALUMINUM	11.0
- LONGITUDINALS (8)	ALUMINUM EXTRUSION	9.0
• SKIRT ASSEMBLY		(22.0)
- WEB	MONOCOQUE GRAPHITE-EPOXY	9.0
- HEADER SUPPORTER	MACHINED ALUMINUM	13.0
• UPPER BULKHEAD		(105.0)
- PANELS	MACHINED MAGNESIUM	98.0
- BEAMS	BUILT-UP GRAPHITE-EPOXY	7.0
• LOWER BULKHEAD		(85.0)
- PANEL	GRAPHITE-EPOXY SANDWICH	60.0
- CRUCIFORM RING	MACHINED ALUMINUM	25.0
• KEELS (8)	GRAPHITE-EPOXY SANDWICH	64.0
• FITTINGS, STRUTS AND ATTACHMENTS	MACHINED ALUMINUM AND MAGNESIUM FITTINGS GRAPHITE-EPOXY STRUTS	(39.0)
TOTAL STRUCTURE		375 POUNDS

6.3 THERMAL CONTROL

6.3.1 THERMAL INTEGRATION CONCEPT

Thermal control of NIMS payload and housekeeping components is maintained by coupling the component mounting panel to exit (cold) header of the external radiator with heat pipe as shown conceptually on Figure 6.3-1. In this integrated thermal concept heat from components are base mounted internally or externally to the equipment panels and component heat is dissipated to the panels and conducted to the external radiator cold header segments by radial panel mounted heat pipes. The 12 foot diameter radiator has four cold header segments as shown on Figure 6.3-2.

The NIMS TCS is implemented by sizing the DIPS radiator such that the working fluid is sub-cooled below the minimum temperature required by the DIPS design point cycle conditions. The sub-cooled fluid is then capable of accepting the heat load from the payload and housekeeping electronics cooling. This heat load, dumped into the fluid at the radiator exit (cold) header, raises its temperature to that required by the DIPS design point cycle conditions.

This **concept** is applicable for components with normal temperature limits of -13°C to $+50^{\circ}\text{C}$ (or higher), such as the electronics commonly mounted on spacecraft direct radiation equipment panels. Components requiring low relative temperatures such as low noise amplifiers, must as usual, be located off the panel and radiate directly to space with heaters and thermostats for fine temperature control.

The two limits that must be considered in the design of the NIMS TCS are:

- 1) The minimum temperature of the DIPS working fluid at the radiator exit (cold) header for normal operations. This design point is established at 80°F (27°C) by consideration of the lower temperature limit on the Dowtherm A working fluid (freezing temperature of 53.6°F (12.0°C)).

Figure 6.3-1. NIMS Thermal Integration Concept

Figure 6.3-2. NIMS Radiator Configuration

- 2) The maximum thermal energy from payload and housekeeping electronics cooling that can be added to the sub-cooled DIPS working fluid before the fluid is returned to the DIPS jet condenser must not result in temperatures greater than 212°F, as required by the reference DIPS design. This thermal load is established at about 1050 Watts(t) maximum for the NIMS design presented in this report. This limit can be changed since it is primarily related to the DIPS power level (fluid flow rate). Minor changes may be accommodated utilizing changes in the DIPS fluid management.

These two limits govern the thermal control for the payload and housekeeping electronics achievable for the NIMS concept during normal operation (no threat encounter).

Items which must also be considered in the thermal design of the NIMS Bus, which are not true limits, include:

- 1) The placement, spacing and arrangement of the payload and electronics components on the eight mounting panels to achieve a uniform and manageable thermal load for each panel.
- 2) The number and thermal capacity of the heat pipes that couple these mounting panels to the DIPS radiator exit (cold) header.
- 3) The technique used to attach these heat pipes to the DIPS radiator exit (cold) header, and thermal conductance achievable.

The initial results presented here verify the ability of the NIMS TEC to provide payload and housekeeping electronics cooling during both normal and threat encounter environments. Additional optimization that would be accomplished during the NIMS detailed design and development is expected to result in improved performance of this TCS, over that identified and presented at this time.

The key to the NIMS enhanced survivability to the laser threat is the integration of the Dynamic Isotope Power System such that cooling is provided for payload and housekeeping electronics. Basically, utilization of the DIPS pumped coolant loop permits thermal energy management capability to be incor-

porated into the NIMS design. The incident thermal energy absorbed is managed by a combination of the following NIMS features. A portion of this energy is:

- 1) Circulated to the non-illuminated side of the spacecraft by the pumped coolant, and re-radiated to cold space.
- 2) Absorbed in the DIPS components and structure, and then removed after threat encounter via the pumped coolant loop and rejected to space.
- 3) Prevented from reaching the payload and housekeeping electronics in significant amounts by using heat pipes, with their inherent thermal lag, to passively couple the electronics to the pumped coolant loop.

In addition, the isotope powered NIMS permits selection of the most favorable spacecraft orientation without regard for considerations such as power generation by solar arrays. This, together with the use of selective optical coatings, minimizes the fraction of the incident energy absorbed by the NIMS.

6.3.2 NIMS THERMAL ANALYSIS

To verify the general concept of using a "cold" fluid leaving the radiator as a heat sink for components, a worst case analysis was conducted using the DSCS IV payload. This payload, derived from well characterized DSCS-III components, is representative of a worst case condition since the total payload dissipation of 770 Watts is high, and the heat flux density of the 40 Watt TWTAs is highest for all payload components evaluated. The 40 Watt TWTAs dissipates 97 Watts in an area approximately 2.9 inches by 2.9 inches. A two bay panel segment with a typical component arrangement as shown on Figure 6.3-3, was analyzed using a 72 node computer math model. Ten nodes were used for the radiator fluid, ten for the header and 20 for the panels. The remaining 32 nodes were located on the components and heat pipes. Component thermal loads, listed on Figure 6.3-3, total 262 Watts or approximately one-third of the total DSCS IV heat load. All components except the 40 Watt TWTAs are mounted to the panel inboard

EXAMPLE EQUIPMENT LIST

ITEM	DESCRIPTION	THERMAL LOAD WATTS	OPERATING TEMPERATURE °F (°C)	
1	COLLECTOR, 40W TWTA #1	97.0	120	(49)
2	POWER SUPPLY & TUBE OF 40W TWTA #1	20.55	122	(50)
3	COLLECTOR 40W TWTA #2	0	97	(36)
4	POWER SUPPLY & TUBE 40W #2	0	100	(38)
5	FREQUENCY GENERATOR	8.61	115	(46)
6	POWER CONTROLLER	30.0	117	(47)
7	COMMAND TLM I/F	5.36	109	(43)
8	REMOTE TLM UNIT	3.3	111	(44)
9	40W TWTA FILTER	3.91	131	(55)
10	40W TWTA DRIVER	1.47	109	(43)
11	10W TWTA FILTER	1.41	131	(47)
12	10W TWTA POWER SUPPLY & TUBE	8.58	124	(51)
13	10W TWTA COLLECTOR	25.5	138	(59)
14	FILTER, 10W TWTA	1.41	122	(50)
15	10W TWTA POWER SUPPLY & TUBE	8.58	126	(52)
16	10W TWTA COLLECTOR	25.5	138	(59)
17	BEACON "B"	20.8	122	(50)
18	10W TWTA POWER SUPPLY & TUBE	0	109	(43)
19	10W TWTA COLLECTOR	0	109	(43)

UPPER TEMPERATURE LIMITS (OPERATION)	
<u>TWT'S</u>	
NORMAL	84°C
QUAL	85°C
<u>OTHER</u>	
NORMAL	50°C
QUAL	71°C

Figure 6.3-3. NIMS Thermal Analysis Model
(DSCS III Payload Components)

surfaces. The 40 Watt TWTAs are located on the outboard corner of the panel in order to take advantage of the lowest header fluid temperature and the film coefficient. The 40 Watt TWTAs are attached directly to the heat transfer header to further reduce thermal resistance to the cooling fluid.

The film coefficient used in the boiling and condensing section of the heat pipes was 600 Btu/Hr $FT^{2\circ F}$. The interface contact conductance between heat pipes and headers, and heat pipes and panels was 400 Btu/hr. $FT^{2\circ F}$. These data are based on measured performance data on the heat pipes used on the BSE and CTS spacecraft. At the radiator header, the cool fluid, Dowtherm A, leaves the radiator in four separate 5/8" O.D. tubes at a temperature of 80°F (27°C). As shown in Reference 1 the straight round tube is not the most efficient design for convective cooling. Therefore an annular arrangement was developed having a flow passage geometry OD = 1.5" and ID = 1.42". For a viscous fluid in an annular configuration the Nusselt No. (N_{nu}) for fully developed flow is shown in Reference 2 to be a function of radius ratio of the fluid passage. As $r_i/r_o \rightarrow 1$, $N_{nu} \rightarrow 5.38$. The radius ratio of the selected design is 0.946. In the inlet region of a flow passage the hydrodynamic and thermal boundary profiles are developing, and the Nusselt numbers exceed the fully developing value. The variation in Nusselt number with l/d , r_i/r_o and N_{pr} in laminar flow, is described in Reference 3. Figure 6.3-4 presents selected data from Reference 3, and shows the variation of N_{nu} with $\bar{x}$, a normalized axial co-ordinate for several values of radius ratio r^* . Figure 6.3.5 presents the variation in N_{nu} with N_{pr} for several values of $\bar{x}$ which shows that for small values of $\bar{x}$ the value of N_{nu} is sensitive to N_{pr} but as $\bar{x}$ increase this sensitivity to N_{pr} decreases. For fully developed flow the controlling variable is r^* .

Applied to the NIMS study the geometry and fluid characteristics yield a local film coefficient of 85 Btu/hr. $FT^{2\circ F}$. Because of the sensitivity of N_{nu} to the several variables a value of 80.0 was used in the header at

Figure 6.3-4. Variation in N_{∞} with $\bar{x}$ for Values of r^*

Figure 6.3-5. Variation in N_{nu} with N_{pr} for Values of $\bar{x}$

the TWTA interface and downstream a fully developed value of 60 Btu/hr.ft² was used.

Five cases were analyzed using these flow parameters and the math model, one steady state, and four transients. For steady state (Case 0), the inlet fluid temperature is constant at 80°F (27°C), and the transient cases are based on the varying fluid inlet temperature profiles shown on Figure 6.3-6. Temperatures for the steady state condition (normal operation) are shown on Figure 6.3-3 and in Table 6.3-1. For the transient condition, maximum temperatures, as shown for each component in Table 6.3-1, did not exceed their qualification temperature limits. A further discussion of the transient conditions is presented in Volume III.

Based on this analysis it is concluded that the NIMS integrated thermal concept can adequately control component temperatures under both steady-state and transient conditions, including high heat density components such as TWTA's. The analysis conducted was for a typical payload component arrangement having a total heat load of 262 Watts on two of the eight equipment panels. With further optimization of component and heat pipe locations, a higher heat load capacity appears feasible.

For preliminary evaluation of alternate payloads as presented in Section 5.2.2, a total equipment panel heat dissipation capacity of $4 \times 262 = 1048$ Watts was used.

- REFERENCES:
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Int. Journal Heat & Mass Transfer
Volume 7, p. 763-781, 1964.

Figure 6.3-6. Inlet Fluid Temperature Variation with Time

TABLE 6.3-1 NIMS THERMAL ANALYSIS - DSCS IV PAYLOAD COMPONENT TEMPERATURE

EXAMPLE EQUIPMENT LIST		THERMAL LOAD WATTS	TEMPERATURES	
ITEM	DESCRIPTION		MAXIMUM TRANSIENT f = 270; A = 5 °F (°C)	NORMAL OPERATION °F (°C)
1	COLLECTOR, 40W TWTA #1	97.0	148 (65) ¹	120 (49)
2	POWER SUPPLY & TUBE OF 40W TWTA #1	20.55	131 (55)	122 (50)
3	COLLECTOR, 40W TWTA #2 NON-OPERATING	0	124 (51)	97 (36)
4	POWER SUPPLY & TUBE 40W TWTA #2 NON OPERATING	0	109 (43)	100 (38)
5	FREQUENCY GENERATOR	8.61	122 (50)	115 (46)
6	POWER CONTROLLER	30.0	122 (50)	117 (47)
7	COMMAND TLM I/F	5.36	115 (46)	109 (43)
8	REMOTE TLM UNIT	3.3	117 (47)	111 (44)
9	40W TWTA FILTER	3.91	136 (58)	131 (55)
10	40W TWTA DRIVER	1.47	117 (47)	109 (43)
11	10W TWTA FILTER	1.41	120 (49)	131 (47)
12	10W TWTA POWER SUPPLY & TUBE	8.58	131 (56)	124 (51)
13	10W TWTA COLLECTOR	25.5	145 (63)	138 (59)
14	FILTER, 10W TWTA	1.41	126 (52)	122 (50)
15	10W TWTA POWER SUPPLY & TUBE	8.58	131 (55)	126 (52)
16	10W TWTA COLLECTOR	25.5	144 (62)	138 (59)
17	BEACON "B"	20.8	127 (53)	122 (50)
18	10W TWTA POWER SUPPLY & TUBE } NON	0	109 (43)	109 (43)
19	10W TWTA COLLECTOR } OPERATING	0	109 (43)	109 (43)

1. HIGHEST TRANSIENT: 16°C

CURRENT DOD SPECIFICATIONS FOR UPPER TEMPERATURE LIMITS

TWT'S:	NORMAL OPERATION	64°C
	OPERATING QUAL	85°C
OTHER:	NORMAL OPERATION	50°C
	OPERATING QUAL	71°C

6.3.3 THERMAL CONTROL SUBSYSTEM COMPONENTS

The NIMS Thermal Control hardware and weights, listed in Table 6.3-2, have been derived from similar applications on existing programs such as DSCS III and BSE.

TABLE 6.3-2 NIMS THERMAL CONTROL COMPONENTS

COMPONENT	WEIGHT (LBS)	REFERENCE
Heat Pipes	30.0	150 ft. @ 0.2 LBS/FT (BSE)
Heaters & Thermostats	5.0	DSCS-III
Insulation	40.0	DSCS-III
Thermal I/F Fittings	25.0	NIMS Thermal Analysis
Miscellaneous	5.0	DSCS III & BSE
TOTAL TCS	105.0	

6.4 PROPULSION

Propulsion subsystems are required for orbit rejection and for spacecraft on-orbit reaction control. For orbit injection from STS, incremental velocity (ΔV) is required at both perigee and apogee. The total ΔV requirement for geosynchronous missions is approximately an order of magnitude greater than that required for low earth missions such as DMSP. The selected NIMS solid pocket and bi-propellant propulsion subsystem meet these varied mission requirements with fully developed and available technology and hardware.

6.4.1 PROPULSION SUBSYSTEMS SELECTION

Combination of solid rocket, bi-propellant and mono-propellant subsystems were evaluated to meet the NIMS multiple mission propulsion requirements and four candidate subsystem combinations are shown on Figure 6.4-1, as configured and sized to meet geosynchronous mission requirements. For the NIMS study a specific impulse 293 seconds has been used for solid propellant, 305 seconds for MMH + NTO bi-propellant, and 230 seconds for hydrozine mono-propellant. These values represent measured performance of these propellants in existing thrusters available for the NIMS applications.

A comparison of these alternates is made in Table 6.4-1, and shows that the use of solid rocket motors for the AKM and PKM functions provides the lowest on orbit weight and a low total weight system. This concept, however has the poorest spacecraft packaging efficiency due to the back-to-back rocket motor arrangement shown on Figure 6.4-1. The mono-propellant system is least weight efficient and also requires excessive tankage volume. The selected system uses an ejectable solid rocket motor for the PKM and Bi-propellants for the AKM function and on orbit reaction control. This combination results in the lowest subsystem weight and superior internal and external packaging efficiency.

Installation of the PKM and Bi-propellant tankage and thrusters is shown on the NIMS Bus General Arrangement drawing, Figure 6.1-1.

The propulsion fuel requirement for achieving final orbit for the alternate NIMS missions is shown in Table 6.4-2. Transfer orbit fuel requirements are for the solid rocket PKM for all missions except DMSP and DSSS, and circularization and inclination removal (AKM) fuel is for the on-board Bi-propellant subsystem. The NIMS is 3-axis stabilized during PKM and AKM burn using the bi-propellant system for thrust vector control. The low earth orbit for DMSP can be efficiently achieved using only Bi-propellants and the DSSS is injected into final orbit by the IUS. In addition to the Bi-propellant fuel used for ΔV functions, additional fuel is required for the on-orbit reaction control (RCS) functions.

	<u>WEIGHT</u> (LBS)
STAR 62 PKM	11,300
STAR 48 AKM	3,520
HYDRAZINE RCS	1,153
<u>TOTAL SYSTEM</u>	<u>15,973</u>
<u>ON-ORBIT DRY</u> <u>WEIGHT</u> -	<u>412</u>

	<u>WEIGHT</u> (LBS)
STAR 62 PKM	11,300
BI-PROPELLANT AKM & RCS	4,532
<u>TOTAL SYSTEM</u>	<u>15,832</u>
<u>ON-ORBIT DRY</u> <u>WEIGHT</u> -	<u>644</u>

BI-PROPELLANT PKM - AKM - RCS	
<u>TOTAL SYSTEM</u> <u>WEIGHT (LBS)</u>	<u>16,179</u>
<u>ON-ORBIT</u> <u>DRY WEIGHT</u> (LBS)	<u>2,071</u>

MONO-PROPELLANT PKM - AKM - RCS	
<u>TOTAL SYSTEM</u> <u>WEIGHT (LBS)</u>	<u>21,452</u>
<u>ON-ORBIT</u> <u>DRY WEIGHT</u> (LBS)	<u>2,720</u>

Figure 6.4-1. Candidate NIMS Propulsion Subsystems
(Geosynchronous Mission Configurations)

TABLE 6.4-1 NIMS CANDIDATE PROPULSION SUBSYSTEMS COMPARISON
(GEOSYNCHRONOUS MISSIONS)

PROPULSION	CONFIGURATION	TOTAL WEIGHT (LBS)	ON-ORBIT DRY WEIGHT (LBS)	ASSESSMENT
(1) ● SOLID PKM ● SOLID AKM ● HYDRAZINE RCS	● EJECTABLE PKM ● FIXED AKM & RCS SUBSYSTEM ● "STACKED" SOLIDS	15,973	412	● LOWEST RETAINED WEIGHT ● "STACKED" BACK-TO-BACK SOLID MOTORS HAVE POOREST PACKAGING EFFICIENCY
(2) ● SOLID PKM ● BI-PROPELLANT AKM & RCS	● EJECTABLE PKM ● INTEGRATED FIXED BI-PROPELLANT SUB-SYSTEM	15,832	644	● LOWEST WEIGHT SYSTEM ● BEST PACKAGING EFFICIENCY <div>SELECTED SYSTEM</div>
(3) ● ALL BI-PROPELLANT (MMH + NTO)	● FIXED INTEGRAL PROPULSION SUBSYSTEM	16,179	2,071	● HIGH RETAINED WEIGHT REDUCES PAYLOAD ● LARGE TANK VOLUME
(4) ● ALL MONO-PROPELLANT	● FIXED INTEGRAL PROPULSION SUB-SYSTEM	21,452	2,720	● HIGHEST WEIGHT SYSTEM ● POOR PERFORMANCE ● LARGEST TANK VOLUME

TABLE 6.4-2 FUEL REQUIRED TO ACHIEVE NIMS FINAL ORBIT

MISSION	REQUIRED ΔV , FPS		FUEL REQUIRED, POUNDS		TOTAL
	TRANSFER	CIRCULARIZATION & INCLINATION REMOVAL	TRANSFER *	CIRCULARIZATION ** & INCLINATION REMOVAL	
DMSP BLOCK 6	1013.6 (0^0)	2220.0 ($\Delta i=5^0$)	462	860	1322
GPS ØIII	6700.5 (0^0)	7660.9 ($\Delta i=35^0$)	7802	3532	11334
GPSCS	7998.2 (1.5^0)	5896.5 ($\Delta i=27^0$)	10657	3141	13798
DSCS IV	7998.2 (1.5^0)	5896.5 ($\Delta i=27^0$)	10657	3141	13798
DSSS	← NOT APPLICABLE - IUS USED FOR TRANSFER/ORBIT INSERTION →				
SSS	9873.2 ($\Delta i=0^0$)	4351.6 ($\Delta i=61.5^0$)	12060	1989	14049

* SOLID ROCKET PKM EXCEPT BI-PROPELLANT FOR DMSP

** BI-PROPELLANT FOR ALL MISSIONS

6.4.2 BI-PROPELLANT SUBSYSTEM

The monomethyl hydrazine (MMH) and nitrogen tetroxide (NTO) bi-propellant subsystem is integrated into the basic NIMS spacecraft and provides propulsion for orbit transfer and/or circularization and for on orbit spacecraft control. Subsystem characteristics are summarized in Table 6.4-3, and a block diagram of a typical bi-propellant subsystem is shown on Figure 6.4-2. Four 100 pound thrust engines are used for high-thrust ΔV functions and are mounted on the spacecraft aft bulkhead. Sixteen two pound thrusters are used for on-orbit RCS control and are mounted to the outer radiator by external truss supports. Both 100 pound and two pound bi-propellant engines and associated valves, sensors and hardware have been fully developed by Marquardt and Rockwell and the Marquardt engines are currently specified for the Indiasat program. Existing 29.5 inch and 33 inch diameter fuel tanks, including positive expulsion devices are available from other programs and are used in combinations for all NIMS Missions. Pressurant tanks are Titanium spheres of 15 to 18 inch diameters made from existing tank forgings. Helium is the selected pressurant providing a pressurant system 27 percent lower in weight than nitrogen. A breakdown of Bi-propellant subsystem fuel and hardware for the NIMS missions is shown in Table 6.4-4 including propellant weights tank sizes and configurations, and total subsystem weights. For NIMS, the pressurant tanks have been sized to a burst factor of 4.0 to meet current STS safety requirements. A burst factor of 2.0 is normally used for design of Pressure vessels for use on conventional boosters and the higher factor almost doubles the pressurant tanks weight. The fuel tanks are designed to their original factor of 2.0 since they are not under full pressure until after the STS is in orbit and the cargo bag doors opened. The burst design criteria is presently under evaluation by the STS agencies and alternate, lighter weight approaches may become available prior to implementation of a hardware development program for an STS compatible system.

The mass fraction is approximately 0.85 for the NIMS design which has an RCS function, and the burst factor is 4.0. For the NIMS Bi-propellant system using a burst factor of 2.0 and eliminating the RCS weight, the mass fraction increases to 0.90, normal for propulsion systems of this category.

TABLE 6.4-3 BI-PROPELLANT PROPULSION SUBSYSTEM

- BI-PROPELLANT SYSTEM PROVIDES
 - ORBIT TRANSFER AND/OR CIRCULARIZATION CAPABILITY
 - ON-ORBIT CONTROL
 - SUPERIOR WEIGHT AND PACKAGING EFFICIENCIES
- SYSTEM ELEMENTS
 - FUEL: MONOMETHYL HYDRAZINE (MMH)
 - OXIDIZER: NITROGEN TETROXIDE (NTO)
 - PRESSURANT: HELIUM
 - THRUSTERS: 100 LB FOR VELOCITY CONTROL
2 LB FOR ATTITUDE CONTROL
 - TITANIUM TANKS FOR FUEL, OXIDIZER AND PRESSURANT
 - WEIGHT:

DMSP	-365 POUNDS	DSSS - 191
GPS/DSCS		
IV/GPSCS	-644 POUNDS	SSS - 511
- HARDWARE STATUS
 - FLOWN ON SYMPHONIE AND VOYAGER SPACECRAFT
 - TO BE FLOWN ON INSAT AND LEASESAT
 - ALL THRUSTERS, TANKS AND OTHER COMPONENTS FULLY DEVELOPED AND QUALIFIED FOR 10 YEAR MISSIONS

Figure 6.4-2. Bi-Prope Nit Propulsion Subsystem Schematic

TABLE 6.4-4 NIMS BI-PROPELLANT PROPULSION
SUBSYSTEM BREAKDOWN

ITEM		MISSION				
		GPS	DSCS IV GPSCS	DMSP	S3	DSSS
Δ V Propellant - LBS		3532	3141	1322	1989	0
Orbit Propellant - LBS		100	747	200	758	900
Total Propellant -LBS		3632	3888	1522	2747	900
Tank Configuration	Fuel**	2-29 1/2"D+4-33"D		4-29 1/2"D	6-29 1/2"D	2-29 1/2"D
	Pressurant	4 - 18"D		4-15"D	4-17"D	2-15"D
Capacity - LBS		4004	4004	2136	3204	1068
Fuel Margin-LBS		372	116	614	457	168
*Press Tank Wt. LBS		316.0		168.0	252.0	84.0
Helium WT-LBS		22.0		12.0	18.0	6.0
pped Propellant		34.0		18.0	27.0	9.0
Δ V Thrusters WT LBS (4-100 LB Thrust)		28.0		28.0	28.0	N/R
ACS Thrusters WT LBS (16-2 LB Thrust)		25.0		25.0	25.0	25.0
S/S Hardware & WT Plumbing		26.0		24.0	26.0	22.0
Total Dry Weight LBS		644.0		365.0	511.0	191.0
Total System (Full Capacity)		4648		2501	3715	1259
Mass Fraction		.861		.854	.862	.848

* PRESSURANT TANKS SIZED FOR BURST FACTOR OF 4.0

6.4.3 PERIGEE KICK MOTOR (PKM)

The PKM selected for NIMS is the Thiokol 62 inch diameter Star 62, shown on Table 6.4-5. This motor is currently under development by Thiokol and was successfully fired at sea level in 1978. It has a high performance grain design using HTPB propellant, silica-filled DPDM insulation, titanium case and carbon-carbon nozzle. The motor can be configured in lengths from 72 to 102 inches covering a propellant range of 5,400 to 12,000 pounds by increasing the length of the central cylindrical section joining the case end domes. This "stretching" technique has been fully developed and proven on the smaller Star 37 rocket motor up to double the original propellant loading for that motor.

NIMS-Star 62 PKM size and weights are shown in Table 6.4-6 for those missions requiring a solid PKM. The PKM adapter weight of 100 pounds included the aluminum adapter ring, spacecraft/PKM separation vee-band and separation springs and guides. Spacecraft attached separation rails and fittings weights are included in the spacecraft structural weight. The 100 pound adapter is rigidly attached to the PKM case and is ejected after burnout with PKM.

6.5 ATTITUDE CONTROL SUBSYSTEM (ACS)

Because of the elimination of the large solar arrays, many of the constraints normally imposed on the ACS can be relaxed significantly resulting in better performance. Some of the more important of these are:

1. Eliminates solar array and flexible dynamics limit to control bandwidths allowing better performance.

TABLE 6.4-5 THIOKOL STAR 62 PERIGEE KICK MOTOR (PKM)

- PROVIDES PERIGEE ΔV FROM STANDARD STS ORBIT
- PKM EJECTED PRIOR TO APOGEE BURN

STAR 62 CHARACTERISTICS

- PROPELLANT WEIGHT RANGE-5,400 TO 12,000 POUNDS
- TOTAL LENGTH-72 INCHES TO 102 INCHES
- MASS FRACTION-.943 TO .953
- ISP, SECONDS-294 TO 284

62 INCH DIAMETER STAR 62 SOLID ROCKET MOTOR

- FULLY DEVELOPED SOLID PROPELLANT, TITANIUM CASE AND CARBON - CARBON NOZZLE TECHNOLOGY
- INCREASES IN MOTOR LENGTH UP TO DOUBLE PROPELLANT LOADINGS FULLY DEMONSTRATED DURING 10 YEAR STAR 37 PROGRAM

TABLE 6.4-6 NIMS PKM CHARACTERISTICS
SUBSYSTEM BREAKDOWN

MISSION	CASE DIA (IN.)	TOTAL LENGTH (IN.)	BURN-OUT WEIGHT (LBS.)	PROPELLANT WEIGHT (LBS.)	TOTAL PKMWT (LBS.)	PKM ADAPTER WT. (LBS.)
GPS-Ø III	62	92	543	7802	8345	100.0
DSCS IV CPSCS	62	92	543	10657	11200	100.0
NFCS (SSS)	62	102	600	12060	12660	100.0

2. Reduces solar pressure disturbance and resulting torques by up to a factor of 10.
3. Reduces major source of thruster plume impingement (solar array) allowing better ΔV mode performance.
4. Eliminates the solar array as a sensor blockage/reflection source and sun orientation attitude constraint.

However, the DIPS does have an uncompensated momentum of about 17 ft-lb-sec along the alternator axis which must be accounted for in any system design.

6.5.1 ACS REQUIREMENTS

The ACS requirements for each of the six missions considered are shown in Table 6.5-1.

As shown, on-orbit pointing accuracies of 0.08° in pitch and roll and 0.8° in yaw will satisfy four of the six missions considered. The DMSP and DSSS missions require almost an order-of-magnitude better pointing accuracy. Additionally, these two missions have a requirement that attitude rates be kept very low. Rate control is not required for the other four missions.

6.5.2 ACS CONCEPTS EVALUATION AND SELECTION

To provide the required attitude control in all three axes in the presence of the DIPS' uncompensated momentum of 17.6 ft-lb-sec, three approaches were considered:

1. Compensate with counter rotating flywheel.
2. Place DIPS combined rotation unit (CRU) along yaw axis and counter rotate vehicle (yaw rotator).

TABLE 6.5-1 ACS MISSION REQUIREMENTS

ITEM	GPS ØIII	DSCS III/IV	NFSC S ³	GPSCS	DSSS	DMSP
ON-ORBIT POINTING ACCURACY						
PITCH (36)	0.1 ⁰	0.08 ⁰	0.08 ⁰	0.08 ⁰	0.01 ⁰	0.01 ⁰
ROLL (36)	0.1 ⁰	0.08 ⁰	0.08 ⁰	0.08 ⁰	0.01 ⁰	0.01 ⁰
YAW (36)	N/A	0.8 ⁰	0.8 ⁰	0.08 ⁰	0.01 ⁰	0.01 ⁰
ORBIT						
ATTITUDE (NM)	10,900	19,323	110,375	19,323	19.323	450
INCLINATION (DEG)	63	0	90	0	0	99
SPECIAL	ΔV DISTURBANCES MINIMIZED	-	ORBIT MANEUVERS		JITTER <0.005 ⁰ /SEC ATTITUDE MANEUVERS LARGE INERTIAL DISTURBANCES	JITTER <0.002 ⁰ /SEC YAW MANEUVERS
LIFE	10 YRS	10 YRS	10 YRS	10 YRS	10 YRS	2-5 YRS
PKM/AKM CONTROL	1.0 ⁰	1.0 ⁰	1.0 ⁰	1.0 ⁰	N/A	N/A

3. Place rotating unit along pitch axis and use as bias momentum source.

6.5.2.1 Counter Rotating Flywheel Compensation

This approach requires the addition of a high speed flywheel counter rotating to the DIPS (Combined Rotating Unit), CRU, to cancel out the momentum of the CRU. When the CRU momentum is cancelled, a means of stabilizing the vehicle in all three axes must be added. This could be accomplished by using a reaction wheel in each axis or through a four-skewed wheel system which provides long life redundancy because of the fourth wheel (3 out of 4 required for control).

To insure long life, the counter-rotating flywheel should also be backed up with a redundant flywheel. The use of two such wheels would impose a weight penalty to the system.

6.5.2.2 Yaw Rotator

An alternative method of cancelling the DIPS momentum is to counter rotate the spacecraft to the CRU's direction. In this approach, the CRU would be placed along the yaw axis (axis which points to the Earth's center) and the vehicle rotated in yaw at the rate required to cancel out the system momentum. As in the first case, a reaction wheel system would be required for pitch and roll control.

The primary disadvantage to this concept is that the vehicle rotates about the yaw axis making this approach incompatible with line scan sensors or with narrow beam linearly polar antenna systems which must be pointed to and held on a selected area. The yaw rotator is acceptable for systems such as GPS, with an earth coverage circularly-polarized antenna system, or to DSSS in the scan mode.

6.5.2.3 Use Of DIPS As The Stabilization Source

This concept makes use of the CRU's momentum as the vehicle's basic stabilizing element. The spin axis of the CRU would be placed on the

pitch axis of the spacecraft and its momentum would be used in a bias momentum form of stabilization. Used in this manner, the CRU's momentum provides control of both the spacecraft's roll and yaw axes. The vehicle body would be "despun" and fine pointing control and adjustment of the body would be accomplished through the use of small reaction wheels (redundant) in the pitch axis. Control of roll and yaw axes would be done by redundant reaction wheels mounted in the yaw axis.

This approach is the lightest weight approach to meet the 0.08° (pitch/roll) and 0.8° (yaw) pointing requirement of four of the six missions considered. For the other two missions with the tighter pointing requirement, the addition of star sensors and inertial reference units along with a reconfiguration of the four reaction wheels used will be required.

6.5.3 BASELINE ACS

The baseline ACS is shown in Figure 6.5-1 in block diagram form. Basically, the DIPS momentum is used to provide the pitch bias momentum. Pitch, roll and yaw axis control are provided by smaller momentum wheels mounted along the pitch and yaw axes. Under normal on-orbit operations, the earth sensor provides the necessary pitch and roll error signals. The sun sensors are used whenever three axis information is required (i.e., north/south station keeping). Because of the gyroscopic action, yaw torques cause roll axis precession and control. Yaw errors are coupled into the roll axis twice an orbit and are corrected then.

The DIPS is operating during launch. However, the NIMS is 3-axis stabilized during PKM and AKM burn using the Bi-propellant system for thrust vector control.

6.5.4 DMSP VARIATION

The DMSP variation uses the DIPS momentum as in the baseline approach, but the smaller wheels are now mounted in a "skewed wheel" configuration. All four wheels contribute to pitch control, with two of the four contributing to yaw control and the other two contributing to roll control. The earth sensor is replaced with an Inertial Measurement Unit (IMU) and star sensors for three axis attitude determination. Within the ACS processor

Figure 6.5-1. Baseline Attitude Control Subsystem

Figure 6.5-2. DMSP ACS Variation

electronics is both an updateable star catalog and orbit ephemeris data. This equipment, along with minor software changes, represent the changes necessary to accomplish the finer pointing. This is shown in block diagram in Figure 6.5-2.

6.5.5 DSSS VARIATION

Because of a high slew rate requirement coupled with the mass of this configuration, a reaction wheel system is not capable of providing the required control. A Control Moment Gyro (CMG) system would be used instead of the reaction wheel system. A preliminary sizing indicates that six 15 ft-lb-sec CMG's are required. These units would be mounted in a module on the top surface of the spacecraft. The total weight of the CMG system is about 500 pounds.

6.6 TELEMETRY, TRACKING AND COMMAND (TT&C)

The TT&C subsystem considered for these missions is compatible with the Air Force's Space Ground Link System (SGLS). The system operates primarily in the S-band frequency band (1850 MHz uplink and 1750 MHz downlink).

This unique dual-frequency TT&C subsystem, shown in block diagram form in Figure 6.6-1, was selected to meet the varied mission telemetry and command interface requirements. It allows incorporation of both S-band and SHF COMSEC equipment and permits operation in various jamming environments. It is a hardened, long-life system based on the DSCS III TT&C subsystem.

The S-band part of this subsystem is primarily for spacecraft control, tracking and housekeeping functions through its links with the Remote Tracking Stations (RTS) of the Air Force Satellite Control Facility. It uses a dedicated S-band antenna. The primary function of the SHF link is for payload configuration and control. The SHF link uses earth coverage horns for receive and transmit. Although separate functions can be assigned to the two frequency channels during normal operation, their functions are interchangeable to provide full redundancy whenever con-

Figure 6.6-1. TT&C Block Diagram

TABLE 6.6-1

TELEMETRY CAPACITY (TYPICAL)

SIGNAL TYPE	QUANTITY
BI-LEVEL	490
ANALOG	448
SERIAL-DIGITAL (VARIABLE LENGTH OF UP TO 273 EIGHT BIT WORDS)	18

TABLE 6.6-2

COMMAND CAPACITY (TYPICAL)

TYPE	QUANTITY
DISCRETE MAGNITUDE (UP TO 14 DATA BITS EACH)	672 5
MESSAGE (18 TO 1152 BITS EACH)	24

ditions may warrant. Typical telemetry and command capabilities are shown in Tables 6.6-1 and 6.6-2, respectively.

6.7 ELECTRICAL POWER AND DISTRIBUTION (EP&D)

The EP&D subsystem consists of the DIPS, load control units (LCU's) for payload/subsystem operation, and the electrical harness.

6.7.1 DIPS

The DIPS shown schematically in Figure 6.7-1 provides all of the electrical power required for the spacecraft subsystems and mission-unique equipments/payloads. This power is provided on a full-time basis starting at launch, thus eliminating the need for any electrical storage such as is required for solar array systems. The operation of the DIPS has been described earlier (Section 1.0, Introduction). Table 6.7-1, summarizes the characteristics of this power system. Because of the active cooling loop and radiator system, the DIPS is capable of stable operation with an unbalanced solar input and is resistant to both natural and induced radiation.

TABLE 6.7-1 DIPS CHARACTERISTICS

ITEM	TYPICAL CHARACTERISTIC
Rated output power, Watts(e)	1200 (end of 10 years)
Optional Ratings, Watts(e)	500 to 2000
Input power @ 1300W(e),BOM,Watts(t)	7200
Peak Working Fluid Temperature, Degrees F°	675
Overall thermal efficiency, %	18.4 (rectified to 28VDC)
Weight, Pounds (maximum)	450
Design point output voltages, VDC	28 $\pm$ 2%
Lifetime, years	10

Figure 6.7-1 DIPS Schematic

A top level weight comparison of the DIPS with an equivalent solar array/battery power subsystem for the same end-of-life output power of 1200 Watts full-time is shown in Table 6.7-2. As shown, the DIPS is about 114 pounds lighter than an unhardened solar array/battery system for the DMSP mission and approximately equal for the high altitude missions.

6.7.2 LOAD CONTROL UNIT (LCU)

The LCU is basically a power switching module used to control power to all components. By switching power on and off, function and redundancy control is provided. Two units would be used; one for control of all housekeeping functions, and the other for mission-unique payload control.

6.7.3 ELECTRICAL HARNESS

The Electrical Harness is used to distribute power, route the decoded commands, and transfer the components' telemetry signals to the required units in the spacecraft. The unique feature of the type of harness to be used for these missions is the protection against both EMP and SGEMP it provides. The harness design approach is basically the same as that used on the DSCS III spacecraft.

6.7.4 DIPS RELIABILITY

The reliability analysis performed by Sundstrand Energy Systems* has been reviewed. Initial analysis estimated the DIPS reliability for a 10-year mission at 0.9119, as shown on Table 6.7-3. It was further determined that minor increases in radiator armor and the addition of some redundancy to the DIPS electronics controller would increase the DIPS 10-year reliability estimate to 0.9304.

The DIPS reliability of 0.9304 is less than the about 0.97 reliability estimates current for solar array/battery power systems. However, the overall NIMS Bus reliability, estimated at 0.68, is comparable to the 10-year reliability estimates for a SAB powered bus (no payload).

* Personal communication to John Nonnemann, Sundstrand Energy Systems, January 1979

TABLE 6 -2 POWER SYSTEM WEIGHT COMPARISON

DIPS		UNHARDENED SOLAR ARRAY/BATTERY SYSTEM	
ITEM	WT, LBS	ITEM	WT, LBS
POWER CONVERSION SYSTEM	167	SOLAR ARRAYS	111.6 (133.9)*
HEAT SOURCE ASSEMBLY	185	MECHANISMS AND STRUCTURAL SUPPORT	51.9
RADIATOR	87		
RADIATOR COATING	11	POWER REGULATION/CONTROL	46.0
		BATTERIES (SYNCH.; 50% DOD)	237.2 (337.1)*
TOTALS	450		446.7 (563.9)*

* SOLAR ARRAY & BATTERY WEIGHTS FOR DMSP (BATTERY DOD = 35%)

TABLE 6.7-3 DIPS RELIABILITY ASSESSMENT

COMPONENT	10 Year Mission Reliability Estimates	
	BASELINE DIPS	AS MODIFIED FOR NIMS*
System Accumulator and Filter Assembly	0.9990	
Flow Control Valve	0.9910	
Regenerator	0.9980	
Combined Rotating Unit	0.9801	
Jet Condenser	0.9920	
Radiator Bypass Valve and Filter Assembly	0.9990	
Radiator	0.9801	0.9857
Heat Source Assemblies	0.9978	
Electronic Controls	0.9740	0.9857
Loop Containment Integrity	0.9976	
System	0.9119	0.9304

*Only modified values are shown

6.8 NIMS - STS INTEGRATION

The NIMS spacecraft is configured for launch by STS in either a vertical orientation using an ASE launch cradle, or horizontally attached to the IUS. For IUS launch, no separate cradle or other support equipment is required, and the spacecraft is cantilevered from a launch adapter attached to the IUS.

The vertical launch cradle concept shown on Figure 6.8-1, is based on the design of Standard Test Rack structures for Shuttle designed by General Electric. In this design two aluminum test racks are used for the end frames and are joined by graphite-epoxy cross structure members.

The spacecraft attach fitting is a 64 inch diameter cylinder containing the separation Vee-Band, separation springs and S/C to cradle in-flight disconnects. A "short" attach fitting is used for launch of NIMS configurations having a PKM and a longer fitting replacing the PKM adapter is used for launch of low altitude mission configuration not requiring a PKM. Preliminary weight estimates of the NIMS Launch Cradles for both launch configurations are shown in Table 6.8-1 and the total ASE weights plus a 500-pound launch contingency have been included in the mission weight breakdowns presented in Paragraph 5.2.5.

Launch site operation are illustrated on Figure 6.8-2, which shows the spacecraft as assembled and shipped from the spacecraft contractor. The ASE Cradle, PKM, and Isotope Heat Source Assemblies are shipped separately and mated to the spacecraft at the launch site. The spacecraft assembly is given post assembly check-out tests using launch-site installed Electrical ASE prior to Heat Source installation, and prior to final installation in the STS Cargo Bay for launch. Electrical in-flight disconnects are provided on the cradle to permit checkout of critical spacecraft functions prior to separation from the STS.

6.9 NIMS ASSEMBLY, TEST AND LAUNCH FLOW

The modular NIMS spacecraft has been designed to permit parallel and independent assembly and test of the payload, DIPS and spacecraft subsystems. This concept minimizes the time and cost effects caused by interdependent

Figure 6.8-1. NIMS - STS Vertical Launch ASE Cradle Concept

IHS

*PKM NOT REQUIRED FOR LOW ALTITUDE MISSIONS

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Figure 6.8-2. NIMS Launch Site Operations

TABLE 6.8-1 NIMS LAUNCH CRADLE (ASE)
PRELIMINARY WEIGHT ESTIMATES

CRADLE	CRADLE FOR PKM ATTACH WT. LBS.	CRADLE FOR S/C ATTACH (NO PKM) WT. LBS.
END FRAMES	1500	1500
CROSS STRUCTURE	700	700
STS FITTINGS (5)	500	500
ATTACH FITTING & SEPARATION	100	200
TOTAL	2800	2900

series assembly and tests at the subassembly level. The spacecraft contractor is responsible for overall spacecraft integration and assembly interfaces, conducts the final acceptance testing of the all-up spacecraft assembly, and is responsible for final installations and integration at the launch site. NIMS parallel assembly and test flow is shown on Figure 6.9-1.

The spacecraft contractor, responsible for final spacecraft assembly, fabricates structural sub assemblies or fixtures using close tolerance master tooling to control all assembly interfaces. The payload sensor support structure and payload equipment panels (with heat pipes attached) are shipped to the Payload Contractor for installation and compatibility testing of the payload, and a MAGE fixture is sent to the DIPS Contractor for assembly and test of the DIPS including the external radiator. The DIPS contractor provides and installs Electric Heat Sources (EHS's) simulating the Isotope Heat Sources Assembly (HSA), which are used for all ground testing prior to installation of the flight IHSA at the launch site. The spacecraft Contractor (or propulsion sub-contractor) installs and leak tests the all welded Propulsion Subsystem on the core structure, and the Spacecraft Contractor assembles and tests the housekeeping subsystem panels.

At the completion of all subsystem, DIPS and payload testing, the units are shipped to the Spacecraft Contractor for spacecraft final assembly. The DIPS is first mated to the core structure/propulsion assembly followed by installation of the interconnecting electrical harness and internal insulation. The payload and housekeeping equipment panels are attached and connected next and the payload and housekeeping sensors are mounted and aligned. Installation of the external insulation blankets and (if required), the inert PKM complete the spacecraft assembly. Note that the PKM, the launch ASE and the Isotope Heat Sources are also fabricated and tested in parallel with the spacecraft fabrication and assembly cycle.

At the conclusion of spacecraft final assembly, system level vibration, thermal vacuum and acoustic acceptance tests are conducted at their conclusion, the spacecraft, PKM, IHS, and ASE are all shipped to the launch

site for assembly and check-out prior to installation and launch by the STS.

At the launch site the loaded PKM (if required) and flight IHS are installed and the DIPS is activated and functionally tested. The bi-propellants are then loaded and the spacecraft mated to the launch ASE cradle. The spacecraft/cradle assembly is then installed in the STS cargo bay for launch, completing the NIMS pre-launch assembly and test cycle.

Safety considerations require that auxiliary thermal control be provided for the DIPS isotope heat sources from the time that they are installed into the NIMS until the Shuttle cargo bay doors are opened to deploy the NIMS in the STS parking orbit. (O₂ until earth return in the event of a mission abort).

Hardware development is underway to adapt the Orbiter Active Thermal Control System (ATCS) to provide auxiliary cooling for the Radioisotope Thermoelectric Generator (RTG) powered spacecraft being developed by NASA/JPL. The Galileo mission spacecraft, targeted for STS launch in 1983, utilizes three MHW-RTG units which result in a total thermal load to the Orbiter ATCS of 7200 Watts, the same as that required for the baseline NIMS investigated in this study. The Orbiter ACTS has sufficient capacity to handle this thermal load in flight with the exception of an eleven minute period commencing at liftoff during which high heat loads from the Orbiter equipment reduce the ATCS capability to handle this 7200 Watts(t) load. Workarounds to this solution are available and are being investigated as a part of the Galileo Program*. It has also been determined that the Orbiter ATCS has the ability to handle identified worst-case launch abort/earth return situations for the RTG powered Galileo mission. This capability would also be expected to apply to the NIMS because the thermal loads are the same (7200 Watts). Further work will be required to verify these projections for the NIMS.

The DIPS, as integrated with the NIMS, is assumed to be operating during launch. However, the NIMS radiator will not be rejecting heat during

* Minutes of the Third Galileo/STS Thermal Interface Working Group Meeting;

launch while still contained in the Orbiter, with the cargo bay doors closed. For an STS launch, cooling of the working fluid is accomplished by means of a simple double pipe heat exchanger located in series with, and at the entrance to the NIMS radiator, as shown schematically on Figure 6.9-2. The Orbiter ATCS is connected to this heat exchanger*. The NIMS TCS functions in the same manner as it does on-orbit. Prior to closing the Orbiter cargo-bay doors, a special ASE unit will be required to provide cooling, for example, before the Orbiter ATCS is attached. The feasibility of this auxiliary cooling concept has been established for several years. The design details for this ASE, and items such as valving, redundancy requirements and time-lines remain to be completed.

Auxiliary cooling procedures are also established for the time period between IHS installation and DIPS startup. The DIPS heat exchangers surrounding the IHS in the heat source assemblies are equipped with two heat transfer coils, one for the DIPS working fluid, and one for auxiliary cooling when the DIPS is fueled, but not operating. This is also illustrated schematically on Figure 6.9-2. The ASE auxiliary cooling unit is connected to these auxiliary cooling coils when the DIPS is not operating, cooling the IHS units. When the DIPS is started up on the ground, e.g., for checkout, the ASE auxiliary cooling is switched from the heat source assemblies to the double pipe heat exchanger located at the entrance to the NIMS radiator. The Orbiter ATCS may also be switched to provide direct cooling of the IHS units, through appropriate valving.

It is again noted that similar isotope power system auxiliary cooling concepts, for similar heat loads, are being developed. These are scheduled to be flown on the STS for the isotope powered NASA Galileo mission. Application of these techniques to the NIMS is considered to be a direct, straight-forward technology transfer.

* This heat exchanger weight is included in the DIPS radiator weight estimate.

Figure 6.9-1. NIMS Parallel Assembly and Test Flow

Figure 6.9-2. DIPS Auxiliary Cooling Schematic

SECTION 7

NIMS ATTRIBUTES ASSESSMENT

The overall objective of this study is to identify the attributes of the NIMS bus relative to a Solar Array/Battery (SAB) powered bus. These are to be quantized where possible. The results of this assessment are summarized in Table 7-1, and discussed in the following paragraphs. The enhanced survivability provided by the NIMS is the outstanding attribute identified in this study.

7.1 SYSTEM SURVIVABILITY

This subject is addressed completely in Volume III, NIMS Survivability Assessment, of this report. In summary, the results show that not only will the DIPS enhance power system survivability from the imposed threats, but the capability of the spacecraft and its electronics to survive in the threat environment are greatly enhanced. This is due to the integration of the spacecraft thermal control system with the pumped working fluid loop of the DIPS Organic Rankine System.

The costs associated with the enhanced NIMS survivability, relative to a SAB Bus, are presented in Section 8 of this volume. In summary, the NIMS Bus development costs are \$80.3 million through the qualification unit (1979 dollars), including an estimated \$16.2 million for the development of the DIPS. Assuming that the DIPS is developed separately the development cost of the two buses is approximately equal. The SAB Bus which does not provide enhanced survivability against the laser threat. The unit cost for the NIMS is \$14.7 million, and \$1.5 million greater than that for a SAB Bus. An estimated cost of \$3.5 million must be added for the isotope fuel cost, for a total increment of \$5 million. The costs necessary to provide the same degree of survivability enhancement for a SAB Bus are unknown.

TABLE 7-1. SUMMARY - NIMS ATTRIBUTES ASSESSMENT

	MISSIONS BEING EVALUATED					
	GPS-0111	DMSP/B6	DSCS IV	GPSCS	SSS	DSSS
ENHANCED SURVIVABILITY	● APPLICABLE TO ALL MISSIONS - NIMS HAS SIGNIFICANT ADVANTAGE OVER SOLAR/ARRAY BATTERY SYSTEMS					
SIMPLIFIED ON-ORBIT OPERATIONS	● APPLICABLE TO ALL MISSIONS - SPACECRAFT ORIENTATION INDEPENDENT OF POWER OR THERMAL SUBSYSTEM REQUIREMENTS. - NO ARRAY INTERFERENCE WITH SENSORS AND ANTENNAS. - MAXIMUM MISSION FLEXIBILITY; NO ORIENTATION/MANUEVER LIMITS. - ELIMINATES DEPLOYMENT AND SOLAR ARRAY DRIVE MECHANISMS. - MINIMIZES SOLAR PRESSURE DISTURBANCES ON VEHICLE. - ELIMINATES YAW STEERING FOR POWER COLLECTOR FOR GPS/S ³ TYPE MISSIONS. - ELIMINATES FLEXIBLE BODY/ACS INTERACTIONS. - LAUNCH WINDOW NOT CONSTRAINED BY SUN ANGLE LIMITS FOR POWER OR THERMAL CONTROL. - SAME DESIGN FLIES IN ANY ORBIT WITHOUT MODIFICATION.					
INCREASED ORBITAL ACCURACY	YES	YES	N/A	N/A	N/A	N/A
INCREASED POWER TO USER	● APPLICABLE TO ALL MISSIONS- FULL TIME POWER AVAILABLE; ON-PAD THROUGH END-OF-MISSION. - NIMS DEGRADATION OVER 10 YEARS @ 8% IS LESS THAN 24% FOR SOLAR ARRAY. - WEIGHT ADVANTAGE IS NEGLIGIBLE FOR EQUAL EOM POWER*.					
INCREASED MEAN MISSION DURATION	● REDUCED DEGRADATION RATE RELATIVE TO SOLAR ARRAY/BATTERY SYSTEMS IS ATTRIBUTE FOR LONGER MISSIONS ● EQUAL RELIABILITIES ACHIEVABLE.					

*EXCEPT FOR DMSP MISSION WHERE HARDENED DIPS IS ABOUT 114 POUNDS LIGHTER.

The NIMS Bus will perform all of the six DOD missions investigated. Therefore, its use for more than one mission will substantially reduce its effective development cost per mission. For example, the NIMS effective development costs become \$40 million if used for two missions; \$13.3 million if used for six missions. This includes the laser survivability enhancement capability. These values are substantially less than the \$65.8 million for the SAB Bus which does not provide enhanced survivability.

It may be possible to develop a single SAB Bus to serve the three communications missions and the GPS mission. However, a separate SAB Bus development will be required for each of the DMSP and the DSSS missions. The total SAB Bus development costs to perform all six missions is then about \$197 million (3 X \$65.8 million). And no laser survivability enhancement is provided.

The development of a single NIMS Bus will serve all six missions at a nonrecurring cost of \$80 million, including DIPS. Laser survivability enhancement is provided.

7.2 REDUCTION IN MISSION CONSTRAINTS

The use of DIPS removes many of the constraints normally imposed on solar array-powered spacecraft, as summarized on Table 7.2-1. Many of these are due to the elimination of the large "flimsy" solar arrays while other constraints relaxed or removed are because there is no need to orient the vehicle for power collection or for thermal reasons.

For the GPS and DMSP missions, where orbital predictability is necessary, the more-spherical, more uniform external shape of NIMS will allow for better orbital prediction capability, as previously described and quantified*.

7.2.1 SPACECRAFT ORIENTATION CONSTRAINTS

Depending on the mission considered orientation, constraints on a conventional

* Nuclear-Powered GPS Spacecraft Design Study, Fairchild Space and Electronics Company, ERDA NRA 3063 10 May 1977.

TABLE 7.2-1 REDUCED MISSION CONSTRAINTS

1. Spacecraft Orientation Independent of Power or Thermal Subsystem Requirements.
2. No Array Interference with Sensors and Antennas.
3. Maximum Mission Flexibility; No Orientation/Maneuver Limits.
4. Eliminates Deployment and Solar Array Drive Mechanisms.
5. Minimizes Solar Pressure Disturbances on Vehicle.
6. Eliminates Yaw Steering for Power Collection for GPS/S³ Type Missions.
7. Eliminates Flexible Body/ACS Interactions.
8. Increases Orbital Accuracy Prediction Capability.
9. Can be Flown in any Long Life Orbit.

solar-powered spacecraft are placed by not only the mission payloads, but also the need for power collection and thermal dissipation. These constraints impact all mission phases from launch into the transfer orbit to the final operational orbit.

The launch window, or launch opportunity time, per day is reduced as a function of power collection requirements, thermal constraints, and ACS sensor view requirements. Using DIPS eliminates the power collection constraint and, depending on the payload complement, greatly reduces or eliminates the thermal constraint on-launch and transfer orbit orientation. This has the positive effect of increasing the available time for launching each day.

In the final operational mode, constraints imposed by power collection requirements is again eliminated. Depending on the payload requirements, the orientation constraints due to thermal protection requirements can be eliminated or reduced significantly.

7.2.2 ELIMINATION OF ARRAY INTERFERENCE

In a solar-powered spacecraft system, extreme care must be taken to insure that the solar arrays, when deployed, are out of the fields-of-view of all payload sensors and antenna systems. In addition to this requirement, all possible paths for reflection of light from/to the solar arrays to/from the vehicle and then into the sensor must be determined to insure against "building in" a stray path for interference. To eliminate stray paths, sensors may have to be awkwardly located or solar arrays may have to be moved from the spacecraft's orbital center of gravity (C.G.)

Conversely, the solar array/payload locations must also be evaluated to insure that the payloads do not block the solar arrays under any or all expected sun angles, otherwise, the array power output could be reduced significantly.

The use of DIPS eliminates these array interference considerations.

7.2.3 MAXIMUM MISSION FLEXIBILITY

Because of the elimination of the large, flexible solar arrays and their

impact on the configuration design and layout, the mission spacecraft can be optimized with respect to the mission payload. This will improve or enhance mission performance and operation significantly. As discussed above, the elimination of the solar arrays eliminates any orientation constraints which would otherwise be imposed for optimum power collection. Thermal constraints are likewise eliminated or greatly reduced because of the integrated thermal control system approach.

For threat minimization, spacecraft agility, or the ability to move/maneuver rapidly, is becoming of more interest and importance. The elimination of the large solar arrays permits moving the vehicle more rapidly without harm to any spacecraft element.

7.2.4 ELIMINATION OF DEPLOYMENT AND DRIVE MECHANISMS

Even though solar array release and deployment systems and solar array drive mechanisms have been used successfully on many space missions (i.e. GE's Nimbus, Landsat, & BSE and other missions) elimination of these devices eliminates a potential source of early mission failure. In addition, much design and test validation time and money can be saved if such devices/systems are not required.

7.2.5 REDUCTION OF SOLAR PRESSURE DISTURBANCES

One of the major disturbance forces which act on a spacecraft is solar pressure. Given a large solar array which may be offset from spacecraft's orbital C.G., the resultant force from the center-of-pressure (CP)/C.G. offset can drive the ACS design in the momentum storage and/or torque requirements areas. Elimination of these large solar arrays will dramatically reduce the solar pressure disturbance. This, in turn, will result in superior attitude control performance.

7.2.6 ELIMINATION OF YAW STEERING REQUIREMENT

Two of the six mission spacecraft considered (GPS&S³) are in orbits which, over a yearly cycle, will have the sun "walk" around them. This

is shown in Figure 7.2-1, where β is the angle between the sun and the orbit plane. Since S³ (or NFCS) is in a polar orbit (90° inclination), the β angle varies from 0° to 113.5° (90° + 23.5° variation of the equatorial/ecliptic planes as a function of time). GPS is in a nominal 64° inclination and the β angle can vary from 0° to 87.5° (64° + 23.5°). Because of this β angle variation and the earth-pointing requirement, both of these satellite systems yaw their solar arrays to the sun on a continual basis for power collection. This, of course, can and does add another degree of complexity to the system. With the use of DIPS, this requirement is eliminated, greatly simplifying the spacecraft and its performance, as well as simplifying overall mission operations.

7.2.7 ELIMINATION OF FLEXIBLE BODY/ACS INTERACTIONS

Because of typical weight constraints and other forcing functions such as minimizing array drive motor/train power, size, and weight, solar arrays are extremely flexible appendages. Their flexibility can be a problem to the ACS in that any motion of these arrays will feed back into the main body of the spacecraft causing it to move. This motion would be detected by the ACS which would try to react to it. This correction could feed back to the arrays, thus causing the system to go into oscillations. The design of the ACS must, of course, take this into account and prevent this unstable condition from occurring. Obviously, with DIPS, this problem is eliminated.

7.2.8 INCREASED ORBITAL ACCURACY PREDEICTION

GPS requires very precise knowledge of its orbit to meet the overall user position location accuracy enough which cause orbit perturbation will reduce this accuracy. Unknown variations in solar pressure caused by spacecraft pointing errors represent one such error source. The basis for this is the non-symmetry of the solar array-power spacecraft. Use of the DIPS permit design of a spacecraft which would be nearly symmetrical under all aspect angles, thus reducing or eliminating this problem source.

Figure 7.2-1. β Angle Variation for GPS and NFCS

The above applies to the DMSP mission in that it is required to be able to accurately predict the vehicle's ephemeris over a day's period. Reducing the cross-section of the spacecraft will simplify and improve the prediction capability.

7.3 VALUE OF MISSION CONSTRAINTS REDUCTION

The value of reducing the mission constraints discussed lies in improvement in mission performance while minimizing ground and/or spacecraft control functions. Potential cost savings realized through relaxation of mission constraints are not readily quantified. To do so would require the complete system (Space and Ground) designs and requirements specifications for both a solar-powered and a nuclear-powered spacecraft for the same mission and costing of both systems. After this, a cost comparison could be made to identify the cost savings that could be realized through the use of nuclear power.

At the present state of the NIMS design concept and except for survivability, it is only possible to indicate where and what overall mission benefits and simplifications occur with the use of nuclear power. The enhanced survivability cost/benefit assessment is presented in Section 8 of this volume.

SECTION 8

ECONOMIC ASSESSMENT

This section presents the details of the Rough-Order of Magnitude (ROM) cost estimates completed for this study. These data have been summarized and presented in Volume I, Summary.

The scope of this section includes the NIMS bus development and unit costs, and an assessment of the development and unit costs for a solar array/battery powered bus of the same power level (1200 Watts(e)). These costs are not directly comparable since although both bus concepts are enhanced to the same degree against the nuclear threat, only the NIMS bus provides enhanced survivability to the laser threat. The cost to implement this key NIMS attribute is therefore delineated. The key guidelines and assumptions employed in this cost assessment are presented.

8.1 GUIDELINES AND ASSUMPTIONS

This paragraph presents the guidelines and assumptions that are particular to this cost assessment.

8.1.1 GUIDELINES

The major guidelines employed are:

1. Maximum use is made of applicable cost data available at General Electric Space Division from recently completed and on-going spacecraft programs such as NIMBUS, LANDSAT, BSE and DSCS III. These data are utilized to define the development and unit costs for functional operations such as systems engineering, integration and assembly, test and evaluation, software, and program management. Specific subsystem costs such as the ACS, TT&C, thermal control, propulsion, electrical power, and structure

are also derived from this data base, as applicable to either the NIMS or solar array/battery powered spacecraft bus.

2. The Dynamic Isotope Power System (DIPS) costs, except for the radiator and its optical cover, are based on data supplied by Sundstrand Energy Systems through the DOE. These data are presented in Appendix A of this volume.

3. Component or subsystem costs that are particular to the NIMS bus, such as the radiator and the thermal control system are generated from recent engineering estimates. Maximum use is made of previous related work.

4. Costs of shared components are pro-rated. The prime example is the NIMS radiator which is utilized by both the DIPS and the NIMS thermal control system.

5. Isotope fuel (Pu-238) costs are treated parametrically. The actual fuel costs would be determined by separate negotiations between the Department of Energy and the user agency.

8.1.2 ASSUMPTIONS

The key assumptions employed in this cost assessment are:

1. No payload costs are included for any mission.
2. All estimates are in 1979 dollars.
3. Total dollar costs only are presented. There is no separate identification of material and labor costs.
4. An eight (8) percent fee is included in the summary costs.
5. No potential subcontractors are delineated, nor is any related additional G&A and fee included.
6. Unit costs assume a production rate of two-to-four units per year.

8.2 NIMS BUS COSTS

This paragraph presents the ROM cost assessment for the Nuclear Integrated Multimission Spacecraft (NIMS) development and unit costs, based on the NIMS design presented in Section 2.0 of this volume. These costs included enhanced survivability against the laser and the JCS nuclear threats.

8.2.1 NIMS BUS DEVELOPMENT COSTS

The estimated NIMS Bus development costs are summarized on Table 8.2-1*. The total of \$80,340,000 includes one engineering model and the qualification unit. No costs are included for any payload. These costs are based on recently completed or on-going spacecraft programs at the GE Space Division except for the DIPS electrical power system, and the radiator thermal control system which are presented in subsequent paragraphs. Only the thermal control and electrical power subsystems are significantly different from those in use on solar array/battery powered spacecraft. These costs are further detailed below.

System engineering includes areas such as system design, reliability analysis, survivability and production engineering. These activities are common with on-going solar array/battery powered spacecraft programs being conducted for the Government. Integration, assembly and support requirements are self-explanatory. Jigs, fixtures and other mechanical support equipment are required to support these assembly activities.

Test and evaluation costs include engineering development and qualification tests costs. The estimate given reflects a 10 percent allowance for the large size of the NIMS, but no estimate is included for a qualification test with isotope fuel. It is assumed that the qualification test can be accomplished utilizing electrical heat sources to simulate the isotope fuel.

*Rounded to nearest \$10K

TABLE 8.2-1 NIMS BUS DEVELOPMENT COST SUMMARY THROUGH
QUALIFICATION

<u>COST ELEMENT</u>	<u>\$ THOUSANDS</u>
Subsystems	
Attitude Control	6500
Tracking, Telemetry and Control	6800
Thermal Control	2290
Propulsion	4000
Electrical Power	18400
Structure	4800
Systems Engineering	12000
Integration, Assembly and Support	1600
Mechanical Support Equipment	200
Test and Evaluation	4000
Communications Subsystem Simulator	700
Ground Command and Control Equipment	1700
Command and Communications Software	3400
Launch and Orbital Operations Service	1200
Program Management	6800
Total NIMS Bus Development Cost Through Qualification	74390
Fee @ 8%	5950
Selling Price	80340

Other key cost elements, which are also common to solar array/battery powered spacecraft, include:

- 1) The communications subsystem simulator which interfaces with the TT&C subsystem.
- 2) Ground command and control equipment.
- 3) Command and communications software (computer codes).

Although it is possible that these may be adapted from other spacecraft programs, they are included for completeness for both for the NIMS and the solar array/battery spacecraft cost assessment presented in paragraph 8.3.1, below.

Launch and orbital operations supporting services development are required by all spacecraft programs. The costs shown are representative. Program management costs include project administration, cost control, configuration management, schedule management, and subcontract and material management. The costs estimated are similar to those for programs of this size.

8.2.1.1 Radiator Development Costs

The NIMS radiator development costs are estimated at \$1,854,000 through the unit for the NIMS qualification. One engineering unit is included. This cost is based on a separate engineering estimate performed for this study because of the semi-unique spiral tube configuration of the NIMS radiator, and its large size (12 feet diameter by 6.3 feet high).

The radiator is aluminum. A brazed/welded assembly procedure is defined. Eight separate subassemblies are fabricated where the tubes, armor, and skin are brazed together. The requirement for eight sections is dictated by the size of available brazing furnaces. The complete radiator is assemble by riveting the eight sections together, followed by welding of the coolant tubes. Special, localized cutouts in the radiator skin are provided to facilitate this welding operation. Subsequently, doublers to provide meteoroid protection are secured over these localized exposed surfaces.

The NIMS is covered with a special optical surface which enhances its ability to survive threat encounters. This surface is in the form of one inch (square) covers which are attached to the radiator metal surface by an existing adhesive. The capability to produce these covers in quantity is under development for other programs. Therefore, the cost of this development is not included in the NIMS radiator development cost. However, the covers are somewhat expensive and are required for the NIMS qualification unit. It is assumed that some fraction of the engineering unit radiator surface would be required to have the covers installed. This is taken at 25 percent for this cost estimate. The cover and cover installation costs for the NIMS are estimated at \$2880/ft² and \$120/ft² respectively. The total NIMS radiator costs through the qual unit are summarized on Table 8.2-2.

The NIMS radiator is common to both the DIPS and the spacecraft thermal control system. About 50 percent of its area is required for waste heat rejection from DIPS, and the remaining 50 percent is required for electronics cooling. Therefore, in the NIMS cost breakdown, 50 percent of the radiator cost is assigned to the electrical power system (DIPS) and the remainder is charged to the thermal control system.

8.2.1.2 NIMS Thermal Control System (TCS)

The NIMS thermal control system consists of an annular panel coupled by heat pipes to the NIMS radiator exit header and approximately one-half of the radiator, insulation blankets, local I²R heaters, structure, and related design and development. Electronics and other payload or house-keeping elements are mounted on this annular panel, and their waste thermal energy is dissipated through the heat pipes and the radiator, as detailed in paragraph 6.3 of this volume.

The cost of the annular panel including the attached heat pipe system is estimated at \$360,000 through the NIMS qualification unit. This is based on the cost history of flight hardware currently operating in space. The balance of the NIMS TCS development cost is estimated at \$1,000,000. The total cost of the NIMS thermal control system, including the 50 percent of the NIMS radiator required by this subsystem (paragraph 8.2.1.1) is \$2,287,000. as summarized in Table 8.2-3.

TABLE 8.2.-2 NIMJ RADIATOR DEVELOPMENT COSTS

COST ELEMENT	\$ THOUSANDS
Radiator Development and Engineering Unit (less covers)	960.0
Covers for 25 percent of Engineering unit	
Covers for 60 FT ² @ \$2880/FT ²	172.8
Installation for 60 FT ² @ \$120/FT ²	7.2
Radiator for Qual Unit (less covers)	
Covers for 100 percent of Qual Unit	
Covers for 238 FT ² @ \$2880/FT ²	685.4
Installation for 238 FT ² @ \$120/FT ²	28.6
Total Radiator Cost Through Qual	1854.0

TABLE 8.2-3 NIMS THERMAL CONTROL SYSTEM DEVELOPMENT COSTS
THROUGH QUALIFICATION

<u>COMPONENT</u>	<u>\$ THOUSANDS</u>
Annular cold plate panel with heat pipes	360
50 percent of NIMS Radiator (with covers)	927
Balance of TCS	1000
Total NIMS Thermal Control System Development (Through Qualification)	2287

8.2.1.3 NIMS Electrical Power System (EPS)

The estimated development costs for the NIMS electrical power system, including the DIPS, through the NIMS qualification unit are presented on Table 8.2-4. The DIPS costs, except for the radiator, are taken from Appendix A. No further breakdown is available at this time. The DIPS radiator cost is 50 percent of the total identified in paragraph 8.2.1.1 for the NIMS radiator and covers*.

The NIMS EPS costs also includes those elements which are required independent of whether solar array or nuclear power is used. These are:

- 1) The electrical harness.
- 2) Load Control Units (LCU) required to switch payload elements on and off.
- 3) Management costs associated with the integration of the EPS.

8.2.2 NIMS BUS UNIT COSTS

The NIMS bus unit costs are summarized in Table 8.2-5**. The total of \$14,710,000 assumes a low production rate two-to-four units per year. No costs are included for any payload. These costs are based on recently completed or on-going spacecraft programs at the GE Space Division except for the DIPS electrical power system, and the radiator and thermal control system which are presented in subsequent paragraphs. Only these subsystems are significantly different from those in common use on solar array/battery powered spacecraft. These are further detailed below.

8.2.2.1 Impact Of Isotope Fuel Cost

The costs presented on Table 8.2-5 do not include cost for the Pu-238 isotope fuel required by the DIPS in the NIMS Electrical Power System

* Radiator cost for DIPS development alone and independent of NIMS would be about 50 percent greater.

** Rounded to nearest \$10K

TABLE 8.2 -4 NIMS ELECTRICAL POWER SYSTEM DEVELOPMENT COSTS
THROUGH QUALIFICATION

COMPONENT	\$ THOUSANDS
Dynamic Isotope Power System (Less Radiator)	15000
DIPS Radiator ^(a) (less covers)	430
DIPS Radiator Covers and Installation ^(a)	447
Electrical Harness	450
Electronics ^(b)	830
NIMS Integration Management	1240
Total NIMS Electrical Power System (Through Qualification)	18397
<p>a) 50 percent of total NIMS radiator development cost</p> <p>b) Load Control Units (LCU) used to switch payload elements on and off</p>	

TABLE 8.2 -5 NIMS BUS UNIT COST SUMMARY

COST ELEMENT	\$ THOUSANDS
Subsystems	
Attitude Control	1100
Tracking, Telemetry and Control	1200
Thermal Control	690
Propulsion	2000
Electrical Power (less fuel)	2250
Structure	560
Systems Engineering	850
Integration, Assembly and Support	1100
Test and Evaluation	1500
Launch & Orbital Operations Services	570
Program Management	1800
Total NIMS Bus Unit Cost (Not including isotope fuel)	13620
Fee @ 8%	1090
Selling Price	14710

(EPS). These costs are nominally quoted by the DOE in terms of dollars-per-watt (thermal) encapsulated. Encapsulated refers to the fuel containment system required for space use of nuclear power to ensure safety throughout all normal operation and accident environments. Each baseline NIMS, as defined in this study, requires 7200 Watts(t) of Pu-238.

Figure 8.2-1 presents the NIMS bus unit cost as a function of the Pu-238 isotope fuel cost. These costs are based on 1973, prices quoted to DOD by the then AEC*, and include an annual seven (7) percent inflation to bring them up to 1979 levels. It is seen that advanced fuel production techniques could result in cost increments of \$700,000 to \$800,000, about six (6) percent of the total unfueled NIMS bus cost and the same order of magnitude as is associated with fabricating and installing the optical covers on the total NIMS radiator. This fuel cost increment increases to \$2,000,000 to \$3,000,000. for large quantity (300 kg) procurement of the fuel using current fuel production techniques, about 20 percent of the total unfueled NIMS bus cost and roughly three times the cost of the radiator optical covers. A 300 KG quantity of Pu-238 will provide fuel for more than 20 DIPS units, each producing 1200 Watts(e) at the end of a 10-year mission, the NIMS design point.

In the worst case, the isotope fuel cost would be \$6,000,000 to \$7,000,000 based on current fuel production techniques, about 50 percent of the unfueled NIMS bus cost. This would occur if only one or two NIMS flight units were built. These costs are comparable to the fuel costs charged to users such as NASA for Radioisotope Thermoelectric Generator (RTG) units used for spacecraft EPS for outer planet exploration missions.

The actual fuel costs in all cases are subject to negotiations between the user agency and the DOE.

8.2.2.2 NIMS Radiator Unit Costs

The NIMS radiator unit costs are estimated at \$120,000, not including the

* Letter to Dr. M. R. Currie, Director DR&D, DOD from Chairman, Military Liaison Committee to the USAEC, June 1974

BASELINE NIMS (DIPS)
REQUIRES 7200 WATTS (t)
OF PU-238 FUEL

FUEL COSTS SHOWN ARE
CORRECTED FROM 1973
AEC ESTIMATES TO 1979
DOLLARS USING SEVEN (7)
PERCENT ANNUAL
INFLATION (FACTOR OF
~1.5)

Figure 8.2-1. Impact of
PU-238 Fuel Cost on NIMS
Bus Unit Cost for Flight
Systems

optical covers. This cost is based on a separate engineering estimate performed for this study because of the semi-unique spiral tube configuration of the NIMS radiator, and its large size. The optical covers and cover installation costs for the NIMS are estimated at \$2880/ft² and \$120/ft² respectively. The total NIMS radiator unit cost, including the covers is \$833,000, as summarized on Table 8.2-6.

The NIMS radiator is common to both the DIPS and the spacecraft thermal control system. About 50 percent of its area is required for waste heat rejection from DIPS, and the remaining 50 percent is required for thermal control; electronics cooling. Therefore, in the NIMS cost breakdown, the radiator costs are similarly apportioned between the EPS and the TCS.

TABLE 8.2-6 - NIMS RADIATOR UNIT COST

<u>COST ELEMENT</u>	<u>\$ THOUSANDS</u>
Radiator unit cost (less covers)	120.0
Covers for 238FT ²	
Cover Cost @ \$2880/FT ²	685.4
Installation Cost @ 120/FT ²	28.6
Total Radiator Unit Cost	833.0

8.2.2.3 NIMS Thermal Control System Unit Costs

The NIMS thermal control system consists of an annular panel coupled by heat pipes to the NIMS radiator exit header, approximately one-half of the radiator, insulation blankets, local I²R heaters, and related structure. The unit cost of the annular panel including the

attached heat pipe system is estimated at \$120,000. This is based on the cost history of flight hardware currently operating in space. The total unit cost of the NIMS thermal control system is \$687,000, as summarized in Table 8.2-7.

TABLE 8.2-7 NIMS THERMAL CONTROL SYSTEM UNIT COST

<u>COMPONENT</u>	<u>\$ THOUSANDS</u>
Annular cold plate panel with heat pipes	120
50 percent of NIMS radiator (with covers)	417
Balance of Thermal Control System	150
Total NIMS Thermal Control System Unit Cost	687

8.2.2.4 NIMS Electrical Power System (EPS) Unit Costs

The estimated unit costs for the NIMS EPS are presented on Table 8.2-8. The DIPS unit cost, except for the radiator are taken from Appendix A. The DIPS radiator unit cost is 50 percent of the total NIMS radiator unit cost identified in paragraph 8.2.2.2.*

The NIMS EPS unit costs also includes those elements which are required independent of whether solar array or nuclear power is used. These are:

- 1) The electrical harness.
- 2) Load Control Units (LCU) required to switch payload elements on and off.
- 3) Management costs particularly associated with integration of the EPS.

* Radiator unit cost for DIPS alone and independent of NIMS would also be about 50 percent.

TABLE 8.2-8 NIMS ELECTRICAL POWER SYSTEM UNIT COSTS

COMPONENT	\$ THOUSANDS
DIPS Power Conversion Unit	400
DIPS Heat Source Assemblies (less isotope heat sources)	200
DIPS System Engineering and Program Management	800
DIPS radiator (a)	60
DIPS radiator covers and installation (a)	357
Electrical Harness	150
Electronics (b)	110
NIMS Integration Management	170
Total NIMS Electrical Power System Unit Cost	2247
a) 50 percent of total NIMS radiator unit cost	
b) Load Control Units (LCU) used to switch payload elements on and off	

8.3 SOLAR ARRAY/BATTERY (SAB) BUS COSTS

This paragraph presents the ROM cost assessment for a Solar Array/Battery (SAB) powered bus with multimission capabilities similar to that of the NIMS Bus. The SAB Bus has its solar array and batteries sized to provide the same power capabilities as the NIMS EPS, 1200 Watts(e) continuous through End-of-Mission (10 years). The significant difference between the SAB Bus and the NIMS Bus is that the SAB Bus does not include any special features to enhance its survival from the laser threat at the SMATH IV level.

No separate conceptual design for the SAB Bus was developed in this study. The cost data presented were generated from on-going and recently completed spacecraft programs such as NIMBUS, LANDSAT, BSE and DSCS III. Costs are adjusted to correct for differences in items such as power level, and are normalized to 1979 dollars.

8.3.1 SAB BUS DEVELOPMENT COSTS

The estimated SAB Bus development costs are summarized on Table 8.3-1. The total of \$65,800,000 includes one engineering model and the qualification unit. No costs are included for any payload. The major difference between the development cost of the NIMS Bus and the SAB Bus is the \$16,200,000 DIPS development cost.

All subsystems are similar to those in current use on solar array/battery powered spacecraft. The thermal control system is passive, as opposed to the NIMS active cooling system.

Systems engineering includes areas such as system design, reliability analysis, survivability, and production engineering. Integration, assembly and support requirements are self-explanatory. Jigs, fixtures and other mechanical support equipment are required to support these assembly activities.

Test and evaluation costs include engineering development and qualification testing. Other conventional cost elements include:

TABLE 8.3-1 SOLAR ARRAY/BATTERY BUS DEVELOPMENT COST SUMMARY
THROUGH QUALIFICATION

<u>COST ELEMENT</u>	<u>\$ THOUSANDS</u>
<u>Subsystems</u>	
Attitude Control	6500
Tracking, Telemetry and Control	6800
Thermal Control	1700
Propulsion	4000
Electrical Power	5300
Structure	5400
Systems Engineering	12000
Integration, Assembly and Support	1600
Mechanical Support Equipment	200
Test and Evaluation	3600
Communications Subsystem Simulator	700
Ground Command and Control Equipment	1700
Command and Communications Software	3400
Launch and Orbital Operations Services	1200
Program Management	6800
 Total Solar Array/Battery Bus Development Cost Through Qualification	 60900
 Fee @ 8%	 4900
 Selling Price	 65800

- 1) The communications subsystem simulator which interfaces with the TT&C subsystem.
- 2) Ground command and control equipment.
- 3) Command and communications software (computer codes).

Launch and orbital operations supporting services development are required by all spacecraft programs. The costs shown are representative. Program management costs include project administration, cost control, configuration management, schedule management, and subcontract and material management. The costs estimated are similar to those for programs of this size.

8.3.2 SAB BUS UNIT COSTS

The SAB Bus unit costs are summarized in Table 8.3-2. The total of \$13,180,000 assumes a low production rate of two-to-four units per year. No costs are included for any payload.

Key items that differ from the NIMS Bus unit costs (Table 8.3-2) are:

- 1) The Thermal Control System (TCS) is less expensive, \$300,000 vs. \$687,000. However, the SAB TCS does not provide an integrated TCS and does not provide enhanced laser survivability.
- 2) The Electrical Power System (EPS) is less expensive, \$1,150,000 vs. \$2,247,000 plus fuel. This SAB EPS does not include any laser threat survivability enhancement.
- 3) The structure is slightly more expensive, \$710,000 vs. \$560,000 because of the solar array related structural items normally carried in this account.

Most of the other cost elements are estimated to be the same for either the NIMS or the SAB Bus.

TABLE 8.3.-2 SOLAR ARRAY/BATTERY BUS UNIT COST SUMMARY

COST ELEMENT	\$ THOUSANDS
Subsystem	
Attitude Control	1100
Tracking, Telemetry and Control	1200
Thermal Control	300
Propulsion	2000
Electrical Power	1150
Structure	710
Systems Engineering	850
Integration, Assembly and Support	1100
Test and Evaluation	1420
Launch and Orbital Operations Services	570
Program Management	1800
Total Solar Array/Battery Bus Unit Cost	12200
Fee @ 8%	980
Selling Price	13180

8.4 NIMS COST/BENEFITS ASSESSMENT

This paragraph delineates the estimated cost increments attributed to the NIMS bus which provide its laser threat survivability at the SMATH IV level, relative to the SAB Bus.

The incremental NIMS Bus development cost is \$14.54 million, per the data of Table 8.2-4 and Table 8.3-1. The incremental NIMS Bus unit cost is 1.53 million, plus the isotope fuel cost, per the data of 8.2-8 and Table 8.3 -2. The isotope fuel costs are taken to be those representative of large quantity production by conventional techniques. This is valid for comparison with SAB Bus costs for missions in which many (5-to-25) spacecraft are required. A fuel cost of \$487.50 per Watt (thermal), encapsulated is assumed in 1979 dollars. This corresponds to the midpoint of the costs of \$300 per Watt(t)-to-\$350 per Watt(t) quoted to the DOD by the AEC in 1973 dollars*. The baseline NIMS requires 7200 Watts(t) of Pu-238, which therefore translates into a cost of \$3.51 million for each NIMS Bus. The total NIMS bus unit cost to enhance laser hardening at the SMATH IV level is then estimated at \$5.04 million in 1979 dollars, including the \$3.51 million fuel cost. This assumes that the NIMS development costs are written off and not amortized over any NIMS production run.

The impact of amortizing the NIMS and SAB development costs over a number of units is also presented on Figure 8.4-1. The NIMS fuel costs are included at a constant value of \$3.51 million per unit. The costs estimated to enhance laser hardening at the SMATH IV level using the NIMS concept decrease from \$8 million for the fifth flight unit, down to \$5.7 million for the twentieth unit.

Costs necessary to provide the same level of laser hardening for a SAB Bus are unknown.

* op. ct.

Figure 8.4-1. Enhanced Laser Survivability is Cheap with Nuclear Power

APPENDIX A - DYNAMIC ISOTOPE POWER SYSTEM

A1. - SUMMARY

In August of 1975 the Department of Energy initiated the organic Rankine Dynamic Isotope Power System (DIPS) program with Sundstrand. This advanced space power system development is directed for application in the 1980's and beyond. The Phase I effort which was completed in July of 1978 successfully completed:

- 1) a flight system conceptual design
- 2) design, fabrication and testing of a flight prototype ground demonstration system
- and 3) a 1,000 hour endurance test.

Following the Phase I effort the program progressed into the Technology Verification Phase to:

- 1) prove viability of all program technology elements
- 2) demonstrate over 18% system efficiency
- and 3) complete an endurance test of a minimum of 5,000 hours.

This phase of the program will be completed in October of 1980 at which time the system will be ready to commence the ground qualification program in preparation for a flight test or application.

This appendix describes the DIPS as a self contained system, however, as the DOE studies with General Electric in developing the NIMS concept shows, the system can be readily integrated as part of the spacecraft.

A2. - SYSTEM DESCRIPTION

The current baseline Flight System Design (FSD) is illustrated in Figures A1 and A2.

The heat source assembly is located symmetrically around the power conversion system and at the base of the radiator assembly which is mounted to the spacecraft bulkhead as shown in Figure A1.

The major sub systems are:

- Power Conversion System (PCS)
 - Combined Rotating Unit (CRU) - turbine, alternator, pump, and bearings
 - Electrical and fluid controls
 - Regenerator
 - Jet Condenser

Power
Conversion
Module

Radiator

Isotope
Heat
Sources

Figure A1. Flight System Design

Figure A2. Power Conversion System

SYSTEM DESCRIPTION (cont.)

- Heat Source Assembly (HSA)
 - Heat source - General Purpose Heat Source (plutonium dioxide)
 - Boiler and auxiliary cooler
 - Emergency Heat Dump System (EHDS)
- Radiator Assembly (RA) (includes auxiliary cooler)

Key features of the design are an axial impulse turbine, homopolar inductor alternator, centrifugal pump, working fluid (Dowtherm A) lubricated fluid film bearings, once-through radiantly heated boiler, jet condenser and liquid working fluid pumped radiator. The baseline DIPS is designed for 1300 We, 28 v D.C., weighs less than 450 pounds and requires 116 square feet of radiator area. The radiator geometry is flexible since it is a pumped liquid type.

Figure A3 is a schematic of the system illustrating typical operating conditions. The liquid outlet from the pump is split into two loops. The power loop passes liquid through the regenerator to extract energy from the turbine exhaust vapor. At the energy source, additional heat is transferred to the working fluid which is vaporized to drive the turbo-alternator-pump. The cooling loop passes liquid flow through the radiator to reject waste heat and then to the jet condenser to condense the turbine exhaust vapor emanating from the regenerator. The combined liquid is returned through the accumulator to the pump.

A start module is used to start the power system either for a ground start or a space start. The preferred mode of operation is a ground start. In this case, the start module would be disconnected prior to launch and the power system would be operating and producing partial power as required during launch and orbit transfer.

A temperature control support module is used for cooling during the period between loading the heat source into the HSA and power system startup. It is also used as auxiliary cooling for the heat source while the system is in the Space Shuttle. This cooling is supplied by a ground module for the Titan IIIC.

The heat rejection support module is used for on the pad cooling prior to launch for Titan IIIC applications and continuously while in the Shuttle bay. These support modules are decoupled after performing their function and are not part of the DIPS/Spacecraft system that is injected into operational orbit. The STS thermal support module is currently under development for use on the STS launched Galileo and Solar Polar NASA missions, which use isotope power systems.

Figure A4 presents the typical DIPS operating conditions as established when the DIPS is integrated with the NIMS. The thermal interface between the DIPS radiator discharge flow and the heat load from the NIMS payload and housekeeping electronics (including the DIPS electronics control system) is illustrated.

A-5

<u>PRESSURE</u>		<u>TEMPERATURE</u>	
	BARS	(PSIA)	
			°K
			(°F)
①	6.65	(96)	373.8
②	5.96	(86)	348.2
③	5.06	(73)	380.1
④	4.75	(69)	520.1
⑤	3.93	(57)	630.2
⑥	.009	(.137)	558.7
⑦	.007	(.105)	389.1
⑧	2.18	(31)	373.4

Figure A3. Basic System Arrangement

Figure A4. Typical DIPS Operating Conditions - NIMS Integration

PCS DESCRIPTION

Working Fluid

For the DIPS application, Dowtherm A was selected after extensive trade-offs with other organic fluids. Dowtherm A exhibits the characteristic positive slope of the saturated vapor line, Figure A3, allowing 100% quality vapor expansion through the turbine and use of a regenerator for increased system efficiency.

Combined Rotating Unit (CRU)

The Combined Rotating Unit (CRU) extracts work from the hot vapor with the turbine, thereby driving the system feed pump and electrical alternator. The CRU comprises an axial impulse turbine driving a homopolar induction alternator and a centrifugal pump. The rotor is supported radially and axially by working fluid lubricated, hydrodynamic, tilting pad bearings.

The CRU is integrated with the regenerator and jet condenser to minimize volume and weight. The complete package is supported at three points located in the regenerator exit plane. All housings are made from stainless steel and all support struts are aluminum alloy.

Fluid and Electrical Controls

Flow Control Valve:

The Rankine Cycle power conversion system has a flow control device to maintain turbine inlet conditions at the selected state point which insures that the turbine will always be supplied with superheated vapor. In the orbital operating mode, speed is controlled by a speed control parasitic load bank and the flow control valve only corrects for minor turbine inlet temperature fluctuations and isotope heat source decay. During launch and orbit transfer, the parasitic load control is disabled, allowing the CRU to operate at higher than normal speeds which increases bearing load capacity and jet condenser resistance to flooding. This higher speed also increases pump outlet pressure and, hence, the required flow control valve pressure drops.

Radiator Thermal Bypass Valve:

The changes in radiator environment during the mission cause a variation in heat rejection capability. With a cold radiator sink environment average liquid temperature in the radiator could be as much as 40°F cooler than in high sink temperature environments. This subcooling of the working fluid would result in a reduction in system power output capacity as the turbine temperature flow control valve would decrease flow slightly to compensate for the lower temperature fluid entering the boiler. Therefore, heat rejection control is required to assure that the power system can deliver rated load regardless of the radiator environment. This is accomplished by controlled bypassing of radiator inlet fluid around the radiator.

Speed Control and Voltage Regulator:

Independent control of speed and voltage is used to get the best possible quality of power. The speed control regulates the power to the parasitic load resistor to maintain the turbo-alternator at virtually constant speed/

frequency. The voltage regulator independently adjust the alternator field current to maintain a constant output voltage. Simple circuits are used for both and careful parts selection and extensive derating provides high reliability.

Parasitic Load Resistors:

Three resistors and associated power switches are used to reduce the current per switch to a level suitable for available JAN-TX switching transistors. This arrangement places the dissipation in a resistor which can operate at high temperature and keeps the semi-conductor device losses low for high reliability.

Regenerator

The regenerator comprises an all aluminum finned tube wound in a spiral fashion around the electrical generator housing. This configuration minimizes the power generating system length, volume and weight and simplifies the vapor flow passage while keeping the axis of rotation co-linear with the jet condenser axis for launch load considerations.

As the vapor leaves the turbine diffuser, it undergoes a sudden expansion at the entrance of the regenerator. The vapor then flows over the outside surface of the finned tube, giving up the superheat to the internally flowing liquid before entering the annular transition duct which accelerates the flow to the jet condenser.

The regenerator outer shell forms part of the system load carrying structure. The inner shell serves partly to support the alternator. A slip joint in the inner shell allows for ease of assembly and to accommodate differential thermal expansions.

Jet Condenser

The jet condenser consists of three basic components: the vapor funnel, the liquid injector and the diffuser (Figure A5). Low pressure vapor is ducted into a funnel coaxially with liquid jets of subcooled working fluid. The liquid jets are injected into the vapor at approximately 100 fps and aimed at the throat of the vapor funnel. The vapor condenses on the subcooled liquid jets and the combined liquid jet passes through a throat and into a diffuser where it undergoes a sudden expansion to fill the entire cross-section. In the sudden expansion, there is a loss of total pressure but a gain in static pressure of the liquid stream. The behavior of the liquid jet in the diffuser is identical to that which occurs in the diffuser of a cavitating venturi. Testing has shown that for the DIPS condenser, a 50% recovery is achievable.

Since the process is based on liquid momentum for its operation, there is no requirement for a specific vapor pressure drop to insure proper condensate flow direction in negative "g" fields as is the case for surface condensers. Thus, a very low back pressure can be used to improve turbine isentropic head and volumetric flow, increasing turbine and overall system efficiencies. The jet condenser will work during launch and in zero "g" since it relies on injected liquid momentum.

Heat Source Assembly

Each Heat Source Assembly (HSA) consists of ten General Purpose Heat Source

Figure A5. Conceptual Jet Condenser

(GPHS) modules- a boiler fin tube assembly, heat source mounting structure, insulation, housing and mounting provisions. The HSA concept is shown in Figure A6.

The remainder of the HSA consists of an outer case, which is the main support structure for the isotope heat source, the end closure structures, the stainless steel boiler tube/copper fin heat exchanger, and the required multifoil and bulk insulation necessary to maintain heat losses at a tolerable level. There are three in-series heat source assemblies in the 1.3 KW DIPS.

Each heat exchanger consists of a pair (one for auxiliary cooling) of stainless steel helical tube coils with continuous 0.020 inch thick copper fins attached. The inside of the boiler is fitted with inserts for subcooled, 2-phase, and superheated zones which are designed to produce a minimum 12 "g" radial acceleration to counter the expected "g" loads during launch and orbit transfers.

As presently conceived, the emergency heat dump system for the DIPS is an aluminum multifoil insulation blanket surrounding the boiler tube-fin assembly which melts when cooling is lost.

Radiator Assembly

The DIPS space radiator employs a forced convection heat transfer loop. It attaches directly to the spacecraft mounting plate at three places.

The prototype radiator constructed during Phase I is an aluminum, welded, riveted assembly which passes organic fluid through sixteen vertical tube extrusions. Each extrusion serves as a passageway for the organic fluid, as the required frontside and backside meteoroid armor protection, and as vertical stiffening for the radiator shell, which is 0.025 inch thick. Nine equally spaced circumferential frames are riveted to the skin for structural support.

At present, the exterior radiator surfaces are coated with IITRI Z-93 thermal control coating. The Z-93 coating offers acceptable performance-related material properties coupled with good reproducibility. However, future work will enable determination of radiation hardness requirements and a different coating may be applied for some flight applications. Hardening considerations also have led to the use of the spiral tube radiator depicted in Figure A1 rather than the axial tube type.

The system radiator will not be rejecting heat during and after launch while still shrouded inside the launch vehicle. For a Space Shuttle launch, cooling of the working fluid is accomplished by means of a simple double pipe heat exchanger located in series with, and at the entrance to the system radiator. For a Titan IIIC launch, this same heat exchanger is used to pre-cool the radiator which will act as a thermal capacitor during launch until the shrouds are separated from the launch vehicle.

Figure A6. General Purpose Heat Source Assembly

A3. - COST PROJECTIONS

A. 3.1. - DYNAMIC ISOTOPE POWER SYSTEM

The budgetary estimates for costs associated with ground qualification of the DIPS, flight hardware, and support equipment are presented here.

- 1) After successful completion of the technology verification phase in October 1980, the program could proceed to ground qualification of the DIPS in preparation for a flight application or flight test.

Estimated Cost	\$15 Million
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- 2) Fabrication and acceptance testing of flight system elements.

Power Conversion System	\$400,000.
Heat Source Assemblies (3 unfueled)	\$200,000.
Radiator (not integrated with spacecraft)	\$100,000.
Engineering/Technician Support (10 man level)	\$800,000./year

- 3) Three support equipment items are required.

Start Cart	\$75,000.
Heat Source Assembly Cooling Cart	\$50,000.
Control Console	\$25,000.

A.3.2 - ISOTOPE FUEL

Fuel cost estimates for this report to be furnished by DOE.

A4. - DEVELOPMENT STATUS

Based on the flight system design (FSD) described in Section A2, a ground demonstration system (GDS) was designed, fabricated and tested to prove the feasibility of the flight system (Figure A7).

Phase I-

GROUND DEMONSTRATION SYSTEM (GDS)

The GDS as tested is prototypic of the FSD in all major aspects of design and operation. The principal differences are (1) the use of "O" rings and flanges for ease of assembly and to permit convenient disassembly of the GDS hardware, (2) the use of electrical heat source assemblies instead of a radioisotope, and (3) the use of a conventional liquid-to-liquid heat exchanger instead of a space radiator. A prototype radiator was constructed and tested at the component level.

Figure A8 is a schematic of the GDS. Operationally, it is nearly the same as the FSD. Three units of ground support equipment are used: a start module for starting the system; a heater temperature control system which maintains heater block temperatures at desired values when the system is inoperative; a heat rejection module which removes system waste heat in the absence of the radiator.

To minimize air in-leakage and to simulate heat losses associated with space operation, the system was operated in a vacuum chamber. System waste heat was rejected via the auxiliary heat exchanger which would be in use during launch by the Space Shuttle, while the radiator pressure drop was simulated with an orifice.

Initial shakedown and testing was performed on a development system which did not incorporate a later developed high performance turbine. After testing of the development system, the high performance turbine was retrofitted yielding the GDS configuration.

In Phase I, over 1100 hours of operation were achieved on the GDS with more than 1600 hours on those components which were also run on the development system. System performance of greater than 15% overall efficiency has been demonstrated on the GDS and over 18% is achievable with further development. Table AI summarizes the major GDS test results.

Figure A7. Ground Demonstration System

Figure A8. GDS Schematic

TABLE A I
SUMMARY OF PHASE I TEST RESULTS

GROUND DEMONSTRATION SYSTEM

Total Hours on GDS	1114*
System Efficiency (28 VDC Output)	15.15%
Number of Startup Cycles	29

DEVELOPMENT SYSTEM

Total Hours on Development System	540**
System Efficiency (28 VDC Output)	11%
Number of Startup Cycles	118

NOTE: Total hours on components used in development system and GDS	1654
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* Final GDS configuration with NaK sensor flow control valve had two continuous runs of 447 and 573 hours (total of 1020 hours). Intermediate shutdown was due to unscheduled plant power interruption. Intentional shutdown implemented after the next 573 hours.

** Longest continuous run was 470 hours. Intentional shutdown.

Technology Verification Phase (TVP)

At the end of Phase I specific areas had been identified in which improvement in efficiency could be realized with further development. The current Technology Verification Phase is to improve the system efficiency by component development and to further demonstrate system reliability by endurance testing the GDS a minimum of 5000 hours.

Areas identified in the Phase I program where system performance can be improved with further development are as follows:

Dynamic Seals - A major impact on GDS performance is liquid leakage from the turbine bearing cavity onto the turbine wheel, causing additional drag and reduced turbine outlet temperatures. Marked improvement through reduced seal leakage was achieved during Phase I but leakage still had a significant impact on system performance. During the TVP, a seal development test rig will be constructed to permit selection of a seal configuration combining the requirements of near zero leakage and acceptable seal parasitic losses.

Turbine - Turbine performance was improved from 55% efficiency in the development unit to 66% efficiency in the GDS by reducing the blade chord width and

changing the blade channel configuration. Further improvements have been identified to obtain a 69% efficient turbine. TVP component testing will incorporate wheels of still narrower (.3 inch) chord width, full admission nozzle rings, varying blade nozzle lap ratios and some will incorporate a small degree of reaction.

Regenerator - A .95 effective regenerator was demonstrated in the development unit. GDS regenerator performance dropped to .92 effectiveness because of schedule and size limitations of a differently fabricated fin tube. TVP testing will include a retrofit to the original higher performance configuration.

Alternator/Controller - Investigations performed during Phase I identified further improvements which could be made to increase the alternator efficiency one point from the demonstrated 93%. In the TVP, an alternator test rig will be constructed to evaluate potential alternator improvements by reducing the pole face losses and optimizing the rotor/stator "air" gap. In addition, testing will be done with the alternator in conjunction with the electronic controller to maximize overall electrical conditioning efficiency by a varying tuning capacitor size and by evaluating lower loss rectifiers.

Jet Condenser - In addition to the seal leakage requiring extra vapor condensation, maximum recovery was not as high as expected, possibly due to inadequate focusing of the 45 nozzle jet condenser. The higher performance 90 nozzle jet condenser was not incorporated in Phase I due to the low yield of acceptable stainless steel drilled nozzles. In the TVP elimination of nozzle instability of the 90 nozzle jet condenser by the use of nozzles with synthetic sapphire jewel orifices will result in an improved performing jet condenser.

Performance Results and TVP Plan

Table A II presents the results of the Phase I testing and the predicted improvement attainable in system efficiency by the component improvements.

TABLE A II
SYSTEM PERFORMANCE IMPROVEMENTS

	GDS	TVP Goal
System Efficiency	15.1%	18%
Seal Leakage (Lb/Hr)	4.5	0
Turbine Efficiency (%)	66	69
Regenerator Effectiveness	.92	.95
Alternator Efficiency (%)	93	94
Jet Condenser Vapor Pressure (PSIA)	.11	.105

The components offering the largest potential improvement in system efficiency will be concentrated on during this development phase. Work will also be addressed in certain component areas, however, where it is desirable to expand the data base.

When an improved component configuration has been established in the component development test program, GDS hardware for that component will be designed and fabricated. This hardware will then be retrofitted into the GDS for performance evaluation and continuing endurance testing. The GDS will only be shut down to retrofit improved components and thus accumulate maximum run time. After retrofit of the last improved component, the upgraded system performance will be evaluated by tests.

After completion of the system performance tests, the GDS will be placed on a 5000 hour (minimum) endurance test. Every reasonable effort will be made to assure that inadvertent shutdowns do not occur. Continuous running is the goal.

The Flight System Design which was initiated in Phase I will be updated to reflect desirable changes resulting from further analysis, the results of the Technology Verification Phase testing and inputs from spacecraft designers and users.