N94-25701

LOW EARTH ORBIT COMMUNICATIONS SATELLITE

D. Moroney, D. Lashbrook, B. McKibben, N. Gardener, T. Rivers, G. Nottingham, B. Golden, B. Barfield, J. Bruening, D. Wood, D. Pauls, W. Bell, D. Price, and C. Baldwin

Abstract

A current thrust in satellite communication systems considers low-Earth orbiting constellations of satellites for continuous global coverage. Conceptual design studies have been done at the time of this design project by LORAL Aerospace Corporation under the program name GLOBALSTAR and by Motorola under their IRIDIUM program. This design project concentrates on the spacecraft design of the GLOBALSTAR low-Earth orbiting communication system.[1] Overview information on the program was gained through the Federal Communications Commission licensing request.

The GLOBALSTAR system consists of 48 operational satellites positioned in a Walker Delta pattern providing global coverage and redundancy. The operational orbit is 1389 km (750 nmi) altitude with eight planes of six satellites each. The orbital planes are spaced 45°, and the spacecraft are separated by 60° within the plane. A Delta II launch vehicle is used to carry six spacecraft for orbit establishment. Once in orbit, the spacecraft will utilize code-division multiple access (spread spectrum modulation) for digital relay, voice, and radio determination satellite services (RDSS) yielding position determination with accuracy up to 200 meters.

Introduction

GLOBALSTAR is a satellite system designed to provide global radio-determination satellite services (RDSS) for real-time position location and tracking, and voice and data services to mobile users. Rather than being a completely self sufficient system, GLOBALSTAR is intended to be integrated into the existing public switched telephone network (PSTN), personal communications networks, and private, specialized, and cellular networks. By complimenting existing carriers' networks, GLOBALSTAR is designed to make RDSS, voice, data, fax, and freeze-frame video available to users anywhere in the world. The GLOBALSTAR program concept (which defined the design parameters for this preliminary design) is defined in the Federal Communications Commission (FCC) licensing request submitted by Loral Aerospace Corporation.[2]

Specifications

The spacecraft is designed in accordance with the parameters set forth in the filing of the FCC license request. A constellation of 48 operational satellites, positioned in a Walker Delta pattern, provides global coverage and redundancy. There is no allocation for on-orbit spares, since a minimum of two satellites are in view at any time within the target area. Table 15 depicts the design and performance specifications of the spacecraft.

Table 15 Design and Performance Specifications

Orbit	
Altitude	1389 km (750 nmi)
Eccentricity	0° (circular)
Coverage	75° N/S latitude
Launch vehicle	Delta and Ariane
Mission life	5 years
Initial launch date	01 July 1996
Station keeping	
Stabilization	3-axis stabilized
Pointing accuracy	±1.0° all axes
Electrical	
Bus	28 Vdc
Operation	Continuous (100% during eclipse)
Thermal control	600 watts for 33 minutes (max.)
Payload	
Mission	Direct voice and data relay network; voice and data link position location
Power requirements	800 watts for 20 minutes (peak); 50 watts (nominal)
Mass	60 kg

The spacecraft consists of the communication payload and the following subsystems: structure; telemetry, tracking, and command (TT&C); electrical power; thermal control; attitude control; and propulsion subsystems. In addition, launch vehicle integration and orbit dynamics were studied. The Delta II launch vehicle was selected for placing six spacecraft into orbit at a time. The spacecraft is deemed compatible with the Ariane launch vehicle also because of the stringent mass, volume, and environmental loads of the Delta II.

Payload

The mission of the satellite constellation is to provide voice, data, messaging, and position location information to users via cellular telephone during those times when terrestrial cellular networks are not within range or are unavailable. Coverage extends from 75° South latitude to 75° North latitude. Both the user and a gateway must be in the field-of-view of at least one spacecraft to complete a communication link. Radio Determination Satellite Service (RDSS) can provide position accuracy of approximately one mile for single satellite operations and approximately 200 meters resolution when two satellites are in view.

The uplink and downlink between users and the spacecraft are at L-band (1610.0 to 1626.5 MHz). Gateway to spacecraft links are at C-band with uplink at 6525.0 to 6541.5 MHz and downlink 5199.5 to 5216.0 MHz. User channels provide voice and data at a variable 1.2 to 9.6 kbps with encoding/decoding. The spacecraft does not have onboard processing. Encoding and decoding is accomplished on the ground.

Six elliptical spot beams provide coverage over the spacecraft footprint. The elliptical beams are aligned with the major axes in the direction of the ground track path. Frequency reuse is provided by utilizing the same frequency range on two beams at a time but widely separated on the ground. The L-band is divided into 13 subbands of 1.25 MHz bandwidth for frequency-division-multiple-access (FDMA). Within each subband, spread spectrum techniques are used for code-division-multiple-access (CDMA). Two antenna beams of the total six are enabled at any one time over a 60 millisecond duty cycle yielding a system of beam hopping time-division-multiple-access (TDMA). This combination of FDMA, CDMA, and TDMA provides a total of 2626 full-duplex channels per spacecraft.

The payload mass and power budgets are shown in Tables 16 and 17, respectively.

Table 16 Payload mass budget

Component	Quantity	Mass (kg)
C-band antenna	2	1 each
L-band antenna	6	10 total
C-L band transponder	1	20
L-C band transponder	1	20
Timing and control unit	1	8
Total	11	60

Table 17 Payload power budget

Peak load	827.6 watts
Peak transmitted power	43.32 watts
Peak thermal dissipation	784.28 watts

Orbit Analysis

Orbit analysis for the constellation of satellites considered the ground coverage, phasing of satellites, number of planes, and number of satellites per plane for global coverage between 75° North and 75° South latitude. Additional consideration was given to the radiation environment, space debris, and atmospheric drag.

Global Coverage

The payload requirements allowed no flexibility in orbit altitude or eccentricity. The goal of the analysis was to minimize the total number of satellites necessary to provide 100% coverage in the prescribed area. The total number of satellites is 48. They are distributed in eight orbit planes separated by 45° of the ascending node. Each orbit plane has an inclination of 52°. Six satellites are placed in each orbit plane, evenly spaced. Overlap was determined to allow a minimum of two satellites in view at all times. Phasing of the satellites between adjacent orbit planes is 7.5° in true anomaly for collision avoidance. The constellation orbit planes are shown in Figure 7.

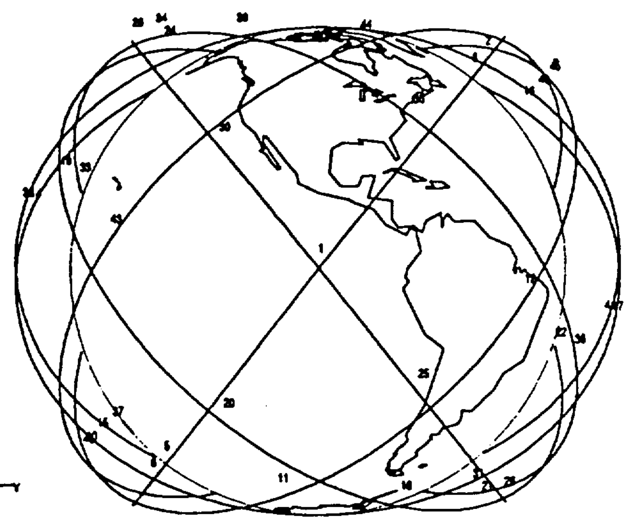

Fig. 7 Constellation orbit planes

Orbit Environment

The altitude of the satellites equates to 1.22 Earth radii. The radiation level experienced due to Van Allen Radiation Belts exposure is lower than that normally experienced by a geosynchronous satellite. Minimal shielding of electronic components is necessary, and solar array degradation over the five-year life of the satellite does not adversely affect the design.

The scheduled launch for initial constellation establishment is at a time of minimum solar activity. However, solar activity may be near a maximum at end-of-life (EOL), which may interfere with satellite communication at that time. Solar activity directly affects the amount of atomic oxygen in the upper atmosphere. 100 particles per cubic centimeter of atomic oxygen are normally present at the constellation's operational altitude during low solar activity. During solar maximum, 10^5 particles per cubic centimeter are present. Atomic oxygen may accelerate the degradation processes near end-of-life.

Orbit Perturbations

Orbit perturbation analysis was performed using the Orbital Workbench® software by Cygnus Engineering. The Cowell propagation method was used with zonal harmonics J2 through J6, tesserals, lunar gravity, solar gravity, and solar pressure forces. Results identify that the only stationkeeping required is in-plane to maintain altitude. Additionally, total on-orbit atmospheric density is less than 10^{-13} kg/m^3 during solar maximum and less than 10^{-14} kg/m^3 nominally. Atmospheric drag at the operational altitude is considered negligible.

Orbit Injection

The launch vehicle will place the satellites into an injection orbit that is slightly elliptic and coplanar with the operational orbit. The injection orbit was chosen to allow one satellite to be transferred into its operational orbit for every three orbit periods of the elliptical injection orbit once the initial satellite is injected. The injection orbit parameters are given in Table 18.

Table 18 Injection orbit parameters

Semi-major axis (km)	8052.22
Eccentricity	0.035
Inclination (deg)	52.0
Radius at apogee (km)	8337.0
Radius at perigee (km)	7767.15
Period (min)	119.85

Launch Vehicle

Description

The spacecraft is designed to be compatible with both the Delta II and Ariane launch vehicles. The Delta II was chosen as the targeted launch vehicle for this study because of the stringent mass, volume, and environmental loads. The Delta II 7925 is capable of putting a payload mass of 3300 kg into the required transfer orbit.[3] The Delta II 7925 launch vehicle configuration is shown in Figure 8. The first stage is a liquid propellant booster with strap-on solid rocket motors. The second stage is powered by a pressure-fed propulsion system. The third stage is a PAMSTAR 48 solid rocket motor.

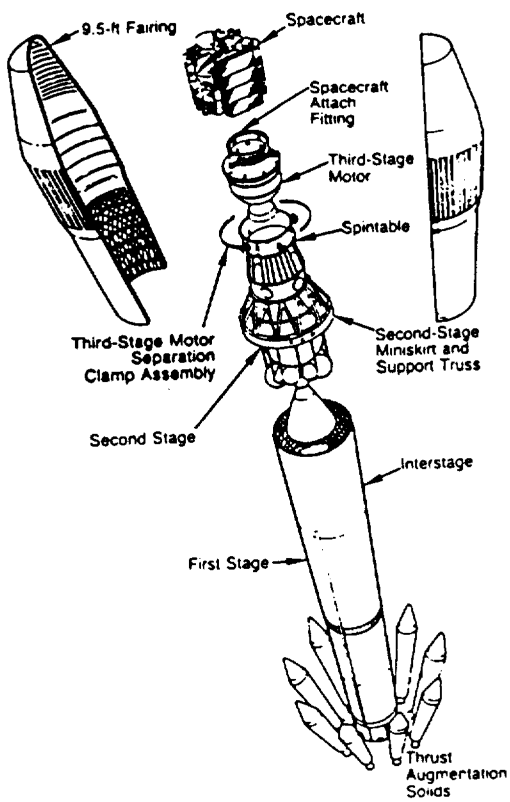

Fig. 8 Delta II 7925 launch vehicle

Satellite Launch Dispenser

Satellites are launched to establish a full orbit plane for a single Delta II launch. The spacecraft are stowed in a Satellite Launch Dispenser (SLD), which holds six spacecraft. The SLD is designed to withstand the launch loads for all six satellites. This eliminates the need for any single satellite to bear the weight of the other satellites in a stacked configuration. The spacecraft are attached to the SLD via two dispensing rails fixed with pyrotechnic bolt-latch fittings, which also serve to compress a spring-plunger mechanism at the end of each rail. The bolts are fired, releasing the springs and deploying the spacecraft. The dispensing rails provide their own bearing race that rigidly supports the spacecraft. The mating rails on the spacecraft are attached at the top of the spacecraft. The SLD is shown in Figure 9.

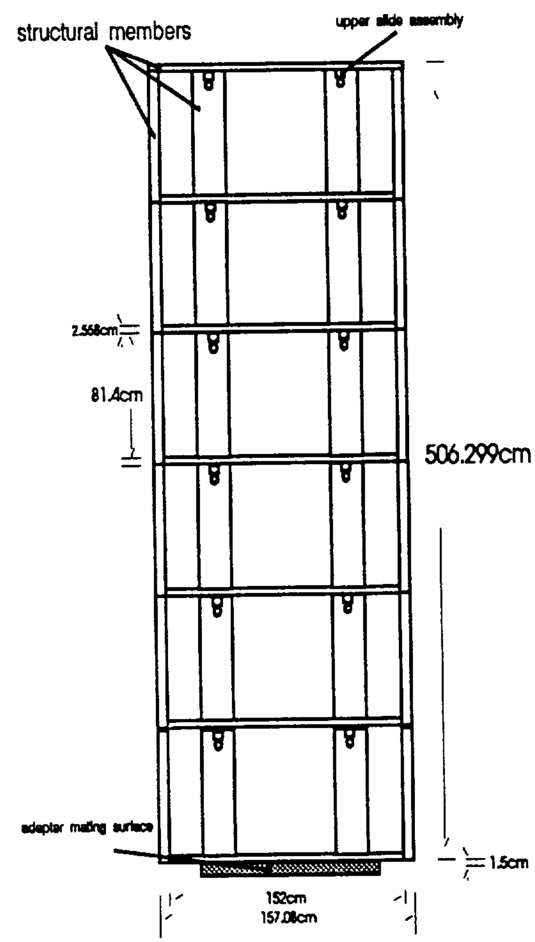

Fig. 9 Satellite Launch Dispenser (SLD)

Subsystem Description

Telemetry, Tracking, and Control (TT&C)

The TT&C system must be able to communicate both spacecraft commands and ephemeris data as well as to downlink subsystem health and performance data. The ground-based portion consists of two TT&C stations located in CONUS to perform tracking and relay functions. Data is then relayed to the Satellite Operational Control Center (SOCC) where telemetry is processed and attitude control and system commands are originated. The SOCC then relays the commands back to the TT&C ground stations and distributes ephemeris data to the gateway network.

The TT&C subsystem is independent of the payload and consists of the remote tracking unit (RTU) and the remote control unit (RCU). Full redundancy is employed. The RTU provides encoding, decoding, modulation, demodulation, and transmit/receive functions. The RCU interprets and performs commands, acts as the databus controller, and stores subsystem data for downlink.

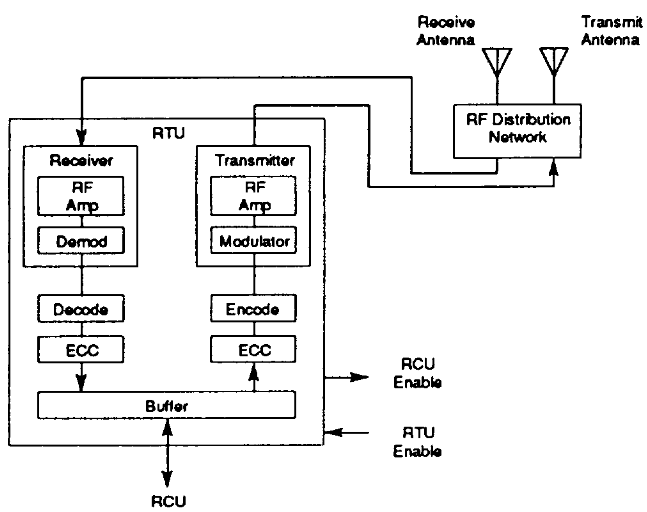

Fig. 10 Remote Tracking Unit (RTU)

The RTU is composed of a receiver, transmitter, coding section, and antennas. Pulse code modulation (PCM) is employed by the transmitter and receiver. Two antenna pairs are connected and attached on the anti-Earth and Earth-facing sides. The anti-Earth antenna pair is used primarily for during spacecraft deployment. The Earth-facing pair is exposed once the payload antenna is deployed. Microstrip antennae arrays are used. Figure 10 shows the block diagram for the RTU.

The RCU consists of two independently addressable microprocessors along with associated memory and timing circuits. When command data is received via uplink, those commands not requiring immediate action are stored in a command buffer within each RCU. Each RCU has memory set aside for the storage of commands to be executed at a specified time. Commands are retrieved and executed by the active RCU from its command buffer until it either receives a command that requires immediate action, or it receives a new list of spacecraft commands. Health and performance data are collected by the RCUs by periodically polling the spacecraft systems. Data is then stored in RAM until it can be down-linked. Figure 11 shows the block diagram for the RCU.

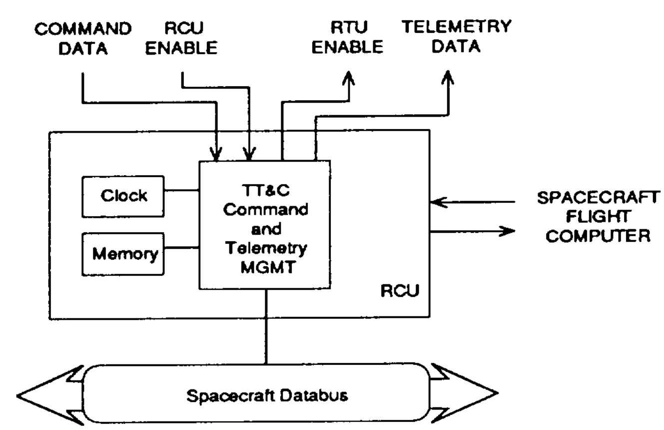

Fig. 11 Remote Control Unit (RCU)

Electric Power Subsystem (EPS)

The power requirements for the spacecraft are driven by the potential use of the communication payload in the global environment. This equates to the global distribution of potential users and the orbital coverage provided. The following factors were considered.

- 90% of the global population is in the Northern Hemisphere.
- China, India, and the former USSR have the largest populations, but are not expected to provide a significant market due to economic limitations.
- The United States, the fourth largest country, has a population of 250 million with a distribution of 42% in the Boston-New York-Washington, D.C. triangle; 10% on the West Coast; and 10% in the Midwest region.
- 80% of Canada's population (27 million) lives within 100 miles of the U.S. border.
- Western Europe is expected to provide the bulk of the European customer market for the remainder of this century due to economic failures in Eastern Europe; particularly large are France (pop. 58 million) and "West" Germany (pop. 61 million).
- Japan (pop. 122 million) is considered the most significant Asian customer region.

Payload power usage was divided over geographic market regions by primary power demand (U.S., Western Europe, and Japan); secondary power demand (S. Korea, Hong Kong, U.A.E., etc.); and tertiary power demand (no significant marketing potential). Peak payload power is expected over the Northeastern U.S. seaboard at 1800 local time. This loading will experience seasonal variation, but is not expected to exceed 800 watts for 16 minutes. Spacecraft housekeeping requirements are 200 watts, meaning that the EPS must provide 850 watts for 20 minutes. The EPS sizing is based on this 255 watt-hour requirement (illuminated or in eclipse). The orbital pattern repeats a similar ground track once every 72 hours.

The EPS design consists of two deployable solar arrays with solar array drive assemblies (SADAs), a single nickel-hydrogen (NiH_2) battery, and the power control unit. The EPS provides a fully regulated dual power bus at 28V. The solar cells are 2 cm x 4 cm x 200 micron, 10 Ω-cm base resistivity, with back surface reflector and back surface field. Each wing has three panels with dimensions 120 cm x 80 cm and 1 panel with 120 cm x 140 cm. Each wing has 40 series strings of 95 cells each. The solar arrays, once deployed, provide two degrees of freedom. Each SADA allows rotation in the panel roll axis for orbit longitude variation, and tilting of the deployment arm at the attachment base for inclination variation.

The NiH_2 battery is composed of 23 cells. Each cell is an individual pressure vessel housed in an insulated aluminum sleeve. The five-year design life requires 24,000 charge/discharge cycles. Space-rated NiH_2 cells are considered safe for this cycle life to a nominal 40% depth-of-discharge. A majority of the discharges are expected to be in the 30% range. This provides considerable power margin and avoids any memory effect from constant cycle levels.

The mass budget for the EPS is given in the following table.

Table 19 EPS Component Summary

Component	Mass (kg)
Battery module	30.0
Solar array subassembly	16.0/wing
Wiring harness	9.6
Deployment mechanism	6.0/wing
Array drive assembly	16.0
Array drive electronics	2.5
Power control unit	5.6
Shunt resistor bank	2.23

Propulsion Subsystem

The propulsion subsystem provides for maneuvers required after dispension from the SLD, including despin, orientation changes, deceleration, and orbit maintenance. Orbit accuracy is set at 1° from the intended position. The propulsion subsystem also provides for deorbit at the end-of-life of the spacecraft. The system consists of six 2.2-N hydrazine thrusters, a 49-cm diameter propellant tank, pressurant tank, and plumbing. The mass summary of the propulsion subsystem is given in Table 20.

Table 20 Propulsion mass summary

Maneuver	ΔV (m/s)	Isp (sec)	Change (kg)	Final (kg)
Launch mass				374.00
Attit. orientation	1.00	224.00	.17	373.00
Orbit injection	149.39	223.96	24.57	349.26
EOL deorbit	146.00	218.23	22.93	324.84
Propellant holdup			0.98	323.84
Margin			57.17	
Component			Mass (kg)	
Tank, pressurant, propellant			70.0	
Thrusters (ea)			0.344	
Piping			4.5	

Figure 12 shows the location of the thrusters on the spacecraft. The operation of the thrusters for injection, station-keeping, and orientation maneuvers are given in Table 21.

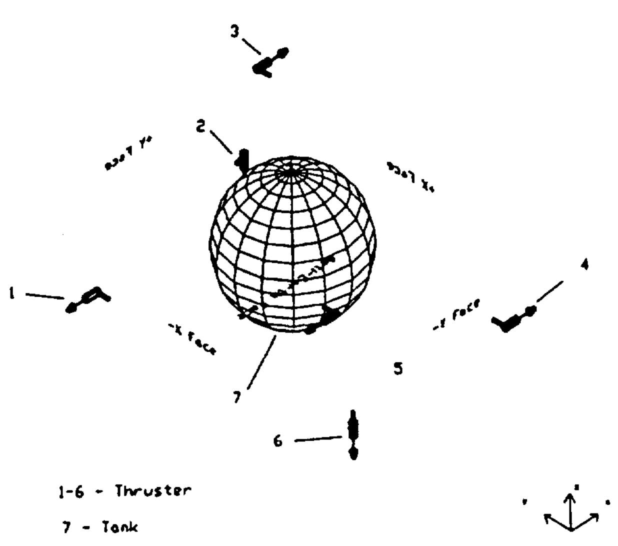

Fig. 12 Thruster locations

Table 21 Thruster operation

Operation	Thruster(s)
Injection	5 & 6
Station-keeping	(1 & 2) or (5 & 6)
Positive/negative roll (+x/-x)	5/6
Positive/negative pitch (+y/-y)	1, 4, or 6/ 2, 3, or 5
Positive/negative yaw (+z/-z)	1 or 3/ 2 or 4

Attitude Control Subsystem (ACS)

The ACS provides autonomous attitude control through the use of a bias-momentum wheel to control pitch and damp disturbances in roll and yaw. Magnetic torque rods are used for roll control and momentum wheel desaturation. The magnetic torque rod used to control roll provides a 50 amp-m^2 magnetic dipole and is offset by an angle of 71°. A single torque rod can counter errors in both roll and yaw. The torque rod used to desaturate the momentum wheel also provides a 50 amp-m^2 magnetic dipole. Attitude sensors include a solar aspect sensor for initial orientation, two-axis scanning Earth horizon sensor (EHS) for nadir angle determination and a two-axis magnetometer to measure magnetic field strength. An attitude control computer provides onboard processing.

The ACS functions include initial orientation establishment, countering torques from spacecraft motors (i.e., solar array drive), and damping internal and external disturbance torques. Initial orientation establishment is done when the spacecraft is first released from the SLD where an initial angular velocity is imparted. Thrusters are fired for despin following solar array deployment. The momentum wheel then starts spinning to provide gyroscopic stability. The spacecraft slowly begins rotating about its major axis to acquire the sun. If after five revolutions it does not acquire the sun, it begins rotating about the roll axis. The spacecraft then rotates about its y-axis in order to locate the Earth. The satellite then maintains its attitude to within ±1° accuracy.

The ACS design considered solar, gravity gradient, magnetic, and internal torques. The magnetic disturbance torque is the primary force acting on the spacecraft. An analysis was done using the MATHCAD software application to find the time constants and gains of the ACS required to overcome the magnetic disturbance torque.

Thermal Control

Thermal control of the spacecraft is intended to dissipate heat generated by the communication equipment and maintain temperatures of the components within operational limits. Thermal louvers, radiators, cold plates, and phase changers encompass the pseudo-active thermal control of the spacecraft. A stacked thermal package design is proposed as the solution. The stacked thermal package is shown in Figure 13. Four of these packages (two per side) comprise the backbone of the thermal control system.

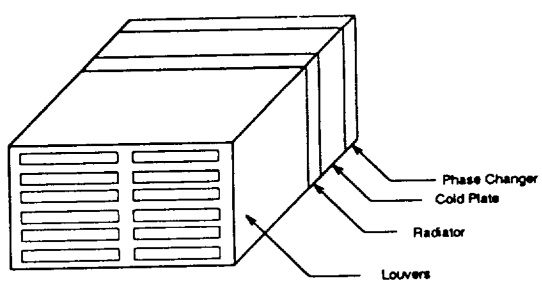

Fig. 13 Stacked thermal control package

Equipment to be cooled bolts to a cold plate through holes in the phase changer. The phase changers themselves are not strong enough to be weight-bearing. Heat dissipates mostly in the phase changer, melting the parafin inside. The cold plate spreads the heat evenly across its surface, where conduction to the optical surface radiators (OSRs) takes place. The heat is then free to radiate to space. The efficiency of the radiative transfer is determined by the opening and closing of the thermal louvers. Each blade of the thermal louver is controlled by a separate bimetallic actuator.

Heaters are required on the fuel tank, the six propellant lines, and the battery elements. Insulation surrounding the batteries, propellant tank, propellant lines and various other system equipment is necessary to reduce the power required to keep them within temperature limits. Thermal blankets and paint will provide a passive means of thermal control. Thermal blankets and thermal insulation are composed of multi-layered kapton separated by a thin nylon mesh.

A detailed thermal analysis of the spacecraft was performed using the PC-ITAS Version 7.3 software application by ANALYTIX Corp. All heat-generating pieces of equipment are included in the model. The results yield node temperatures for steady-state and transient cases. Figure 14 shows the thermal model developed using the PC-ITAS program.

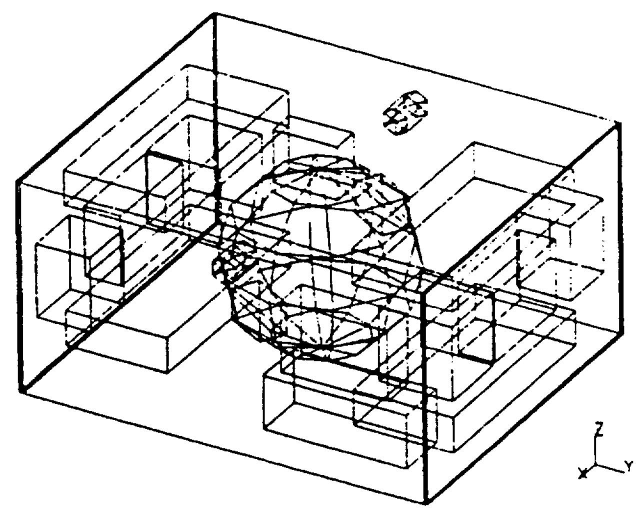

Fig. 14 Thermal analysis model

Spacecraft Structure

The spacecraft structure consists of a frame of rectangular tubing with panels attached to all sides. The ±pitch axis panels provide structural support for the majority of the equipment and thermal control system. The ±roll axis panels support the minor heat generating equipment and the solar arrays. The Earth panels are designed primarily to provide a mounting surface for the phased array antenna. The anti-Earth panel is used as an access panel for construction and integration and as an attachment face for the second TT&C antenna.

The rectangular frame consists of two sizes of rectangular cross-sectional tubing. The lateral tubing has 2.0 in x 1.5 in outside dimensions. The thickness is 1/8-in; the material, 6061-T6 aluminum. The longer dimension is aligned with the Z-axis to absorb loads, to maximize the area moment of inertia, and to minimize deflections. Axial frame members consist of 1/8-in rectangular cross-section 6061-T6 aluminum with 1.5 in x 1 in outside dimensions. The shorter dimension is aligned with the X-axis.

Equipment panels are made of 6061-T6 aluminum honeycomb with core thickness of 25.4 mm and face thickness of 0.2 mm. The honeycomb equipment panels are designed to support 54 kg of component mass under 36 g's of dynamic loading with a fundamental frequency above 30 Hz.

A finite element model was developed and analyzed for the Delta II launch loads in the X, Y, and Z directions. The launch loads and results are given in Table 22.

Table 22 Delta II launch loads and margins of safety

S/C Dir.	g's Accel.	Least M.S.
Yaw axis	8.5	12.0
Roll axis	15.0	1.1
Pitch axis	15.0	76.0

The finite element analysis results show positive margins of safety for all load conditions with a minimum of 1.1 for the load case corresponding to acceleration in the roll axis. The dynamic frequency of the model, however, shows lower mode frequencies than that required for the Delta II launch. The fundamental frequency of the structure from the finite element model is 19.18 Hz. Due to time constraints, this problem could not be fixed. Figure 15 shows the finite element model.

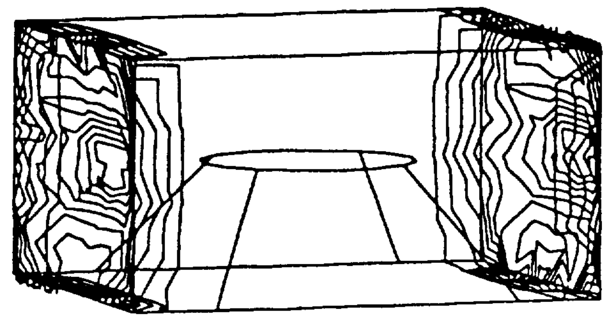

Fig. 15 Structural finite element model

Spacecraft Testing

Testing of the spacecraft is involved in each phase of the program. Testing is done in the development, qualification, acceptance, prelaunch, and on-orbit phases of the spacecraft life. The test plan is designed to perform development testing of qualification and prototype hardware and computer program design concepts as early as possible to ensure early detection and resolution of design, fabrication, compatibility, performance, reliability, and life expectancy problems. The test plan outlines acceptance tests on all units before installation on the spacecraft and prelaunch validation tests to demonstrate that the space vehicle and the launch vehicle have been successfully integrated. Once in orbit, full operational capability is tested. The general test plan is shown in Figure 16. Descriptions of the various tests are given in the final report.

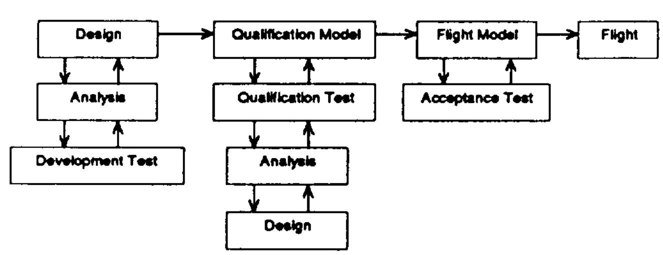

Fig. 16 General test plan

Cost Analysis

Various methods of cost analysis were employed. Bottom-up costing was utilized where component costs were available. Some components were not individually priced and were grouped into categories based on their function. Costs for such components were formulated by use of Cost Equivalent Relationships (CERs).[4,5] CERs are parametric equations based on historical cost data and rationalized to cost versus mass of the subsystem. Therefore, even though components may have been identified, the parametric equations required that they be lumped together to form one total mass. Costs which were not currently available for space flight, such as the payload antenna, overall structure, and the SLD, reflect the additional cost of RDT&E amortized over the entire buy.

Costs for testing the satellites were formulated based on specific costs to test the entire satellite separately and as a stacked unit. Component level costs were not formulated since most components chosen were already flight-qualified. Costs of testing the payload antenna are included in the costs of system testing. These costs were formulated by use of CERs developed for testing. The total cost of testing was then amortized over the 48 satellites and applied to the first unit cost.

The theoretical first unit cost (TFUC), including the amortized system test costs, launch and orbit insurance (16% of the total cost of the program) is $ 271.728 million. The total cost of the procurement of the 48 satellites is $753.231 million. This reflects a learning curve of 95% applied over the entire buy.

Revenue generated from this program is broken down by year for RDSS service as shown in Table 23. This estimate is based on achieving a 40% share of the U.S. market by the year 2006. The incorporation of international markets is expected to increase this revenue base significantly. It is interesting to note, however, that viability of the system is achieved by accounting for only the U.S. market.

Table 23 Revenue from RDSS service

Year	RDSS Customers ($x10^3$)	Total annual revenue ($\$x10^6$)
1997	212	64.3
1998	271	76.3
1999	338	88.9
2000	412	102.1
2001	416	98.2
2002	419	94.3
2003	422	94.9
2004	424	95.5
2005	427	96.0
2006	441	99.2

Conclusion

The preliminary design of a low Earth orbiting communication satellite is presented for global communication. Forty-eight satellites distributed over six orbital planes, evenly spaced within the plane, constitute the space segment. The ground segment consists of at least two control stations and multiple gateway stations for spacecraft control and connection to land-based communication systems, respectively. The spacecraft are launched six at a time via a Delta II launch vehicle. This preliminary design shows the viability of the system. The structure requires further study due to the low frequency of its fundamental mode.

Acknowledgments

The design team would like to thank Prof. Brij Agrawal for his guidance. The team would also like to thank Dr. Edward Euler and Mr. Dan Sakoda of NPS. Mr. Tony Navarra and Mr. Bob Wiedeman of Loral Aerospace Corp. provided the FCC Licensing Request and were very helpful. The Lockheed FSAT Project Office is due our sincere appreciation; in particular, Mr. Larry Van Erdun, Mr. Bill Rust, Mr. Marty Gerbasi, Mr. Larry Tong, Mr. Jim Kim, Mr. Bruce Simpson, and Mr. Kevin Vogler deserve special recognition. Finally, we appreciate the comments and interest expressed by Dr. Kim Aaron of the Jet Propulsion Laboratory.

References

1. Moroney, D. et al. Spacecraft Design Project: Low Earth Orbit Communication Satellite, Naval Postgraduate School, December 1991.

2. GLOBALSTAR SYSTEM APPLICATION before the Federal Communications Commission, Loral Cellular Systems Corporation, June 3 1991.

3. Delta II Commercial Spacecraft Users Manual, McDonnell Douglas Astronautics Company, Huntington Beach, CA, July 1987.

4. Werts, J.R. Space Mission Analysis and Design, Kluwer Academic Publishers, 1991.

5. Elbert, B.R. Introduction to Satellite Communication, Artech House, 1987.