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Systems Engineering Trades for the IRIDIUM® Constellation

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The customer imperative has driven the system design for the IRIDIUM® constellation and ground support infrastructure. Although the goal of the system is to provide global, mobile telephone service, the two principal drivers for the trade studies are 16-dB link margin and commercial success. The systems engineering and constellation architecture trade studies that determined the design of the IRIDIUM telecommunications system were comprehensive and are continuously evaluated. The complexity of the total system drove the engineers to critical evaluations of major factors such as complexity of payload, size/weight of spacecraft, altitude of constellation, size of individual beams, space environment (radiation belts), size of telephone (pocket size), quality of voice/data, number of telephone calls, type of commercial launch vehicles, number of satellites, orbits and inclination of orbits, number of gateways, crosslink types, and type of modulation. The complexity of the decisions and how they trade off to one of the two basic principles of customer satisfaction, 16-dB link margin and commercial success, is discussed.

Introduction

THE IRIDIUM® satellite system consists of an interconnected network of 66 satellites (plus 6 spares) in low Earth orbit (LEO) plus the ground-based infrastructure required to command and control the satellites and connect them to the public switched telephone network (PSTN). Figure 1 shows the system architecture with the rf connections between satellites and the Earth. The satellites orbit the Earth at an altitude of 421 n miles. Their orbits are inclined at an angle of 86.4 deg with respect to the equator, providing global Earth coverage. The constellation is configured as six planes with 11 operational satellites and 1 spare satellite in each plane. The satellites are equally spaced and arranged so that their coverage footprints overlap to provide continuous coverage by handing communications traffic from satellite to satellite as they pass over the Earth. The spare satellites are flown at a slightly lower altitude (350 n miles) until they are needed to replace worn-out operational satellites, at which point the depleted satellite is deorbited and the spare maneuvered into the appropriate slot in the constellation.

The satellites are launched using a combination of three different launch vehicles. The U.S. Delta II rocket is used to launch five satellites at a time, the Russian Proton rocket launches seven at a time, and the Chinese Long March IIC/SD launches two satellites at a time. Each satellite weighs approximately 1460 lb fully fueled. Once in orbit, two large rigid solar array panels are deployed to provide electrical power, and three main mission antenna panels are deployed to enable the L-band communications traffic.

A significant feature of the IRIDIUM satellite system is the K-band rf crosslinks used to provide connectivity between the satellites that make up the constellation. Each satellite contains four crosslink antennas. Two are fixed and communicate with the satellite's nearest neighbors in the same plane: the satellite immediately ahead and the one directly behind. The north–south in-plane crosslinks are maintained continuously. Two additional crosslink antennas are gimbaled and communicate with the nearest neighbor satellites in each of the two adjacent planes. As adjacent planes converge near the poles, the east–west crosslinks are disconnected and then reestablished when the planes diverge sufficiently. The crosslinks carry communications traffic as well as satellite command and telemetry data.

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Satellite Physical Description

The satellite vehicle (SV) is roughly 14 ft tall, with a triangular cross section about 3 ft along each side. This configuration was selected to allow manifesting multiple SVs on a single launch vehicle. In orbit, the SV is three-axes stabilized, with its long axis maintained in an Earth-pointing orientation. Figure 2 shows the SV in its deployed configuration. The spacecraft mass budget is shown in Table 1. The SV is composed of three main assemblies: the space bus module, the communications module, and the main mission antennas. In the following sections many aspects of the space system design will be discussed, and in so doing, the complexity level and some of the unique results from the system trades will be illustrated.

The communications section (CS) consists of equipment to support the communications mission of the SV. This includes a suite of converters, tuners, switches, and modems, which interface with the main mission antenna (MMA) to provide the subscriber link telephony services. Additional equipment is included to operate four SV–SV crosslinks, four feeder links for communications with gateway and control segment Earth terminals, and a secondary link for backup command and telemetry interfaces. Seven SV computers with supporting memory and interface equipment provide the data processing and routing control functions needed to support communications routing and resource management, antenna and link control, and SV flight operations. A functional block diagram of the IRIDIUM flight system shown in Fig. 3 illustrates the extent that the satellite is under software control.

The MMA consists of three deployable phased array antennas, which provide the rf links to the subscriber equipment. Each antenna panel utilizes an array of rf patches controlled by an array panel controller, which is in turn controlled by the CS. A beamformer network distributes the modulated carriers received from the CS and forms the 48 beams, which are transmitted to the ground.

The satellite bus module is capable of supporting the mass, volume, and heat dissipation requirements of the satellite and will be capable of providing the required support and rigidity during all phases of launch, transfer, and orbit insertion. The SV is made from graphite composite material and supports the subsystems described next.

Thermal Control System

The thermal control system (TCS) maintains a controlled thermal environment for SV subsystems and equipment through a combination of passive and active controls. Active thermal control is performed using heaters and heat pipes. Passive thermal control employs surface finishes, paints, and radiators, as well as thermal blankets to control temperatures. A battery radiator assembly is used to dissipate heat generated in the batteries during charging. Battery charge control is performed by the bus control software, which runs within the space vehicle computer (SVC).

Electrical Power Subsystem

The electrical power subsystem (EPS) provides the power to the satellite electrical loads over the expected lifetime. The bus voltage varies from 22 to 36 V and is converted to required equipment voltages by power converters located at the loads. The EPS is configured as a series regulated solar array system with the battery always on line. The subsystem includes the following major elements: high-output photovoltaic solar arrays, long-life nickel hydrogen battery, and software configurable battery charging controls. The EPS contains fault protection features, which automatically respond to subsystem malfunctions or to excessive main bus loads. All power subsystem automatic functions can be overridden in response to ground commands. Extensive telemetry is provided to allow a comprehensive evaluation of EPS in flight performance, as well as complete equipment status. The solar array consists of two sun-oriented, two-axis gimbaled planar wings. During liftoff, the panels are folded against the spacecraft body. Following launch vehicle separation, the array is deployed and full power will be available.

The following mechanisms are used to deploy the solar array: multipoint solar panel support during launch utilizing paraffin actuated releases and solar arrays sequenced using two-axis stepper motor actuators. Stepper motor drives are used for sun tracking. Power from the rotating array is transferred to the main bus by slip rings. Each wing is independently oriented by stepper motors.

The design incorporates high efficiency GaAs/Ge solar cells on a lightweight graphite fiber reinforced plastic substrate for minimum size and weight. Array design and sizing will meet all power requirements over the satellite mission life.

One 60 A-h nickel hydrogen, single pressure vessel battery comprises the battery system providing primary power during eclipses and during peak loading. Battery charge current will be controlled by switching array strings based on charge control laws processed within the onboard computer, with temperature backup controls within the power control unit. The battery consists of 22 cells. Battery charge control is performed by the bus control software, which runs within the SVC. Table 2 sets forth the electrical power budget and indicates end-of-life (EOL) power capabilities.

Table 1 Spacecraft mass budget

Subsystem	Mass, lb
Structure	265
TCS	22
Propulsion subsystem (dry)	31
Guidance, navigation, and control subsystem	41
Electrical power subsystem	230
Antenna subsystem	221
Communications electronics subsystem	349
Spacecraft mass (dry)	1159
Consumables	253
Spacecraft mass (wet)	1412
Spacecraft reserve mass	48
Spacecraft wet mass with reserve	1460

Attitude and Orbit Control System

The attitude and orbit control system (AOCS) maintains the satellite pointing and stability and provides the propulsive capability to support orbit transfer and station keeping. The AOCS is configured as a three-axis momentum bias system. Primary attitude control of the satellite is achieved by the gyroscopic stability of a single pitch axis momentum wheel assembly (MWA). Primary nadir attitude reference is determined from a fine horizon sensor assembly (FHSA), which provides pitch and roll errors via infrared detection of the Earth's horizon. Pitch axis errors are removed by commanding changes in wheel speed, which results in an exchange of momentum with the satellite. Both roll and yaw errors are corrected in response to a roll error signal due to the gyrocompassing effect from the MWA. The roll and yaw errors are both controlled by commanding torques from two magnetic torque rods along the roll and pitch axes. Torquing results from the interaction between the magnetic fields of the torque rod coils and the Earth's magnetic field. The direction of the Earth's magnetic field is determined by a three-axis magnetometer. Momentum dumping of the MWA is also accomplished using magnetic torquing.

The AOCS utilizes additional hardware for various phases of operation. A three-axis gyro assembly is used for attitude reference after separation from the booster. A coarse horizon sensor assembly, which has a much wider field of view than the FHSA, is used

Table 2 Average power budget, 24 h

Subsystem	EOL, W
Communication electronics	252
Phased array	280
SV housekeeping	88
Total load	620
Instantaneous peak power	>4000

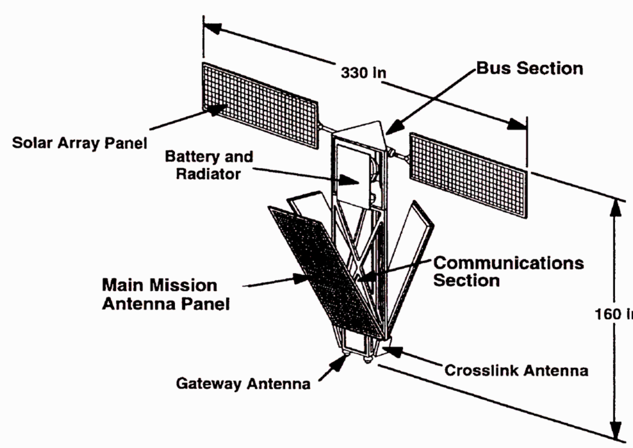

Fig. 2 Deployed IRIDIUM space vehicle.

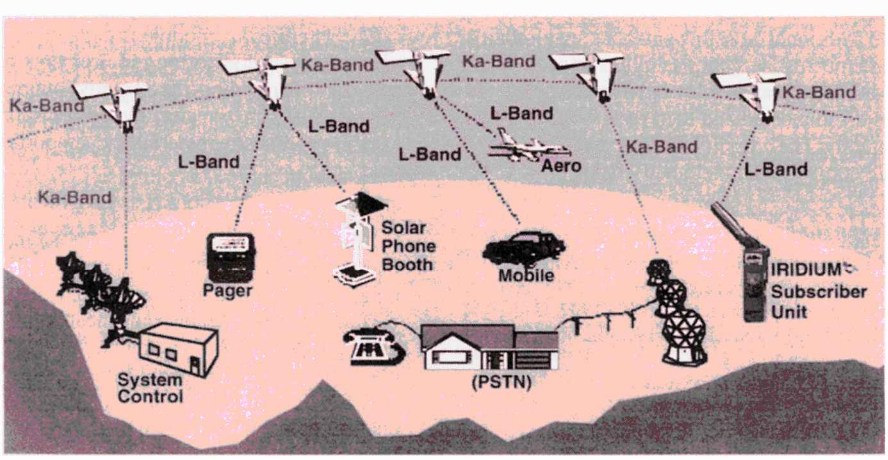

Fig. 1 IRIDIUM system architecture.

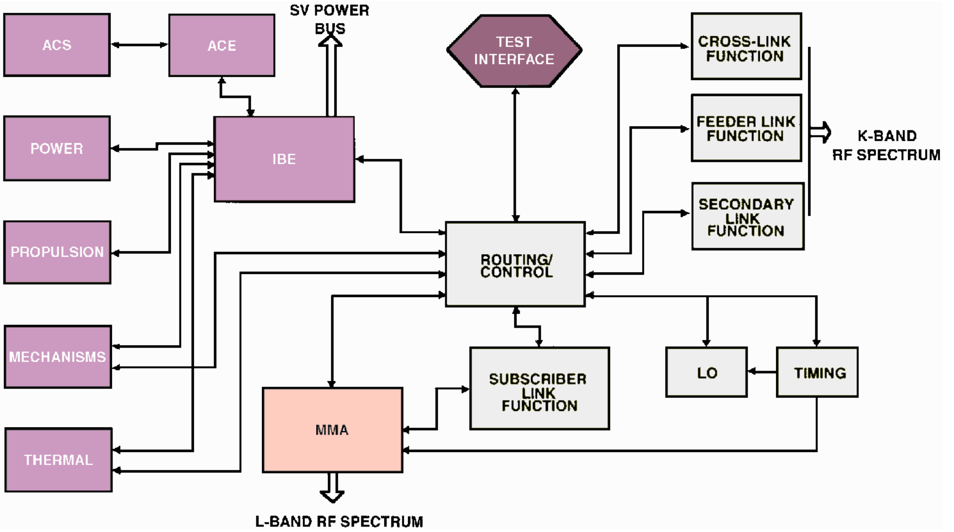

Fig. 3 IRIDIUM flight system functional block diagram.

for attitude reference during initial Earth acquisition, storage orbit, ascent, and deboost. Seven hydrazine reaction engine assemblies (REAs) are used for attitude control during separation Earth capture, MWA spin up, ascent, and deboost. An electrothermal hydrazine thruster (EHT), which has a high specific impulse, is used for ascent and deboost. The REAs are also used for station keeping and as backup to the EHT for ascent/deboost.

Ascent, station keeping, and deboost can be performed autonomously or in conjunction with ground commands. The approach uses a method wherein each satellite is maintained within a prescribed box with position within the box determined by the AOCS. The ephemeris of each satellite and its neighbors are propagated onboard and are updated by the ground approximately once a week. The positions and velocities of neighboring satellites are used for crosslink antenna pointing.

The system is fully autonomous with ground control backup capability after the satellite is placed in its final orbit and initially attitude stabilized. Position and attitude are available at 90-ms intervals with the following accuracy: attitude (+/− deg 3 sigma): 0.2 roll, 0.3 pitch, 0.4 yaw and position (+/− km 3 sigma): 6 in track, 5 cross track.

Propulsion Subsystem

The satellite uses a blowdown monopropellant hydrazine subsystem to provide all propulsive functions. This subsystem applies external torques and forces to the satellite to perform the functions of orbit insertion, orbit adjustment, maintenance, reaction control, and deorbit. The principle components of the subsystem include a propellant storage pressure vessel, an EHT for major delta-velocity maneuvers, standard hydrazine REAs for attitude control and station keeping, latching torque motor and manual valves, plumbing, and telemetry instrumentation to evaluate the in-flight performance of the subsystem.

The propulsion subsystem includes a propellant load and thrusters capable of providing the delta-velocity increment required to insert the spacecraft into the designated orbital slot and effectively deorbit the spacecraft at EOL. The on-orbit functions of the subsystem include altitude, mean longitude, and orbit inclination maintenance for the duration of the operational life. The on-orbit propellant load has been sized to provide a 60% consumable reserve for operational orbit maintenance functions (8-year operational orbit capability).

The hardware implementation of the subsystem utilizes one surface tension propellant tank manifolded to the thrusters, each having a series redundant propellant control valve and its own thermal elements. Thrusters are located to provide limited redundancy. The surface-tension device in the fuel tank ensures gas-free propellant delivery under all acceleration environments. Propellant and pressurant are loaded into the tank through a pair of manual valves, each with triple-redundant seals. A single latching torque motor valve prevents propellant from flowing into the delivery manifold until commanded open after separation from the launch vehicle. An in-line filter rated at 10 μm absolute is located downstream from the propellant tank outlet to prevent any fine contaminants from reaching the thrusters. A pressure transducer and thermistors located on the propellant tank and lines provides health and status of the propulsion subsystem.

When the latching valve is opened after launch, the pressurant gas in the propellant tank forces the propellant through the plumbing to the thruster inlets. Propellant feed pressure slowly decays as propellant is consumed (blowdown operation) and the volume occupied by the pressurant gas in the propellant tank expands.

Because of the low thrust of the engines, large orbit adjust maneuvers such as the initial orbit raising and the deorbit maneuver are accomplished with numerous long-duration EHT firings. The REAs are operated in pulse mode for attitude control during the EHT firings. Operational orbit maintenance is performed with short pulses on an REA, principally for drag makeup.

Each thruster valve is controlled by a drive circuit in the integrated bus electronics (IBE), which provides the interface between the propulsion subsystem and the main payload processor. A background task running on the main processor selects the proper valves and timing for whatever function is commanded from the ground or by the onboard attitude control task. The status of the subsystem and maneuver bookkeeping is monitored via telemetry.

Telemetry, Tracking, and Control Subsystem

The telemetry, tracking, and control (TT&C) subsystem provides the functional hardware required for the reception, processing, and implementation of command data, and the collection, storing, multiplexing, and transmission of satellite telemetry data. The TT&C subsystem operates with or without the simultaneous functioning of the mission communications system and does not degrade or interfere with the mission communications operations. The spacecraft antenna arrangement and the communications hardware configuration ensure that command and telemetry functions are accessible during all phases of the mission. Typical functions of the TT&C subsystem are satellite electronic equipment redundancy switching, multiple-point system monitoring for health and status evaluation, control of system initialization and testing, and general housekeeping tasks.

When the satellite is serving as a node in the network, telemetry downlink transmissions will utilize the 19.4–19.6 GHz band, and telecommand uplink transmissions will utilize the 29.1–29.3 GHz band. TT&C data are multiplexed into the wideband 20/30-GHz system control facility ground link (using crosslinks as necessary for satellite-to-satellite relay). During transfer and sparing orbit operations, the TT&C capability is provided by transmission over narrow-band channels in the frequency bands just noted. These narrow-band channels use circularly polarized omnidirectional spacecraft antennas to permit direct communications with the system control facility

Table 3 TT&C satellite transmission characteristics

	Nonnetworked	Networked
Frequency band	19.4–19.6 GHz	19.4–19.6 GHz
Antenna polarization	LHCP	LHCP (feeder link)
Modulation	BPSK/PM	QPSK[a]
Data rate	1 kbps	3.125/12.5 Mbps (total feeder link/ crosslink data rate)

[a]Quadrature phase-shift keying.

Table 4 Command signal characteristics

	Nonnetworked	Networked
Frequency band	29.1–29.3 GHz	29.1–29.3 GHz
Antenna polarization	RHCP	RHCP
Modulation	BPSK/PM	QPSK[a]
Data rate	1 kbps	3.125 Mbps (total uplink data rate)

[a]Quadrature phase-shift keying.

regardless of the satellite attitude. This link supports control and data communication during postseparation maneuvers, orbit raising and lowering procedures, and other times when the satellite is not in the network. The narrow-band TT&C communication link signal margins are adequate to permit a ground station antenna of moderate beamwidth (approximately 1 deg) to receive and transmit TT&C link signals. This meets the requirements of initial acquisition and reacquisition operations. When the narrow-band transmission mode is in operation, all of the TT&C data and control signals are transmitted in digital form at a rate of 1.0 kbps each. Large signal margins and binary phase-shift keying/pulse modulation (BPSK/PM) are used to minimize the effects of anticipated antenna pattern irregularities and grating lobes, which are characteristic of the satellite omnidirectional antennas. Some of the attributes of the telemetry downlink are summarized in Table 3.

The command subsystem is designed to maintain positive control of the spacecraft during all mission phases. It provides reliable control during launching maneuvers and for all satellite operating attitudes. It also maintains the orbital velocity of the satellite and controls housekeeping functions and communications subsystem configurations. The command messages are authenticated to provide security, protecting the satellite control subsystem against unauthorized access.

The command transmissions received from the ground are demodulated into a digital bit stream. When the destination satellite receives a command from the TT&C Earth control station, the command is authenticated and, if successful, executed at the specified time. The ground can then verify its execution based on telemetry. Some of the attributes of the command uplink are summarized in Table 4.

Operational Lifetime and Space Segment Reliability

The operational lifetime and reliability of each satellite is determined by a number of factors, including solar array degradation, station keeping fuel consumption, battery degradation, and random parts failure. Redundancy is provided on critical hardware as determined to be necessary through reliability analyses and predictions to achieve a 0.58 probability of success for a 5-year mission.

System Trades

The systems engineering and constellation architecture trade studies that determined the design of the IRIDIUM telecommunications system were comprehensive and are continuously evaluated.

Complexity of Payload

The complexity of the payload in the IRIDIUM SV is a result of the need to minimize the processing burden placed on central ground stations and to provide the greatest flexibility (and hence the greatest service) in the routing of communications traffic. Onboard processing can route voice and data in a dynamic environment without the overhead of continual ground intervention. Delays are minimized for added voice quality. Much of the complexity is provided by state-of-the-art, application-specific integrated circuits and real-time software, which reduces the hardware complexity.

Much of this complexity is allowed because the design philosophy provides redundancy at the system level instead of the hardware configuration item level. Autonomous operation and dynamic resource management and routing provide constellation failure mitigation. In effect the traditional hardware redundancy is spread over many spacecraft.

Size/Weight of SV

The primary driver in sizing the SV is the payload. Whereas communications electronics can be packaged in many forms, the antenna size (and technology) determines the form factor, along with the consideration of packing multiple spacecraft into existing launch vehicle shroud envelopes. Payload electronic demand determines the size of energy generation hardware (solar arrays) and energy storage hardware (batteries).

Once the overall target of 16-dB L-band link margin was validated, the selection of phased array technology moved the IRIDIUM SV design from a six-sided form to the existing triangular form, leading to ease of deployment of the L-band panels and keeping a high vehicle packing density within the launch vehicle shrouds. Corresponding increases in electrical power needs are handled by taking advantage of the large spacecraft surface area exposed, allowing maximum solar array size without complex panel folding.

The aspect ratio of the design leads to the use of composite structures to provide the required stiffness for the launch environment. Composite technology keeps the weight below that which older metal technology would have permitted.

Type of Commercial Launch Vehicles

The major concern with the deployment of a satellite constellation with a significant number of nodes is the availability and reliability of sufficient launch system assets to guarantee the complete deployment within a short time frame. Among the technical considerations are the number of vehicles required (dependent on the size and number of payloads to be placed in orbit), the potential for adding custom attach and dispense hardware, and the overall reliability of the launch system. Economic and political considerations are extremely important if the number of required launches is high: availability of launch opportunities (manifest slots) and ability to overcome export barriers if nondomestic carriers are anticipated.

For a commercial venture, economic considerations all but eliminate the expensive, massive vehicles currently available for government and crewed operations. The selection of the Delta II, Long March IIC/SD and Proton launch systems recognizes the need to deploy significant numbers of satellites using affordable, commercially available assets that guarantee the successful deployment of the IRIDIUM constellation.

Size of Individual Beams

For a communications system utilizing space nodes, the most important design requirement is the rf link performance. Recognizing that the service provision for subscriber units located anywhere on the surface of the globe requires multiple beams (or cells), technical parameters such as overall gain requirements, maximum allowable gain variation, polarization, and uplink equipment performance must be determined.

Hardware tradeoffs must be made between available beam generation technology (phased array or waveguide/reflector implementations) and usable SV resources (area, volume, and mass distribution). Other factors involved are beam shape (hexagonal, elliptical, and complex), cooperative mechanisms with adjacent space nodes (means of transitioning communications services from one spacecraft to the next as the constellation moves with respect to the user), and, most unusually, ground coverage control (the need to limit dynamic geographical coverage to satisfy political and licensing restrictions).

Space Environment

The establishment of spacecraft constellation altitude is first established using communications parameters [beam coverage, system gain, and gain to temperature ratio (G/T)] but the unique radiation environment of space must be used at a minimum as a hardware design constraint and, possibly, as a major driver in system design. The primary concern is the use of affordable, commercial-off-the-shelf electronic components while delivering acceptable system availability and reliability performance. The IRIDIUM system altitude is significantly below the first major peak in charged particle environment, allowing state-of-the-art components to deliver the necessary communications performance. At higher altitudes, either the overall mass fraction of payload electronics would decline as more and more shielding would be necessary, or communications services would be degraded to accommodate older, more expensive, and less efficient semiconductor technologies within the payload electronics.

Size of Telephone (Handheld or Suitcase)

The size of the subscriber unit for voice communications is dictated by how much technology can be economically packaged into an acceptable form factor. Acceptability is determined by the user market, with familiarity and ease of use. The trades for the subscriber unit directly affect the physical and functional characteristics of the IRIDIUM SV. Technically, the size is driven by antenna capability and battery capability. However, the economies of scale of production influence what might seem an obvious outcome. Although space hardware demands premiums of reliability and compactness (smallest mass possible), the production of millions of hand-held units weighs highly in the choices of system design, balanced against 66 active space vehicles. And although the space vehicle function is far beyond the notice of the customer, the handheld subscriber unit is the ultimate interface for the system.

Quality of Voice/Data

Quality of voice and data communications is the ultimate driver of the IRIDIUM system. Extensive analyses and testing were undertaken to fully characterize the expected link performance of the system. The bit error rates in this digital system were established to provide adequate customer perceptions (for voice traffic) and acceptable throughput performance (for digital modes of communication via several protocols). Applying these error rates to the expected deep fading environments led to the system requirement of 16-dB L-band link margin over theoretical minimums. Delays expected due to spacecraft altitude and packet routing were shown to be acceptable.

Following the development of the derived link specifications, total system performance was modeled to demonstrate the interaction of all parameters and to demonstrate the lack of system self-interference and the ability to transport the required volume of traffic.

Number of Telephone Calls

The number of telephone calls supported by the IRIDIUM constellation is determined by market selection and spectrum capability. Once this target is established, all design requirements are then traded to provide adequate margin to this number while maintaining an economically viable system. Dependent design parameters include packet routing paths, spectrum reuse, and gateway partitioning. Independent considerations are the nature of human voice communications (activity factor), external interference environment, and market expectations of user profiles and revenue capacity.

For the spacecraft, the number of telephone calls supported is determined by processing load and power availability. The payload hardware and software are designed for the maximum peak load ever expected during the life of the constellation. The spacecraft power subsystem is designed to take advantage of the peak-to-average nature of the user distribution (most users are in the northern hemisphere land areas, with little nighttime traffic). By balancing the peak power capabilities with a multiorbit recharge scheme for energy balance, the mass of the spacecraft is minimal to support the traffic load.

Altitude of Constellation

The communications system impacts of constellation altitude are primarily rf physical considerations (system gain and G/T) and quality of service (delay). For a fixed payload electronics design, higher altitude means larger coverage areas, requiring more antenna beams of higher gains, lower link margins as interference environments grow, and possibly higher (therefore, more costly) performance requirements on the subscriber units.

Secondary considerations for altitude are economics of insertion (too high of an altitude means very costly launch services and/or high mass penalties), ease and speed of maintenance (replacement of older spacecraft from wearout or decayed orbits), and most importantly ease of deorbit. Altitude-dependent environmental factors (radiation, atomic oxygen, drag, and debris) directly affect reliability and availability. Eclipse duration directly affects the power subsystem design for energy storage.

Also important are collision probabilities as the quantity of other constellations of other spacecraft represent a growing threat to ongoing service. However, the IRIDIUM program mandates a zero debris policy and will deorbit its satellites, after life, to ensure a minimal probability of collision with any other satellite while in the operational constellation.

Number of Orbits and Number of Satellites

The number of orbits (orbital planes and vehicles per plane) in the IRIDIUM constellation is selected in concert with the beam size to optimize the coverage. At the equator, vehicle footprints must cover the greatest area, with the smallest amount of overlap. As the satellites move closer to the poles, overlap increases, and the constellation begins to shut down selected cells on individual spacecraft to minimize interference. This has the additional benefit of lowering overall spacecraft power demand. Too much overlap results in inefficient use of assets, whereas too little overlap leads to potential gaps in worldwide coverage.

Special consideration must be given to seam coverage (seam is defined as the area where adjacent spacecraft orbits are moving in opposite directions). This condition arises from using all spacecraft continuously (as opposed to only half the orbit, ascending or descending). Ground coverage demands different orbital spacing (east/west), adjustments to spacecraft hardware for different crosslink tracking rates, and accommodations in ground pattern handling within mission software.

An important parameter of the constellation definition is the cost of filling all orbital positions. Launch vehicle capacity, both for deployment of initial vehicles and for the maintenance as spacecraft are replaced, is extremely expensive. Sparing philosophy, reaction time for spacecraft failure mitigation, and desirability for partial constellation functioning as population rises must be weighed against the economic realities of limited flexibility in today's launch services market.

Inclination of Orbits

The IRIDIUM constellation contains satellites in circular, near-polar orbits. The exact selection of orbital inclination is based on the need to maximize the stability of the orbits and to maximize the closest approach distance as satellites near the poles.

Number of Gateways

The selection of the number of gateways handling the communications connections in the IRIDIUM system are chosen for economic reasons (control of revenue collecting operations), political reasons (licensing provisions, access to PSTNs, access control), and technical reasons.

The prime technical driver is system capacity. Fewer gateways mean that each gateway must service a greater portion of the calls made within the system. Too few and the required feeder links bandwidth increases, and traffic on intersatellite links will be overburdened (fewer satellites will be involved in the total feeder connection from constellation to the ground). Too many and more feeder links must be placed on the satellite to accommodate a greater number of simultaneous feeds, as more stations requiring service would be within view at any given time.

Although a uniform spacing of Earth stations would make system design easier, the gateway operations must be aligned with population concentrations and political boundaries.

Crosslink Types

The IRIDIUM system uses intersatellite links to provide communications services without added ground delays and bottlenecks. The selection of the physical link design is dependent on the communications architecture, interference environment, orbital geometries, and system reliability constraints.

Because of the dynamic pointing environment of crosslinks and spacecraft attitude, the use of optical crosslinks would require extensive development for accurate beam pointing, whereas standard rf links allow simpler, proven hardware implementations.

Type of Modulation

In a constellation of high inclination, circular orbits, the north/south separation between spacecraft remains constant, whereas the east/west separation is continually changing. The use of time-division multiple access architectures in the rf crosslinks, along with spectrum limitations, provides a simple half-duplex link design that meets all traffic requirements. This architecture also eases the problem of system self-interference, inasmuch as multiple spacecraft will be in view of crosslink antennas.

Prelaunch and Launch Operations

Prelaunch and launch operations are basically the same for all launch vehicle suppliers. The system control segment (SCS) will coordinate the overall launch plans and launch schedules with the launch segment. The launch vehicle is assembled and tested before mating with the satellite dispenser. The satellites are then mated with the dispenser and then with the launch vehicle. After all stages have been mated, the launch system will undergo final preflight integration testing, fueling, and checkout. The SCS provides the parameters that specify the desired parking orbit based on the constellation fill plan and schedule.

The SV constellation will be boosted to an initial separation orbit from three separate sites, which comprise the launch segment, in Russia, China, and the United States. Each launch scenario is similar, but the exact timeline, separation sequence, and initial orbital characteristics may differ for each launch vehicle. The SVs are powered off during launch with the exception of the IBE, which provides the necessary interfaces to detect the separation of the SVs from their dispenser. In general, the primary boosting stages deliver an SV dispenser (which includes a rocket motor and maneuvering system) to an initial parking orbit. The dispenser then maneuvers into a final separation orbit and adjusts its attitude for release of the SVs. Once released, all SV operations timelines for initialization and deployment activities are nearly identical.

The SCS will nominally acquire the satellite using predicted vector data received from the launch provider in the prelaunch time frame. The SCS can update the initial acquisition vectors at the telemetry, tracking, and control (TTAC) stations based on updated ephemeris information. This can be in the form of an injection vector or real-time feedback based on vehicle performance provided by the launch segment. In cases of nonnominal launch vehicle performance an antenna search pattern will be initiated at the TTAC to assist in SV acquisition.

Upon release from the launch vehicle dispenser, the SV senses that it has separated from the dispenser and applies power to the SV. After separation, the SV will autonomously boot up its computer and begin initialization of onboard electronics and systems components. The attitude control electronics will be powered on along with the components supporting initial attitude determination and propulsion system operations. Heaters (tank, line, and catalyst bed) are turned on in preparation for first use of the propulsion system and are allowed to warm up for a period of 15 min. The SV will use thrusters to three-axis stabilize the attitude rates and then acquire and maintain coarse Earth nadir pointing.

Mechanism deployment begins soon after completion of Earth acquisition activities. Both solar arrays are released and rotated in azimuth to allow clearance for MMA release and deployment. All three MMAs are then released simultaneously, after which the solar arrays are positioned in elevation to optimize power. The feeder link antennas are then released and uncaged. During the early orbit phase, the secondary link supports low-rate ground communications, tracking, and command activities. The secondary link will be initialized within the first orbit to support initial health and status checks postlaunch.

Prior to initiating an ascent or orbit raising maneuver, which is required to place the SVs in either the storage or mission orbit, a series of ground communications passes will be required to collect ranging data in support of orbit determination activities. During the course of ranging data collection, the SCS will assess the health and status of the SV.

The target orbit, storage or mission, is preprogrammed for each SV prior to launch but may be modified by the SCS prior to scheduling the ascent maneuver. Once an orbit solution is achieved, an ephemeris update will be uplinked to the SV. Upon receipt of the update, the SV will be commanded to generate an ascent maneuver plan, which is also sent to the SCS via telemetry. The SV will autonomously schedule the ascent maneuver based on the onboard plan. The ground may override the ascent maneuver should the need arise.

The SV will autonomously control the SV for attitude and delta velocity by initiating thruster firings based on the schedule derived from the maneuver plan, to both raise the orbit and adjust inclination as necessary. During the course of the ascent, the SV will require periodic updates of ephemeris data, requiring ranging and orbit determination activities. During the ascent phase the SV will maintain average three-axis attitude determination and control within the specifications.

Conclusion

These complex trades leading to the design of the IRIDIUM space system have led to a tremendously robust telecommunications network for bringing information to anyone, at anyplace. These trades were conducted over many years with detailed engineering analysis supporting the decisions. The systems engineering teams in each segment of the system, supported by the diverse teammates across the product teams, indeed looked at all options that seemed reasonable, even if nontraditional. The number of satellites, the LEO altitude, and the complexity of the payload seemed counterintuitive until the factors of voice quality and customer satisfaction were included in the analysis. The systems architecture of the IRIDIUM telephoning constellation emphasized the systems engineering trades from the very beginning of the project. In fact, the concept and preliminary trades were accomplished within the systems engineering department. The number and diversity of trade studies accomplished prior to the launch of the first satellite are remarkable. The IRIDIUM system will indeed support total customer satisfaction.

Acknowledgments

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Associate Editor