# OVERVIEW OF THE SCD1/PEGASUS MISSION

**Jaime Fernandez**
**Pegasus Mission Manager, Orbital Sciences Corp**
**and**
**Dr. Carlos Santana**
**SCD 1 Program Manager, INPE**

## I. Abstract

An overview of the third operational Pegasus mission, carrying the Brazilian Satelite de Coleta de Dados 1 (SCD1) satellite is presented. Developed by INPE, the 250 lb, spin stabilized satellite will perform real time repeating of environmental data gathered by automatic ground stations throughout Brazil. The target orbit, 405 nm (750 km) at 25 deg inclination, was chosen to provide coverage of the entire Brazilian territory. A review of the SCD1 design, development and testing highlights the effectiveness of lightsat philosophy complemented by pc-based check-out and control equipment. A six month schedule from go-ahead to launch was achieved in spite of delays. Simple and effective hardware and interfaces allowed a straightforward, efficient and relatively short payload integration and test process. Mission planning addressed complex operations, including a cross country ferry flight , ground operations significantly removed from the control room, and a first time east coast Pegasus launch from a new range. A review of the flight results includes flight environments and final Pegasus guidance and performance results

## II. Mission Overview

The third operational Pegasus flight, carrying the Brazilian Satelite de Coleta de Dados 1 (SCD1) satellite, occurred on February 9, 1993. Developed by INPE, the 250 lb, spin stabilized satellite will perform real time repeating of environmental data gathered by automatic ground stations throughout Brazil. The target orbit, 405 nm (750 km) at 25 deg inclination, was chosen to provide coverage of the entire Brazilian territory.

The Pegasus/SCD1 was ferried from the vehicle integration site at the Dryden Flight Research Facility to the Kennedy Space Center in Florida, a trip similar to the Space Shuttle ferry flight. The one day ferry flight occurred February 7th, and included a stop at Sheppard AFB Texas for refueling.

The Pegasus was launched from the B-52 carrier aircraft at an altitude of 42,000 ft at 09:30:34 Eastern Standard Time (EST) at 29° N, 79.88° W. The vehicle performed nominally throughout its 11 minute flight, placing the satellite in a 392 nm by 426 nm orbit at 24.97 deg inclination. Prior to payload separation, the Pegasus aligned the SCD1 satellite to the sun vector and spun up to 120 rpm. All payload mission requirements were met, and SCD1 transmitter frequencies were immediately received by the range.

## III. Satellite Description

The purpose of the SCDI mission is to provide direct relay of environmental data acquired by Data Collecting Platforms (DCPs) located in the coverage area of the Cuiaba' Earth station. This region corresponds to a circle of 3000 km radius centered in Cuiaba, limited by parallel 38 (see Figure 1) . The data of all DCPs located less than 1200 km from Cuiaba' are received at least seven times a day. The data of the DCPs located between 1200 and 3000 km from Cuiaba' are received less frequently, with a minimum limit of once a day for the DCPs located in the borders of the coverage area.

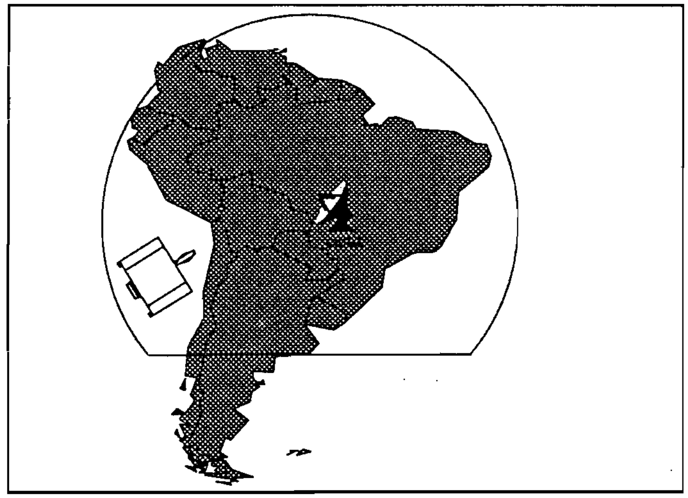


**Figure 1** Ciuaba Coverage Area

The DCP data are received by the Cuiaba' station, processed by the Mission Center in Cachocira Paulista and subsequently stored in a data bank which can be accessed by the users through bitnet, telex, PC connection, etc.

The SCD 1 is capable of operating simultaneously with up to 500 DCPs that follow the Argos system message format. Half those platforms can operate in the same transmission frequency as the Argos system, thus being able to be received either by the SCDI or by other satellites which offer this kind of service.

The SCD 1 satellite lifetime is expected to be at least two years. It will be followed by the SCD2 satellite which will continue the data collecting mission.

### Design Overview

The satellite flies in a nominally circular orbit inclined about 25 degrees with respect to the Earth's equatorial plane, at an average altitude of 750 km. The environmental data are collected by automatic DCPs that are located in remote, unattended locations. During the passes visible from the brazilian Cuiaba' tracking station, any DCP's within the coverage angle of the satellite antennas will have its UHF signal relayed by the satellite, in S-band.

The shape of the spacecraft mechanical architecture is a 80cm tall right octagonal prism whose base fits within a 1m diameter circle (see

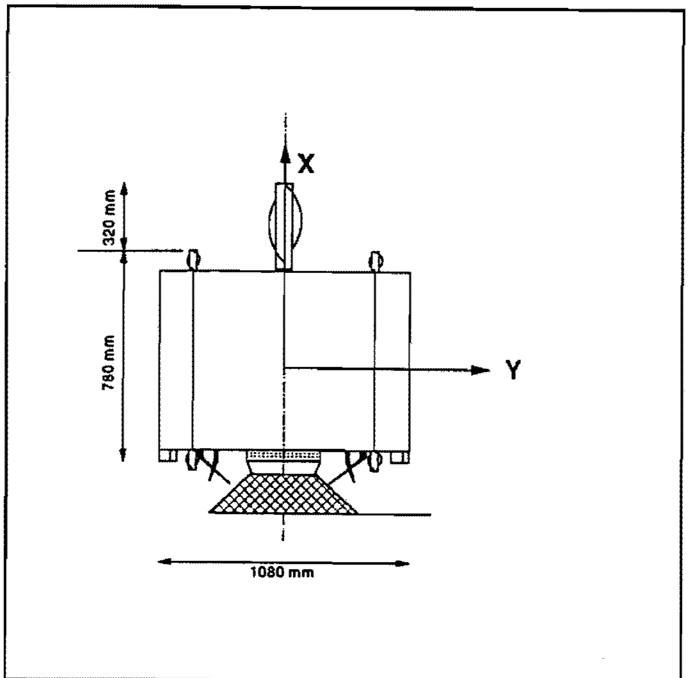



**Figure 2** The SCD 1 Satellite

Figure 2). The main structural element is a central cylinder which supports horizontal panels that carry the satellite equipments. Lateral panels covered with solar cells are attached to the horizontal panels, and the upper panel is also covered with solar cells.

The thermal control of the spacecraft is achieved using only passive means. Since all satellite surfaces but one are covered with solar cells, the ways to obtain thermal control are: selective painting and coating of the interior surfaces and electronic boxes, use of heat shields, disposal of the excess heat through the bottom panel, and control of the conduction heat paths.

The TMTC subsystem is responsible for the housekeeping telemetry transmission, telecommand and ranging. It provides ESA standard S-band telecommunication between the ground segment and the satellite. Command functions are performed through a cross-strapped system in which the two receiver outputs drive redundant decoders. Telemetry information is provided by a Pulse Code Modulation (PCM) system. Redundant TM encoders modulate either of two telemetry transmitters via a cross-strap switch. The transmitter outputs are fed to two communication antennas located on the top and bottom panels of the spacecraft, thus forming a quasi-omnidirectional radioelectric coverage, so that communications are assured in all phases throughout the satellite life. The TMTC subsystem employs the unified S-band concept where a single RF carrier is used in each direction of transmission. The up-link and down-link carrier frequencies are related by the exact ratio 221/240 and the latter carrier is generated from the former by means of a coherent transponder which relays to ground the received signal modulated by the ranging tones, thus allowing the determination of distance and velocity of the satellite.

The spacecraft receives primary electrical power from its solar arrays. A nickel-cadmium battery with 8 Ah nominal capacity supplies secondary electrical power to the spacecraft via a discharge controller. DC/DC converters transform the main bus voltage into the secondary regulated bus voltages. A power distribution unit routes electrical power to the subsystems under telecommand action.

The satellite attitude in normal operation is spin stabilized with negligible nutation by action of a nutation damper. Attitude determination is provided by sun and magnetic sensors. Magnetic and optical data are utilized on ground to estimate the satellite attitude. The attitude control subsystem is equipped with a magnetic torque coil for spin axis attitude maneuvers.

The onboard supervision subsystem comprises a redundant computer whose purpose is to acquire, process and store data from the various subsystems during the satellite passes that are not visible from the tracking station. The computer can substitute for the real time telemetry encoder, and can also distribute telecommands that are not to be executed in real time.

During the satellite integration and the launch operations, considerable flexibility and cost reduction was obtained with the use of a satellite check-out station that is small enough to be transportable and of simple enough operation to be easily adaptable to different test needs. Such features were attained with a PC-based satellite check-out station. The hardware part of this check-out station is composed of a standard desktop computer with two dedicated expansion slot circuit boards for acquisition of telemetry and generation of telecommand signals. These two cards interact directly with the computer data bus to transfer the PCM telemetry data and to compose the desired telecommand message. The connection between the check-out station and the satellite is effected only through telemetry and telecommand PSK video lines. Dedicated software running under DOS is menu driven to allow the visualization of various telemetry screens and the transmission of telecommand sequences, with continuous time-stamped storage of the telemetry and the creation of a log of the telecommands issued. The software permits an a posteriori view of a test result besides allowing the user to modify the screen content and the telecommand sequence to best fit the needs of the satellite test being conducted. Alarm tags can be related to any telemetry and the out of range telemetry identifications are displayed even if they are not being shown in the chosen screen. With little implementation, the check-out station used for testing was transformed into a low cost spacecraft control system.

## IV. Payload Interfaces

### Mechanical Interface

The payload envelope, shown in Figure 3, presents a schematic of the SCD1 payload within the Pegasus payload fairing. This figure identifies both Pegasus (46 in) and SCD1(41.3 in) dynamic envelopes, and reflects the large clearance margins for this mission. Likewise, there is more than sufficient length margin within the payload fairing, which greatly simplified payload fairing mate procedures. The payload access door is shown in its standard position and was used only for visual inspection of the payload and adapter once the fairing was installed.

The adapter/separation system is the shaded region forward of the avionics deck (station 518.96). The payload interface plane made allowances for locating a single 42 pin electrical connector within the adapter. The connector was mounted on a bracket in the center of the 10.25 in interface to the SCD1 satellite, and provided all electrical pass throughs to the payload.

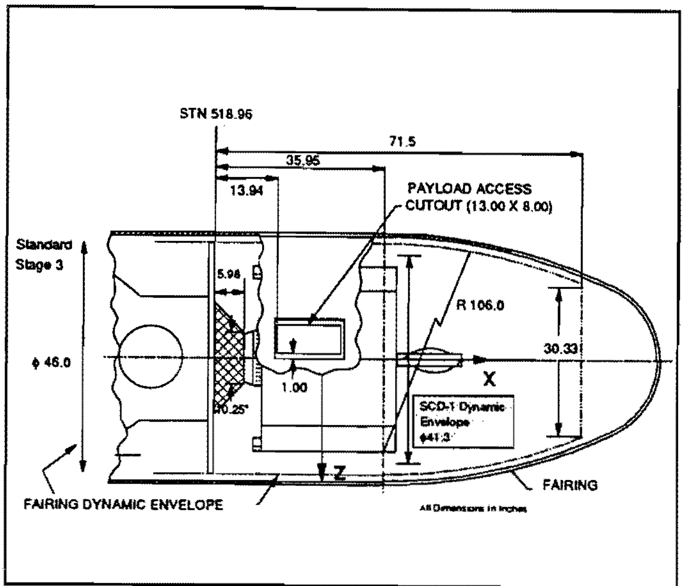



**Figure 3** Pegasus Mechanical Interface

## Electrical Interface

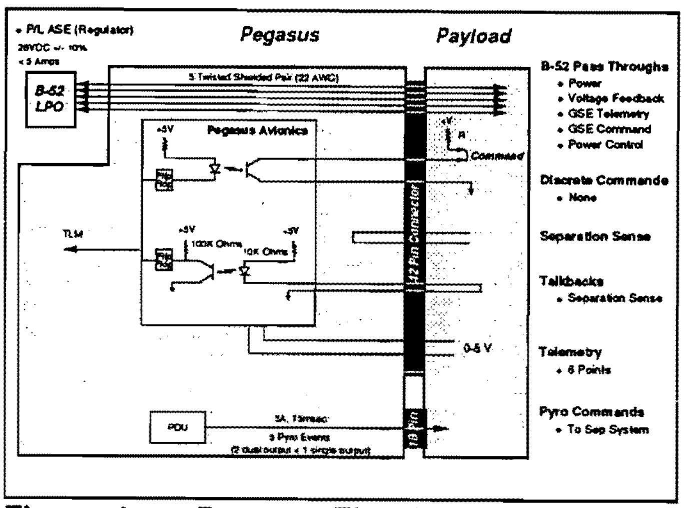



**Figure 4** Pegasus Electrical Interface

A block diagram of the Pegasus/SCD-1 electrical interface is presented in Figure 4. Though none of the available discrete commands were used, separation sensing was available both to the satellite and as telemetry talkback. Although no pyrotechnic events were required by the payload, the Pegasus pyro driver unit (PDU) provided electrical firing pulses to the separation system.

SCD1 required a telemetry interface with the launch vehicle to monitor status and health data and to display it in the control room for the payload managers. Six analog telemetry signals, ranging from 0 to 5 volts, were monitored by the Pegasus telemetry multiplexer and included in the launch vehicle telemetry stream. The monitored data included bus and battery voltages and currents, and provided a direct view of power transfer status and battery health.

The SCD1 satellite used all 5 available payload pass-throughs to the B-52, all controlled via a mission specific piece of ASE developed by INPE. This component included a DC to DC converter to regulate the 28 V power available from the B-52 and a port for a separate GSE computer which provided commands and telemetry to the satellite once the fairing was installed. The ASE box was commanded via the standard payload switches on board the B-52, and controlled B-52 external power and satellite power transfer.

## Mission Specific Hardware

The primary element of mission specific hardware was the adapter/separation system. This 6 in tall aluminum cone attached to the standard Pegasus interface, a 23.25 in diameter bolt circle of 32 equally spaced holes located on the top of the avionics deck. The other end of the adapter mated to the 10.25 in diameter Scout 200-E compatible interface of the SCD1 satellite. The separation system is an integral Marmon ring/ V-band assembly with two externally mounted pyro-technically initiated bolt cutters which are electrically connected to the Pegasus PDU. The SCD1 interface nng was mounted on top of the adapter ring and the V-Band was tensioned during payload integration.

Testing of the adapter included static loads testing and vibration testing at INPE with payload mass simulator. Multiple separation tests were performed, several tests at OSC with a payload model, and a final test in flight configuration at INPE with the SCD1 satellite. The early separation tests provided essential shock data to INPE, and in fact contained several key components mounted within the payload model to assure component survivability.

A second mission specific component consisted of redundant accelerations switches used to monitor spin rate prior to separation. Commonly known as G-switches, these units were normally open switches which close when a uni-directional acceleration reaches a set-point. Three switches were mounted radially on the avionics deck, the switches and their mounting location precisely selected to sense the 120 rpm nominal spin rate required by SCD1. After the spin up started, the switches were polled by the Pegasus multiplexer to provide data to the flight computer. Either the Pegasus IMU accelerometers or 2 out of 3 switches were sufficient to turn off the spin thrusters. Telemetry data from the mission indicates excellent performance from the G-switches.

# V. Mission Integration Overview

## Mission Schedule

The Pegasus/SCD1 program officially began at the end of August 1992, immediately setting into motion the mission analysis and interface requirements definition processes . The goal of the program was to launch the SCD1 satellite by the end of 1992, cutting at least 13 months from the nominal 18-24 month mission timeline. Therefore, many of the mission integration phases were performed in parallel.

Mission analyses included both trajectory analyses and a coupled loads analysis to determine spacecraft loads. Mission analysis results were included in the range coordination and documentation process as soon as they were available, primarily for flight safety analyses. Interface definition also impacted range coordination, culminating in several range data flow tests in preparation for launch. Also begun immediately, the export control process shown here includes all licensing through the U.S. Dept. of State(export) and Dept. of Transportation (commercial launch licensing).

An adapter/separation system was needed to mechanically attach the SCD1 satellite to the Pegasus and to provide for spacecraft separation on command. This adapter was designed, built and tested by OSC prior to shipping to INPE for final vibration and separation testing with flight hardware.

Pegasus vehicle integration began October 15th, with the arrival of the stage 2 and 3 motors at the Vehicle Assembly Building (VAB). The stage 1 motor arrived at the end of October, leading to stage mate and vehicle integrated testing. The SCD1 satellite arrived at the VAB near the end of November in support of the original launch window of 12-18 December.

Eastern Range announced some conflicts with the launch window in early December. Given the proximity of the holidays, a range standown of approximately 30 days was announced. After the range standown, vehicle integration resumed and final payload integrated testing was performed. The final launch window was selected as 9-21 Feb, with the ferry flight occurring on Feb 7th.

### Integrated Payload Operations

The SCD1 satellite arrived at the Vehicle Assembly Building on November 16th in preparation for integrated testing and mate. The nominal spacecraft integration flow is presented in Figure 5. After transportation and unpacking, the GSE was installed and satellite final integration began. After final integration, the satellite was mated to

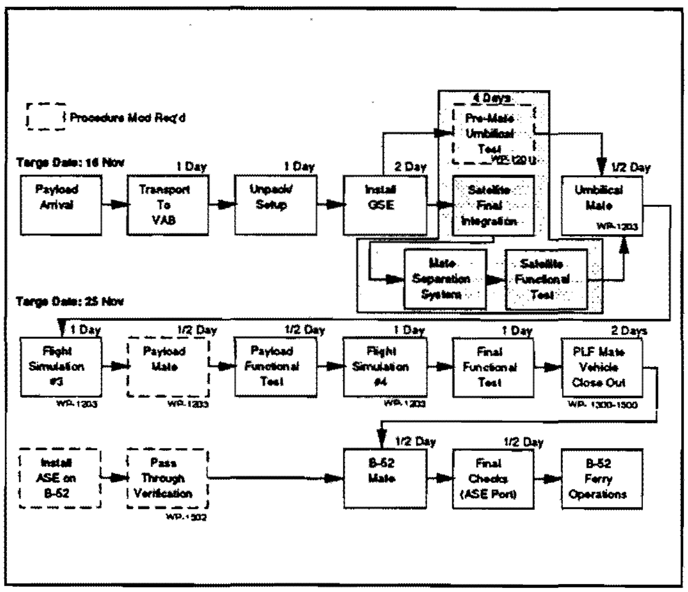



**Figure 5** Pegasus/SCD1 Integration

the adapter/separation system and the clamp band was tensioned to flight levels (a 1 day procedure). Prior to electrically mating to the Pegasus launch vehicle, a series of satellite functional tests and interface tests were performed. Finally, at the end of this testing sequence, the satellite was electrically mated to the launch vehicle via test umbilicals in preparation for the final flight simulations.

Flight simulation #3, a real time simulation of all flight events while on external power, was performed on schedule, November 25th. All telemetry from the flight simulation was collected and reviewed following the test. Following this test, the umbilicals were removed and the payload was mated to the vehicle. A full series of functional tests were again performed on the satellite just prior to the final flight simulation (#4). The final flight simulation is performed as close to flight configuration as possible. All flight electrical connections have been mated and the vehicle is on internal (battery) power. A final series of satellite functional tests were performed after flight sim #4. This simulation completes the process of integrated vehicle/payload testing, leaving only payload fairing mechanical mate and vehicle closeout prior to mating with the B-52 carrier aircraft.

Prior to B-52 mate, and in fact prior to integrated testing, the SCD1 ASE Controller was installed on the B-52 and functionally tested. All payload pass through wires were verified from the LPO station to the B-52/Pegasus mating connectors. The pass through wires were also verified on the vehicle in the umbilical pre-mate testing, assuring the functionality of all interfaces from the B-52 to the satellite.

### Payload Mating Operations

Because the adapter/separation system was mated to the satellite first, the physical lifting and mating of the payload to the Pegasus vehicle was essentially similar to the previous two non-separating payloads. The final mate to the Pegasus again involved rotating the satellite to a horizontal position, locating it on the avionics deck, and installing the 32 fasteners. The exceptional quality and functionality of INPE's mechanical GSE made for an extremely smooth mating operation.

After the payload was mated to the adapter, it was bolted to a rolling GSE fixture in a vertical position. This provided the most stable position for the satellite and allowed access for functional testing and connector/umbilical mating. At this point the payload was cleaned and moved to the class 10,000 clean room for final mating.

The payload was moved from its vertical stand to a mechanical break-over fixture via an overhead crane in the clean room. This fixture securely held the satellite as it was rotated from a vertical to a horizontal position. With the satellite held in a horizontal position, a lifting fixture was attached to the front and rear satellite hard points. This lifting fixture, unlike those used throughout satellite build-up and test, carried the satellite in a horizontal mode, and allowed final positioning for mechanical mate with the Pegasus.

Once the satellite was raised to an approximate mating height, it was moved adjacent to the launch vehicle, and the team proceeded with fine adjustment of position. The payload was positioned in contact with the avionics deck to allow insertion of the 32 fasteners through the deck to the base of the adapter.

After the mechanical mate, the 42 pin connector flight umbilical was mated to the Pegasus electrical connector on the avionics deck. Final close out of the separation system included installation of the ordnance lines and time delays. Final ordnance hook-up occurred just prior to fairing mate.

## VI. Mission Operations

### Operations Overview

The SCD1 mission was performed with a standard three stage Pegasus vehicle. Standard vehicle integration and ground processing was performed at OSC's Vehicle Assembly Building (VAB) at Dryden Flight Research Facility (Edwards AFB, California-see Figure 6). Spacecraft mate and Pegasus/SCD1 integrated testing also occurred in the VAB. However, due to the low inclination of the orbit, flight operations could not be carried out from the Western Range (WR) as

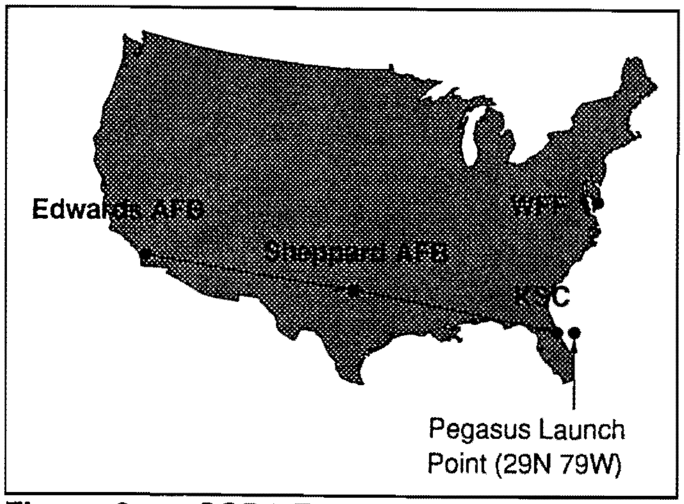



**Figure 6** SCD1 Ferry Flight

in the previous Pegasus launches. Wallops Flight Facility (WFF) was selected as the lead range, while the Pegasus flight operations occurred off the coast of Florida near Eastern Range (ER) assets.

**Range Support**

NASA's Wallops Flight Facility was selected as the lead range for this mission and was the location of the mission control center, though ground and launch operations were performed at KSC. A variety of Eastern Range (ER) assets were called upon to support launch operations (see Figure 7). Pegasus telemetry was received at MILA

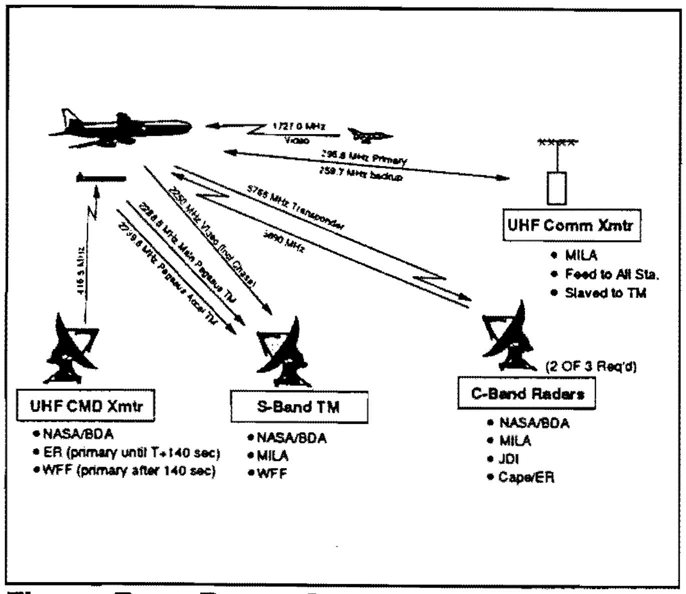



**Figure 7** Range Support Assets

along with B-52 video. Video from the chase plane was received on the B-52 and was downlinked along with the forward and aft bomber cameras. UHF communication to the B-52 was also accomplished through remote microphone keying at MILA. C-Band tracking transponder radars were located at MILA, Patrick AFB and Johnathan Dickerson Facility in West Palm Beach. Finally, UHF Command Destruct Transmitters were controlled by the ER Range Safety Officers during captive carry and throughout the initial portions of the flight.

After approximately 140 sec, both Bermuda (BDA) and Wallops (WFF) assets were able to assume primary command destruct responsibility. Telemetry and C-Band Tracking functions were also available at each site.

All data sources were taped at the receiving stations and simultaneously transmitted to the Control Room at WFF. Both land lines and satellite relays were used for safety critical data. OSC telemetry display screens were used to display vehicle status.

**Ferry Flight**

Pegasus was mated to the B-52 and final payload checkout was performed at DFRF. The B-52 then ferried the Pegasus and the SCD1 payload across the country to the Shuttle Landing Facility at Kennedy Space Center in Florida, with an intermediate stop for refueling occurred at Sheppard AFB, Texas.

The B-52 took off from Dryden at 0630 Pacific Standard Time on February 5th after a 30 min weather delay. Vehicle, payload and bomber telemetry were recorded during all takeoffs and landings. Weather was excellent during the first leg of the trip, with the B-52 arriving at Sheppard AFB at 1100 CST . During the descent, landing and taxi, onboard nitrogen purge was used to avoid condensation within the payload fairing. Nitrogen purge was maintained until low humidity, heated, ground air was available.

The refueling and turn-around maintenance at Sheppard took somewhat longer than expected. In addition to fueling, water for the engine injection system and payload nitrogen were replaced. Though the best case nominal turn-around time was 1 hr 30 min, the B-52 did not take off again until 1500 CST, partly due to concern over the weather in Florida.

Even with cloud cover over most of Florida, weather did not affect the final leg of the trip. On approach to KSC, Range assets were brought on-line to verify telemetry, radar and UHF links. Data was received at KSC and transmitted to the control room at Wallops in an end to end test of all required assets. UHF communication with the B-52 was successful along with excellent telemetry, multi-source video, and radar coverage. Touchdown at KSC occurred at 1830 EST, where ground operations began immediately and continued well into the night. A full dress rehearsal was held the next day, February 8th, with an F-18 flying the B-52 route and all range assets online.

After the ferry operation, Pegasus/SCD1 remained on the ground one day at KSC for final checkout and motor thermal conditioning. Launch operations started range set up at 0200 EST on the morning of February 9th. B-52 takeoff occurred at 08:15 EST, with the flight out to the drop point taking approximately 74 minutes. A 2.5 hr launch window was available.

## VII. Flight Overview

**Nominal Mission Description**

The nominal mission profile is presented in Figure 8 for the 750 km (405 nm), 25° inclination target orbit. The nominal drop point for the mission was 29° N 79° W, just off the coast of Florida's Eastern Range. Standard B-52 drop conditions of 41,500 ft and .82 Mach (760 fps) were assumed. Similar to the first Pegasus mission, the

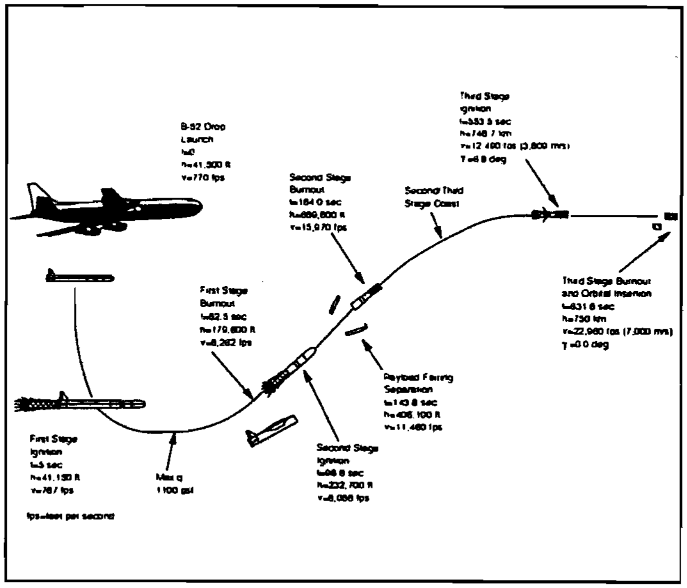



**Figure 8** Pegasus Nominal Trajectory

second stage burns occurs almost immediately after stage 1 burnout. Payload fairing separation occurs 144 sec into the flight, during second stage burn. After second stage burnout, a long coast takes the stage 3/payload combination to the proper orbital altitude, where the final burn essentially circularizes the orbit.

Immediately after stage 3 burnout, the OXP-1 secondary payload is separated. After a short period of motor outgassing, Pegasus performs a orientation maneuver to align the payload with the predicted sun vector (+30 deg declination, +45 deg right ascension from sun). Finally, 60 seconds after stage 3 burnout, the cold gas RCS system spins up the stage 3/SCD1 stack to 120 rpm and separates the primary payload.

There were significant performance margins on this mission due to the small payload coupled with a low inclination orbit (Table 1). This excess performance allowed considerable flexibility in selecting the

**Table 1** Mission Margin

| Item | Weight | |
|---|---|---|
| | lbs | kg |
| Nominal Capability (750 km, 25 deg)<br>Assumes Standard Pegasus Configuration<br>Based on Estimated Motor Performance<br>Assumes 220 fps Velocity Reserve | 381 | 173 |
| **Estimated Payload Weight** | 253 | 115 |
| **Payload Adapter/Separation System** | 20 | 9.1 |
| **OXP-1 Secondary Payload** | 29.9 | 13.6 |
| **SCD1 MISSION MARGIN** | 78.10 | 35.30 |

drop point. A "dogleg" maneuver could be used to move the drop point north of 25° latitude while maintaining a 25° inclination orbit (i.e. without a dogleg the most northerly drop point would have been 25°). In addition, Pegasus' adaptive guidance routines allowed in-flight energy dissipation maneuvers to further scrub excess energy prior to orbital injection.

## Flight Results

The primary source of flight reconstruction data is Pegasus telemetry data. Significant amounts of guidance, navigation (IMU and GPS) and control telemetry is available along with vehicle temperature, pressure and strain data. Sequencing discretes are issued from the flight computer and are also recorded. Various sources and types of data are available from the range for comparison.

At the release point, the B-52 was at 43,060 ft with an earth relative speed of 844 fps (INS altitude and speed). The actual release point was 29.04 N, 78.99 W (Nominal 29 N, 79 W) at 14:30:34 GMT (09:30:34 EST). All vehicle sequencing references the drop time as T=0.

Stage 1 ignition occurred at its nominal time, 5 sec after drop. At this point the vehicle was at 42,630 ft with a speed of 852 fps and its rate of decent was 130 fps. Stage 1 burn was slightly longer than nominal, with burnout occurring 420 msec after the 82.5 sec mark. Stage 1 separation is a time commanded event which is followed by Stage 2 ignition via an ordnance time delay. Likewise, fairing separation is a commanded event and occurs at the expected time.

Stage 2 separation and Stage 3 ignition times, again linked by an ordnance time delay, are variable depending on the real time guidance solution after Stage 2 burnout. Since there was excess velocity after Stage 2 burnout, the Stage 3 ignition time was delayed to 557.96 in order to perform an energy scrubbing maneuver. The propulsive segment of the flight was completed 10 min 38 sec after drop.

Experiment activation (OXP-1 separation) and SCD1 spin up are initiated on a timer from the burnout point. Electrical initiation of the pyrotechnic bolt cutters used to separate the payload occurred 14 sec after the start of spin up. Immediately following separation, WFF and BDA receivers verified SCD1 transmitters had turned on (breakwire initiated upon separation).

Roughly 20 minutes after drop, the B-52 and chase plane returned to the SLF at KSC.

## Range Tracking Results

Time tagged radar data is available from the various tracking sites which can be compared to the IMU data. IMU flight data closely matches pre-launch nominals except for the effects of delaying the third stage ignition time (discussed above). Therefore, nominal curves are not included on ground tracks and instantaneous impact point (IIP) charts for the purposes of clarity.

Radar data includes magnitude of the velocity vector, altitude, latitude and longitude (both present position and vacuum impact predictions) and other position products such as downrange and crossrange distances. In addition to any inherent error in the tracking systems, the error associated with identifying the drop time makes radar data relatively inaccurate for detailed guidance and navigation analyses. This function is much better served by the independent GPS fixes (when available). However, radar data is useful for providing aggregate flight results and for broad based comparisons to telemetry data.

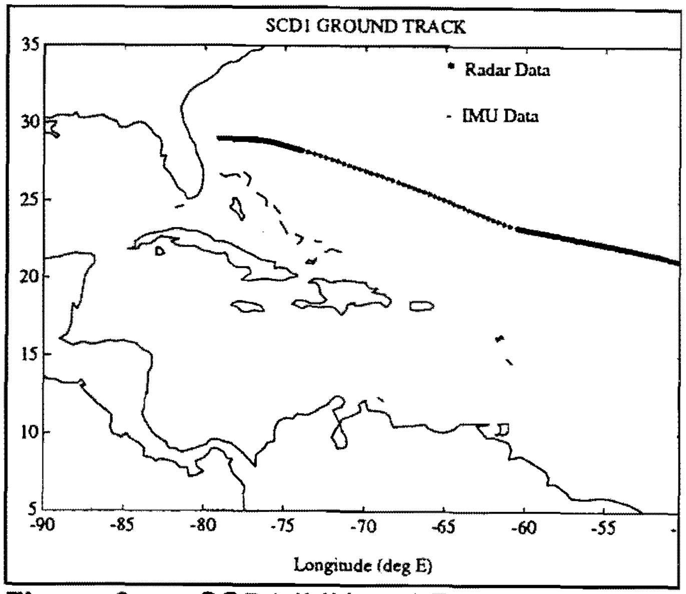



**Figure 9** SCD1 IMU and Radar Position

A comparison of IMU position to radar position is presented in Figure 9. Radar data agrees well with IMU position throughout the flight and shows both out of plane energy management maneuvers. One "dogleg" is used to achieve the 25 deg final orbit inclination from the 29 deg N drop point. The second dogleg is a pre-planned energy scrubbing maneuver, used in place of ballast due to the low payload weight, during third stage burn.

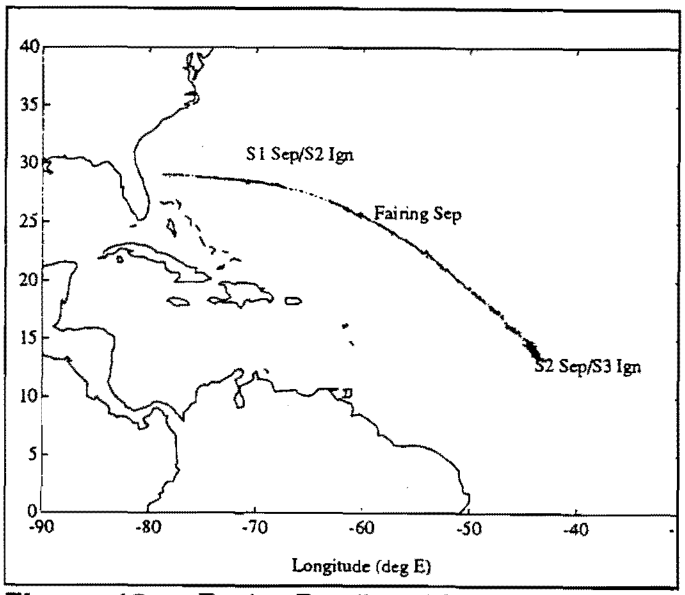



**Figure 10** Radar Predicted Impact Points

Radar impact point predictions through stage 3 ignition are shown in Figure 10 overlaid on a regional map. The location of stage and fairing impact points are highlighted (based on vacuum predictions correlated by vehicle sequencer time), indicating the relatively benign locations of impacting bodies. Third stage burn carried the instantaneous impact points across southern Africa as in most east coast launches (note: the third stage is orbital and does not re-enter). The 4 sec total dwell time over the land mass at the end of third stage burn was taken into account in the flight safety analysis and did not pose any unusual risk.

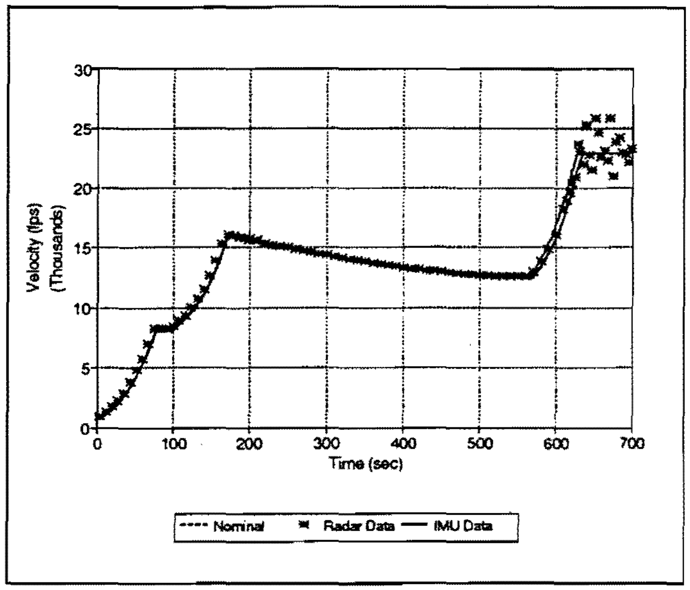



**Figure 11** IMU and Radar Velocity

The final two figures, Figure 11 and Figure 12, show respectively velocity and altitude as a function of time (drop=0). Radar data again shows good agreement with the Pegasus IMU at a macro level. The velocity chart clearly shows the successive burns of stages 1 and 2 as steep increases in velocity. The long coast up to orbital altitude manifests itself as a slow decrease in velocity over 370 sec, followed by stage 3 burn. Note that there are significant errors in the radar velocity following stage 3 burnout. The nominal (expected) velocity curve reflects the nominal stage 3 ignition time.

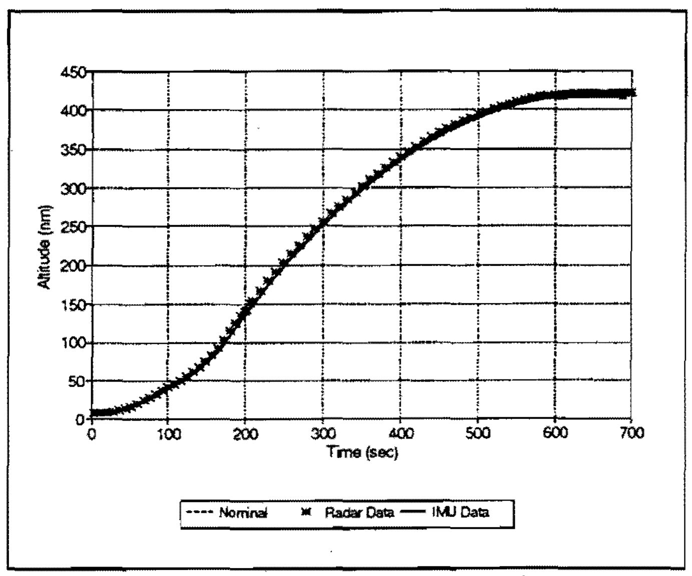



**Figure 12** IMU and Radar Altitude

Radar altitude data does not suffer as severely at the end of stage 3 burn, however increases in error are seen. Stage 1 and 2 burn (through 180 sec) reflect the initial pull up maneuver and increase of the vehicle flight path angle. Altitude increases over 200 nm during the long stage 2/3 coast. Note that stage 3 burn (568-638 sec) occurs essentially at the final orbit altitude. Finally, radar results are seen to be consistently higher than IMU data (drop time variations of even 1 sec can significantly alter results).

**Mission Constraints**

## Table 2 Mission Constraints vs Actuals

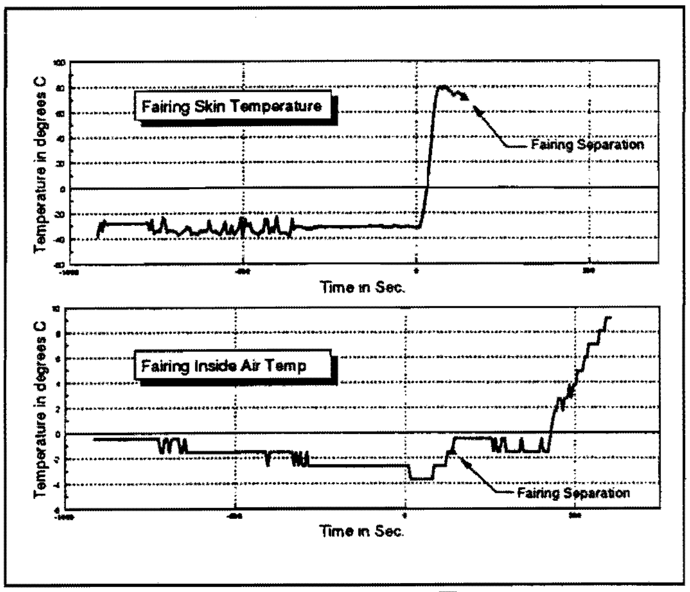
| Item | Target | Required | Actuals |
|---|---|---|---|
| Altitude | 405 nm | ±100 km (54) nm | 392 x 426 nm |
| Inclination | 25 deg | ± .2 deg | 24.97 deg |
| Spin Rate | 120 rpm | ± 20 rpm | 120.1 ± 3 rpm |
| SCD1 Weight (OXP-1) | 253 30 | N/A | N/A |

Pegasus/SCD1 mission constraints are shown in Table 2 along with final mission results. Orbit elements were generated through multiple orbit tracks by NORAD. The spin rate estimate at separation was developed from IMU acceleration telemetry and was confirmed by initial SCD1 telemetry. The ± 3 rpm value reflects uncertainties in the data and variability in IMU placement. Pegasus easily met all payload requirements and mission success criteria for this mission. Though Table 2 reflects the ICD requirements for the mission, the mission success criteria were much broader. Orbital accuracy goals were ± 250 km (135 nm) for apogee and perigee and +2/-3 deg for inclination, while the spin rate goal was 80 to 160 rpm.

### A. Payload Environments

Most of the key payload environments are available from Pegasus primary PCM telemetry. Body accelerations are measured by the IMU, while temperatures are available at various locations within the fairing. Random vibration levels and the drop transient acceleration, were determined via accelerometer data and closely matched expected levels. Acoustic data in the payload fairing was closely monitored during the three inert flights and the first mission. Given its low levels, acoustic data is not collected on each mission.

Payload fairing temperatures are shown in Figure 13 for both the captive carry and free flight portions of the mission. Zero on the time scale represents drop, therefore the graphs cover from 15 min prior to drop through orbit insertion. The first graph is a representative temperature on the external skin of the fairing cylindrical section. Though temperatures are obviously higher at the nose (stagnation point) and along the ogive, the temperature most likely experienced by a majority of the spacecraft components is that of the cylinder. Even though this data represents an external temperature, this provides a fairly good estimate of internal skin temperature since the fairing is entirely graphite composite ( in general there is a small temperature gradient, on the order of 10° C, across the fairing wall).




## Figure 13 Payload Fairing Temperatures

The second graph relates air temperature within the payload fairing as measured by a thermistor mounted to the Pegasus avionics deck. The data shows a slow reduction in temperature during the captive carry due to the cold external wall temperature. Note that the wall temperature remains constant throughout this period. During powered flight, the expected aeroheating temperature rise is experienced. Aeroheating is much less pronounced due to the air launched nature of the Pegasus.

This data correlates very well with temperature data for other components such as the avionics batteries, various avionics components and the RCS tank. The environment is relatively benign and the temperature gradients are small compared to ground launch vehicles.

Vehicle body acceleration, as recorded by the IMU, compares well with expected maximum loads as represented in the ICD. Figure 14 shows the axial acceleration (X-axis) levels recorded during free flight. After stage ignition, acceleration quickly ramps up to a modest level, and then gradually increase as propellant mass is expelled. Acceleration tails off quickly at stage burnout, except for stage 1, where the graph clearly reflects fin rocket burn (around 90-100 sec). As expected, the low payload weight causes the highest accelerations, over 9 g's, just prior to stage 3 burnout. Such levels were not a concern to the SCD1 satellite, which was designed and tested for 17 g's axial acceleration. The expected levels were also used to static load the adapter/separation system.

Lateral and vertical acceleration levels, though not as extreme, also drive design and test requirements. Figure 15 shows the Y and Z axis accelerations from drop through stage 3 burnout. The Z axis is positive downward while the Y axis points out the right wing. Z-axis acceleration levels clearly show the 2.35 g (nominal) pull up maneuver during stage 1 burn. As is evident, most activity in the lateral and vertical directions occurs during stage 1 burn, most likely the result of aerodynamic forces.

Because IMU data is recorded at 25 Hz, it does not have the resolution to clearly capture significant transient events such as drop.

## Payload Integration and Testing

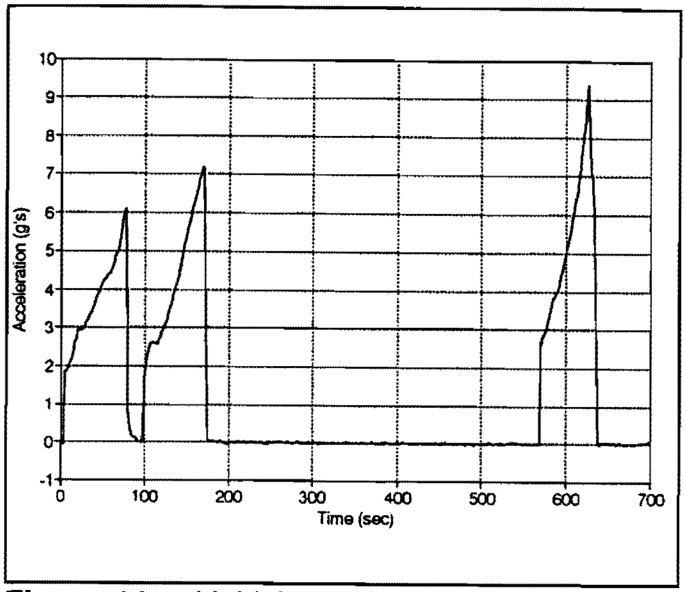



**Figure 14** Vehicle Axial Acceleration (From IMU Data)

This information will be available later from independent accelerometer data.

## VIII. Lessons Learned

### Program Management

The SCD1 program was an excellent example of Pegasus philosophy. An accelerated mission schedule was made possible by simple interfaces, a well understood and uncomplicated payload and straightforward integration operations. The SCD1 satellite's robust, uncomplicated design simplified operations planning, the safety analysis process, and range coordination. The ability of Pegasus to be completely integrated and tested prior to payload arrival allowed payload and mission specific hardware testing to proceed in parallel with vehicle build up. The small team approach, by both INPE and OSC, aided early program planning and allowed issues to be identified and resolved quickly.

For future international missions, the technology transfer and licensing process will be started earlier earlier. Though this was not possible given the time constraints of this project, it would allow early involvement of State Dept/DOT and would simplify the documentation and review processes.

### Interfaces

Simple and flexible interfaces were essential in the success of the program. Even though the satellite was essentially completed by contract go-ahead, most interface decisions were made during the first working group in Brazil. Both parties worked aggressively to solve interface discrepancies. Concentrating primarily on Pegasus standard services made hardware design, testing and evaluation easier, as well as simplifying planning, documentation, and operations.

### Payload Integration and Testing

Several key issues allowed integration and testing to occur within a short period. SCD1 mate and testing was very similar to previous mission. Also, modified procedures were kept to a minimum, and interface testing schedules were kept flexible. Adapter fit checks and testing were performed with the satellite prior to arrival in the VAB. Finally, INPE had excellent GSE hardware, greatly simplifying payload mating procedures.

There were, however, some initial concerns with VAB cleanliness upon payload arrival. Though cleanliness levels were not a concern, there was a delay due to insufficient preparation of the building. To address this issue a Payload Integration Plan (PIP) will be established to transmit requirements to the field site prior to payload arrival.

## IX. Conclusion

The SCD1 satellite was successfully and accurately placed into orbit by its Pegasus launch vehicle on February 9th, 1993. The flight was the culmination of months of accelerated mission planning and integration by INPE and OSC, and reflects many of the strengths of lightsat philosophy. A six month schedule from go-ahead to launch was achieved in spite of programmatic delays. Simple and effective hardware and interfaces allowed a straightforward, efficient and relatively short payload integration and test process.

Significant contributions by NASA, Dryden, Wallops and the Eastern Range were essential in the success of the mission. Mission planning addressed complex operations, including a cross country ferry flight , ground operations significantly removed from the control room, and a first time east coast Pegasus launch from a new range.

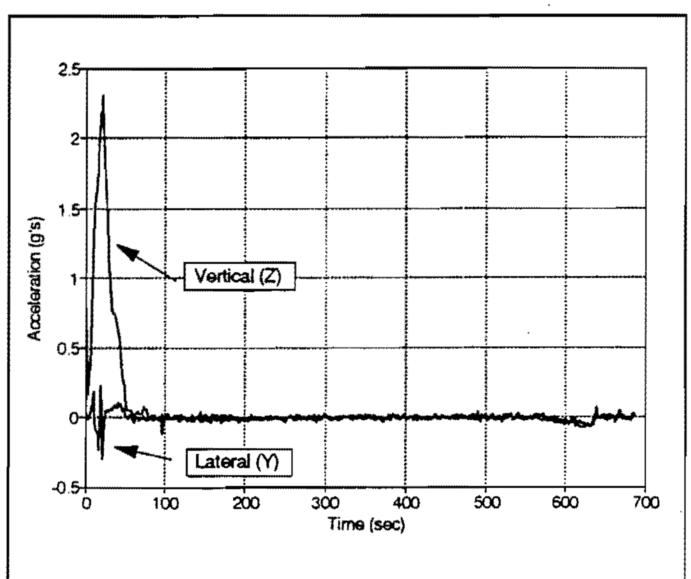



**Figure 15** Vehicle Vertical and Lateral Accelerations (+Z-axis down, +Y-axis right)