PEGASUS* FIRST MISSION - FLIGHT RESULTS

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ABSTRACT

On April 5, 1990, after release from the wing of a B-52 carrier aircraft over the Pacific ocean at an altitude of 43,198 ft, the three stage Pegasus solid propellant rocket successfully completed its maiden flight by injecting its 423 lb payload into a 273 X 370 nautical mile 94 degree inclination orbit. The first flight successfully achieved all mission objectives; validating Pegasus's unique air launched concept, the vehicle's design, as well as its straightforward ground processing, integration and test methods. This report summarizes the results of the first launch, including measured vs. predicted motor performance, drag and lift coefficients, payload environmental parameters, structural loads, aerodynamic heating, and vehicle trajectory. In all areas, measured flight results were close to design predictions, and in the case of the actual payload environment, were significantly less than predictions. The Pegasus first flight validated the fundamental aerodynamic design, established a baseline performance capability, validated the vehicle's GN&C system, and validated the aerodynamic and aero-thermal models.

1.0 Introduction

On April 5, 1990, at 1210 PM PDT, Pegasus was released for the first time from the wing of a NASA B-52 carrier aircraft flying on a southerly heading over the Pacific ocean at an altitude of 43,198 ft. After falling for 5 seconds to clear the carrier aircraft, the three stage solid propellant winged rocket ignited its first stage motor and flew an optimal lifting ascent trajectory and placed its 423 lb payload into a 273 X 370 nautical mile 94 degree inclination orbit. The maiden flight of Pegasus represents the first time that an air launched rocket has placed a payload into orbit. The launch was conducted by the Defense Advanced Research Projects Agency (DARPA) as part of its Advanced Space Technology Program (ASTP) to test and evaluate Pegasus for future military applications. To support evaluation, the first Pegasus mission flew a payload environment instrumentation package developed for DARPA by NASA Goddard Space Flight Center (GFSC) and a hypersonic aerodynamics wing and wing-to-body fillet instrumentation package developed by NASA Ames Dryden Flight Research Facility (DFRF). Important additional mission objectives included the successful deployment of a Small Experimental Communication Satellite (SECS); the delivery to orbit, pointing and initial spin-up of a NASA GSFC barium canister experiment (PEGSAT) for later on-orbit barium canister deployment; and finally collecting Pegasus engineering data. All mission objectives were achieved. Pegasus's unique air launched concept has been demonstrated and the vehicle's simple and robust design, and digital avionics system validated.

The Pegasus air launched space booster, which has received a United States Patent, is the product of three year privately funded joint venture of Orbital Sciences Corporation and Hercules Aerospace Company. The first Pegasus mission was funded by DARPA through its Advanced Vehicle Systems Technology Office (AVSTO) with support from NASA Ames DFRF and the Air Force Space Division through agreements with DARPA. Under this cooperative program, OSC and Hercules funded all vehicle development, tooling and facilities costs, while the government funded non-recurring costs associated with carrier aircraft operations, range safety operations, vehicle safety certification, and some vehicle design changes requested by the government, as well as the recurring cost for its missions.

The vehicle was developed to provide a cost effective, reliable, and flexible means of placing small satellites into low earth orbit. It is carried aloft by a conventional transport/bomber-class aircraft (B-52, B-747, L-1011, etc.) to a nominal level-flight drop condition of 42,000 feet at high subsonic velocity. After release, the vehicle free falls with guidance active to clear the carrier aircraft while executing a pitch-up maneuver to place it in the proper attitude for motor ignition. After first stage ignition, the vehicle follows a lifting-ascent trajectory to orbit.

2.0 Vehicle Description

The Pegasus air launched space booster, shown in Figure 2.0-1, is 50 foot long, 50 inches in diameter and weights 42,000 lbs. Major components include three graphite composite solid-propellant rocket motors, a fixed high mounted graphite composite delta wing, an aluminum aft skirt assembly, three graphite composite fins controlled by electro-mechanical actuators, a composite avionics/payload support structure, and a two-piece composite payload fairing. The three solid rocket motors and payload fairing were designed and developed specifically for Pegasus by Hercules Aerospace. The graphite composite delta wing has a 22 ft wing span with carbon composite spars and foam filled panels. The three foam core graphite composite fins provide aerodynamic control throughout the first stage burn. Pitch and yaw control during the second and third stage operation is provided by electro-mechanical thrust vector control (TVC) actuators. Roll control after stage-1 separation is provided by an avionics structure mounted nitrogen cold gas reaction control system. The wing, fins, and wing-to-body fillet were designed and built by Scaled Composites of Mojave, California. The aft skirt subsystem is an aluminum cylindrical section supporting three electro-mechanical fin actuators.

Pegasus Cutaway Drawing

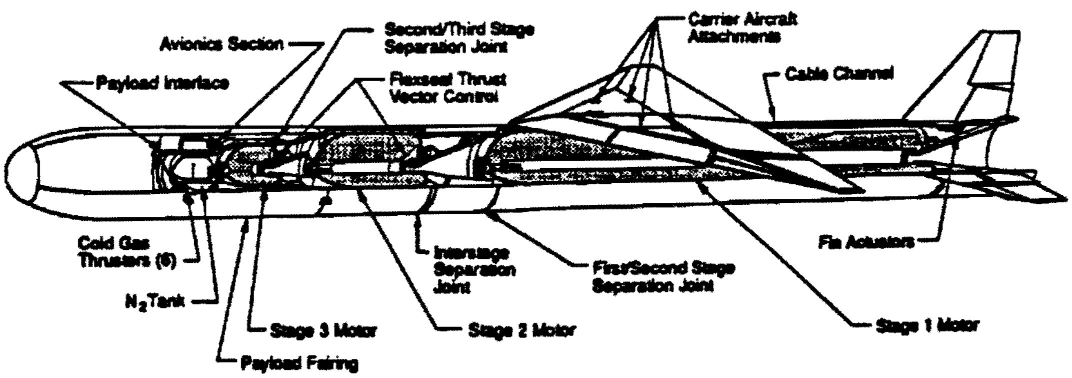

Figure 2.0-1

The graphite composite avionics structure and aluminum honeycomb deck supports most vehicle avionics as well as the payload. The payload fairing, which encloses the payload, avionics subsystem, and third stage motor, is a pyrotechnically separated two-piece graphite composite structure with the same 50 inch outside diameter as the second stage motor.

The vehicle avionics system, shown in Figure 2.0-2, was designed to be simple, robust and reliable. Pegasus is controlled by a Motorola 68020 CPU based flight computer which commands all flight events, executes the autopilot program, and formats vehicle telemetry prior to transmission. An Inertial Measurement Unit (IMU) provides vehicle attitude, velocity and navigation information. All remote avionics units, which include Pyrotechnic Driver Units (PDUs), Telemetry Multiplexors (MUXs), Thruster Driver Units (TDUs), and Thrust Vector/ Fin Actuator Controllers have integral microprocessors which communicate with the flight computer using digital RS-422 communication lines. During flight, all critical vehicle performance parameters are transmitted to the ground using a single 56 kbps S-band telemetry channel. The use of an industry standard RS-422 communication protocol simplifies vehicle wiring, streamlines testing and integration, and significantly reduces test and ground support equipment costs.

Pegasus Functional Diagram

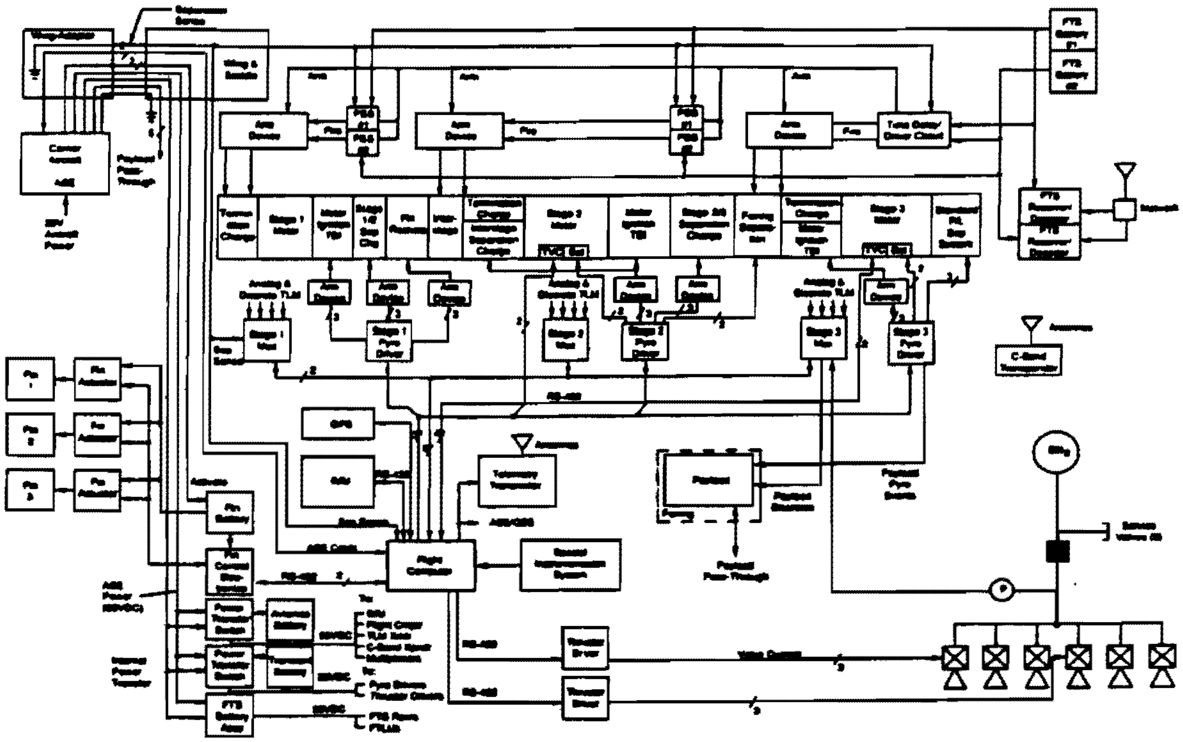

Figure 2.0-2

3.0 Ground and Captive Carry Operation Results

3.1 Ground Operations

A primary goal of the Pegasus program is to minimize field integration effort, facilities and equipment. Pegasus is integrated horizontally, as shown in Figure 3.1-1, at a convenient working height which allows easy access to all areas of the vehicle for component installation, test and inspection. Custom designed articulated dollies support all integration activities and eliminate the need for lifting motors in the field. The use of a RS-422 communications protocol in Pegasus avionics simplifies avionics testing and ground support equipment. The integration and test process has been developed to ensure that all components and subsystems are thoroughly tested before and after final flight connections are made, and include several "fly into orbit" simulations which exercise all actuators and pyro initiation outputs. Pegasus configuration control and integration activities are controlled by Work Packages and Procedural Guides, which describe in detail and document every step and aspect of integrating and testing a Pegasus vehicle and payload.

Field integration of the F-1 vehicle, as depicted in Figure 3.1-2, required 89 days from delivery of the stage 1 motor to launch, including almost four weeks of deliberate hold time. The integration went smoothly, with a minimum number of red lines and other corrections. There were no open non-conformance reports at the pre-launch review.

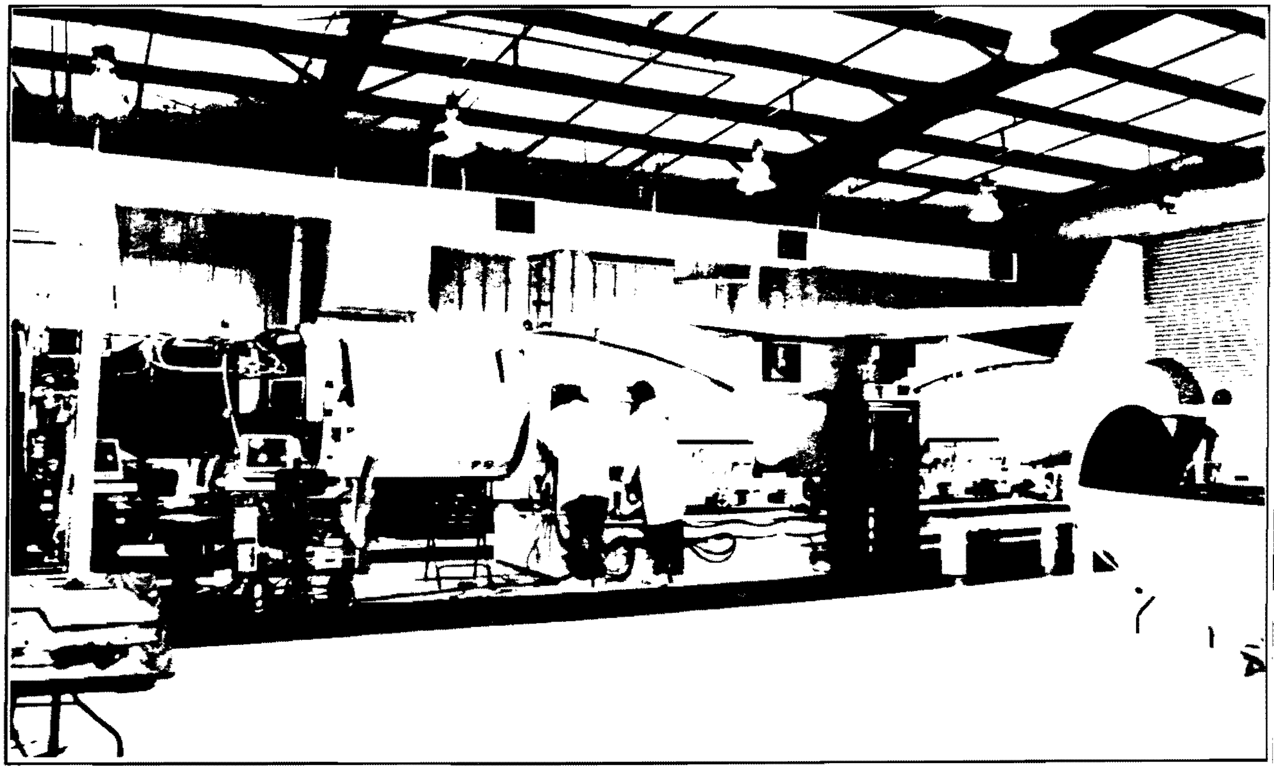

Figure 3.1-1

The "L" number above and to the left of each box represents the baseline days before launch for that step based on achieving the program goal of a 14 day delivery to launch schedule. The right hand numbers represent the actual date and days before launch. As can be seen, after accounting for the 1 week hold, the achieved schedule tracked very well from the time the vehicle was ready for payload mate to launch.

F1 Payload Processing Flow

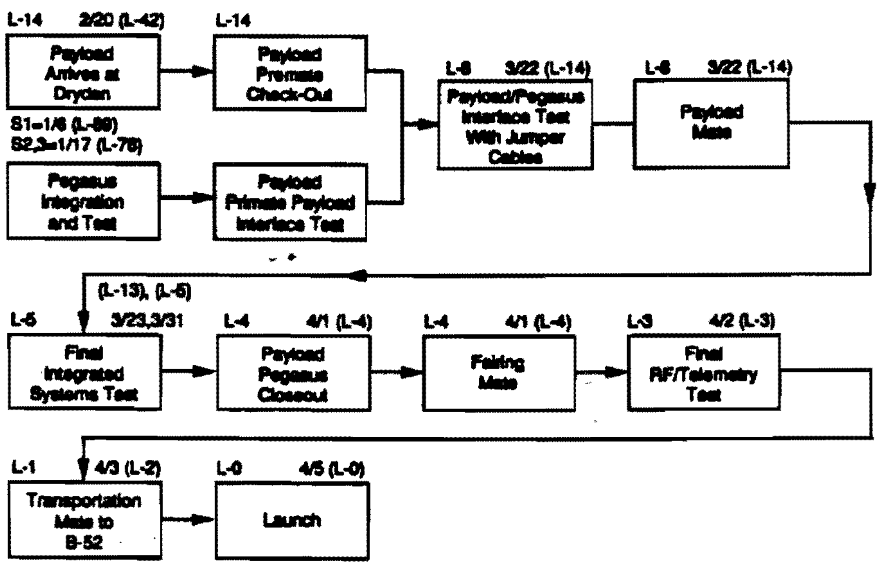

Figure 3.1-2

3.2 Launch Operations

Pegasus launch operations combine a mix of conventional launch vehicle processing and aircraft operations, with an emphasis on simplicity, flexibility, and operational discipline. The NASA Ames Dryden Launch Control Center (Blue Room) was used to control the F1 mission with telemetry, voice and real time video support from the Western Test Range (WTR) in Vandenberg, California. All communications networks functioned nominally during the mission. Launch operations were executed without incident using mission checklists developed during the Pegasus inert vehicle flight test program, which consisted of three captive carry flights conducted with a full size inert vehicle between November 1989 and January 1990. On April 4, ground operations, as shown in Figure 3.2-1, proceeded to the point that the carrier aircraft was ready for takeoff when the mission was scrubbed due to weather (a cloud deck at 25,000 feet would not have allowed the carrier aircraft to remain VFR throughout the flight as required by the mission rules). A time line for the actual launch operations, which resumed on April 5, 1990, is summarized in Table 3.2-2.

All ground operations proceeded nominally. Avionics and payload temperatures inside the fairing remained well within tolerances using only the on-board nitrogen purge system. Take off occurred on schedule, and all post take-off activities were performed as planned. In flight vectoring to compensate for unexpectedly favorable winds (the aircraft was several minutes ahead of schedule when it reached the vicinity of the drop point) over-compensated and resulted in the carrier aircraft arriving at the drop point approximately 3 minutes late, but well within the 20 minute launch window required by the payload. All launch criteria were satisfied at time of drop.

Pegasus on B-52 Aircraft

Figure 3.2-1

Pegasus Flight 1 Timeline and Parameters

Checklist	Planned Start	Actual Start
Range Set-Up	0633	0633
LPO Entry	0733	0733
LPO Power-On	0738	0735
Range-Safety Checks	0748	0740
LPO Verification	0833	0843
Payload Verification	0848	0826
LPO Pre-Engine Start	0903	0855
B-52 Crew Entry	0923	0924
Pre-Engine Start	0923	0924
Pre-Taxi	0953	0953
B-52 Taxi	1013	1013
Pre-Take Off	1028	1028
B-52 Take-Off Roll	1103	1102:48
Climb/Cruise	1108	1108
Power-Up	1140	1133
Final	1203	1206:15
Launch (PDT)	1207	1210:15:29

Launch Parameter	Planned	Actual
Altitude	42,000 ft	43,198 ft
Latitude	36.0° N	35.98974° N
Longitude	-123.0° E	-123.00946° E
Carrier A/C Speed	> Mach 0.81	Mach 0.82
Drop Cross Range Error	< 2 nm.	1,300 ft
Drop Down Range Error	< 2 nm.	400 ft

Table 3.2-2

4.0 Avionics, Telemetry, and Electrical System Performance

4.1 Avionics System Performance

For the F1 launch, all vehicle avionics were in normal flight configuration and operating properly with the exception of the bus current sensors (the calibration of which, due to a known design problem, were somewhat sensitive to power supply configuration) and some non-critical stage 1 motor temperature points (which operated properly during normal ground system testing but that were found to be intermittent during taxi and were waived as non-critical to the flight objectives). The first flight completely validated operation of the Pegasus avionics system. All avionics, functioned normally throughout ground operations, captive carry, and flight. All internal communications operated perfectly, and countdown power transitions occurred smoothly. All telemetry, with the exception of the previously mentioned current sensors and stage 1 motor temperature points operated properly. Units in the avionics section experienced temperatures ranging from 0°C to +30°C during operation. All three battery heaters cycled normally and appear to be correctly sized.

The communication link with the B-52 LPO station computer operated satisfactorily throughout the captive carry phase. Bus voltage did not vary substantially from transfer to internal power through the end of flight. The avionics batteryvoltage actually increased, possibly due to internal heating during self-discharge. Avionics battery performance indicated that actual capacity could support the vehicle avionics for about twice the current mission duration. As expected, minor transient voltage reductions could be seen on the transient buss during pyro events and thruster firings. The FTS and avionics busses showed no glitches. Thermal batteries on the Fin Actuators and Thrust Vector Controllers (TVCs) performed nominally.

The IMU correctly supplied data to the flight computer throughout captive carry and launch. Alignment, mode transitions (including the change to free inertial navigation at drop), and aided navigation occurred nominally. Fin and Thrust Vector Controllers operated as expected with the exception of minor GN&C related anomalies which will be discussed in a later section. All pyrotechnic and sequencer events occurred as scheduled and all safe and arm devices rotated successfully on the first attempt between release and stage 1 ignition. Telemetry accurately reported all arming, separation events and GN&C phase changes.

S-band vehicle telemetry, with a FM peak deviation of 70 KHz, was received strongly at Dryden throughout captive flight and ascent. While real time telemetry reception ceased shortly after the end of stage 3 burn, when the vehicle crossed the WTR horizon, reception at a downrange ARIA aircraft continued through SECS deployment, Pegsat spin-up, and the end of mission. The separate payload S-band telemetry (which shared the Pegasus S-band antennas) was received correctly throughout the flight. C-band radar tracking was successfully performed both at Dryden and WSMC throughout the captive carry phase and WSMC maintained tracking from drop through orbital insertion.

During the captive carry phase, the flight termination receivers operated correctly. The receivers remained locked on the Command Destruct Transmitter throughout the flight, with signal strength indicated 4.7 to 5.0 until release and with a subsequent gradual decline to between 3.0 and 4.0 at about 490 seconds. This performance indicates that link margins are excellent. All three Flight Termination Logic Units (FTLUs) functioned normally with no anomalies.

4.2 Guidance, Navigation, and Control (GN&C) Performance

4.2.1 Inertial Navigation System Performance

The Pegasus vehicle uses a Litton LR-81 strapdown Inertial Navigation System to determine attitude, position and velocity. During captive carry flight the LR-81 is "aided" to improve instrument error estimates and navigation states up to the point of drop using navigation updates from a high accuracy inertial measurement unit (Litton LN-39H) on the carrier aircraft. For the F1 mission the LN-39H and Pegasus IMU were aligned at approximately 0800 PDT using an initialization lat/lon/alt of the carrier aircraft's location on the ramp at NASA Ames DFRF. The observed accel bias and gyro bias terms were slightly higher than anticipated but were within mission rules for drop. The LR-81 operated nominally through the entire powered flight mission and no data dropout or failures occurred during the pyro shock events at stage separation.

4.2.2 Navigation System Performance

Pegasus has a sophisticated closed-loop guidance system that compensates for off nominal motor performance, variations in drag and lift coefficients, winds, and other factors. To meet payload constraints, the F1 guidance algorithm, was targeted to inject at a 320 nmi perigee, with no constraints on apogee (a nominally performing vehicle would achieve a 360 nmi apogee), at an inclination of 94.0°. During the F1 mission, the guidance system functioned well, reducing the stage 2/3 coast period by 11.4 seconds to compensate for actual vehicle velocity and altitude at the end of stage 2 burn (340 fps and 2.4 nm less than nominal). Real time telemetry from the Pegasus INS at the time of injection indicated that orbital injection occurred at an altitude of 305 nmi, in an orbit with a 304 nmi perigee, 348 nmi apogee, and a 93.998° inclination. The lower than desired apogee/injection altitude resulted from the lower vehicle velocity and altitude at the end of stage 2 burn.

Subsequent tracking data for Pegasus/Pegsat, however, showed that the actual achieved orbit had a 273 nmi perigee, a 370 nmi apogee, and a 94.15° inclination. Figure 4.2.2-1 shows the predicted, INS indicated and actual altitude vs time profile for the mission. Analysis indicates that the unexpectedly high INS error has two components. The majority of the error (>90%) was caused by a subtle algorithmic error in the Litton LR-81 software. This problem has been identified and will be corrected for future flights. The remaining error was due to worse than predicted gyro and accelerometer error estimates. These will be compensated for by additional tuning of the Kalman filter. The predicted 3-sigma Inertial Measurement Unit (IMU) velocity errors at burnout were 8, 23, and 15 m/s (27, 75 and 48 fps) in the North, East and Down directions respectively. Corresponding F1 actuals were 15, 20 and 95 m/s (50, 65 and 310 fps).

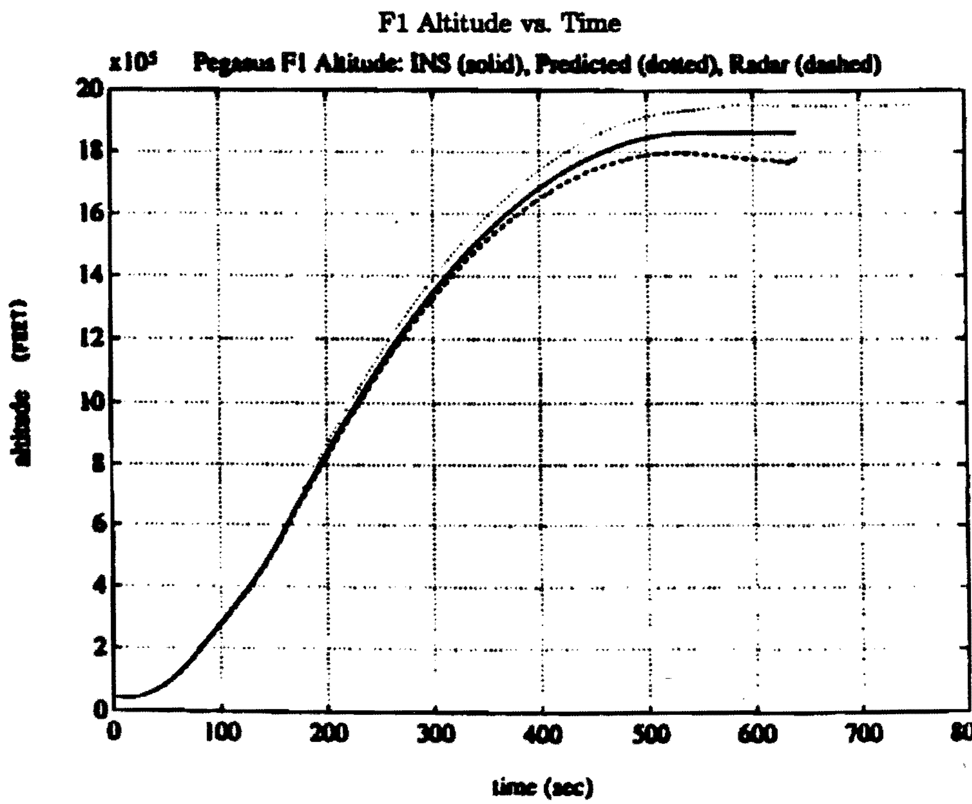

Figure 4.2.2-1

4.2.3 Attitude Control System

The attitude control system performed as expected, easily maintaining control of the vehicle throughout all phases of flight. While somewhat higher than expected pitch and roll rates were observed immediately following release from the B-52, the control system corrected for them in a smooth and well damped manner, limiting the peak roll to about 10°. Despite having a 20° angle of attack , the only evidence of transonic buffeting was a small roll rate transient, which was easily corrected. Peak vehicle pitch up was 3° greater than nominal, as required to maintain desired vertical acceleration in spite of a somewhat less than planned lift (See section 4.4). For most of the stage 1 burn, the elevator deflection angle was about 5° larger (more negative) than predicted and analysis indicates that this was due to having a center of pressure location slightly different than expected combined with a small thrust vector misalignment in pitch.

There were, however, four anomalies. The most serious anomaly occurred near the end of stage 1 aerodynamic flight. During the 37 seconds preceding stage 1 separation and to a lesser extent during the first 10 seconds after drop, a sustained 14 Hz pitch and yaw limit cycle oscillation was observed in the fin actuators. Analysis and testing indicates that this limit cycling was caused by the interaction of three elements: flexibility of the vehicle structure, mass unbalance of the fins, and the autopilot. The peak-to-peak fin deflection due to this mode was under 1° and this oscillation induced payload vertical and lateral accelerations of 1 to 1.5 g's at 14 Hz. While actuator current increased dramatically, the total electrical consumption was well within the capabilities of the fin thermal battery. Detailed testing and analysis is underway to identify an effective solution to this problem. Attitude limit cycling also occurred with the RCS thrusters for two distinct periods during the coast between the stage 2 and 3 burns. Analysis indicates that problem was caused by a vehicle structural mode to autopilot coupling. This problem has been effectively solved in simulations and can be eliminated by the addition of a simple low-pass filter in the autopilot. In addition, two minor limit cycles were observed during thrust vector control system operation during the stage 2 and 3 burns. Simulations indicate that these oscillations were caused by TVC actuator hysteresis, which will be corrected for future flights.

4.3 Propulsion System Performance

4.3.1 Motor Pressure, Impulse and Throat Erosion

Overall, the Pegasus motors performed well. All ignition events were within specification. Measured pressure time curve shape for all stages were very close to predictions, indicating near normal surface regression. The observed pressure time curve for the motors is shown in Figures 4.3.1-1, 4.3.1-2, and 4.3.1-3. First stage burn time and nozzle erosion rates were very close to predictions, although the pressure integral was 2.6% below predictions and the total impulse low by approximately 0.23%. Vehicle velocity at stage 1 burnout was 210 fps low due to lower motor impulse and increased aerodynamic drag. The achieved stage 1 performance about 9,735 fps, representing a shortfall of about 25 fps from the pre-flight prediction. The reconstructed stage 1 velocity loss due to drag was about 1055 fps, which represents a loss of about 185 fps over the pre-flight prediction. The second stage motor had a throat erosion rate 10.3% higher than predictions and a burn rate approximately 4 mil/sec less than nominal. The higher erosion rate effectively lowered the average expansion ratio, reducing total impulse by 1.28%. Overall, the achieved stage 2 performance was about 10,080 fps, representing a shortfall of about 130 fps from the pre-flight prediction. The third stage pressure integral was 0.9% below predictions its burn rate was approximately 4 mil/sec below predictions. Overall, the achieved stage 3 performance was about 10,285 fps, representing an increase of about 110 fps over pre-flight predictions. Of this increase, about 60 fps can be attributed to lower inert weight due to a higher than predicted consumption of RCS propellant during the stage 2/3 coast period as discussed in Section 4.2.3. The remainder of 50 fps is due to some combination of better than expected motor performance, inaccurately measured propellant weight, and inaccurately measured inert weight. Third stage throat erosion rate was essentially as predicted although the rate dropped somewhat due to the motor's longer burn time.

Stage 1 and 3 motor performances were well inside expected margins. The stage 2 performance was less than expected due to the significantly higher throat erosion rate. A review of manufacturing documentation indicates that this F1 throat was not as dense as assumed for the analysis and this explains the higher erosion rate. A process change is being implemented to reduce the erosion rate for future motors.

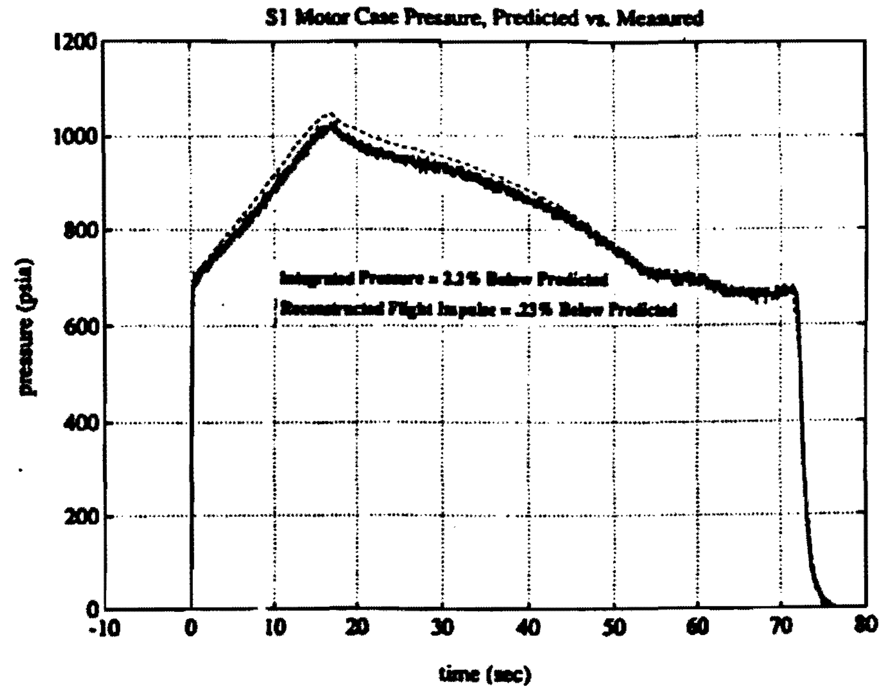

Figure 4.3.1-1

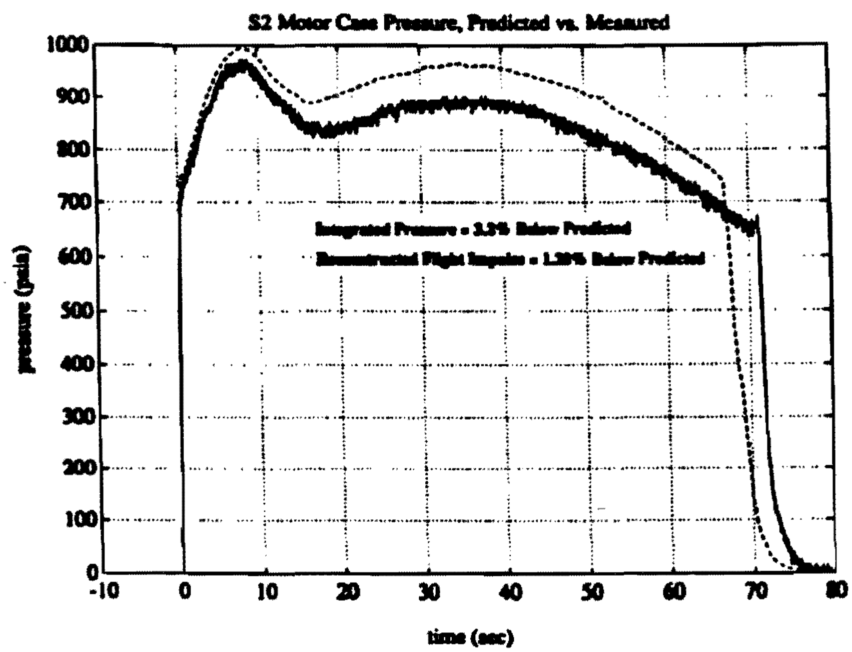

Figure 4.3.1-2

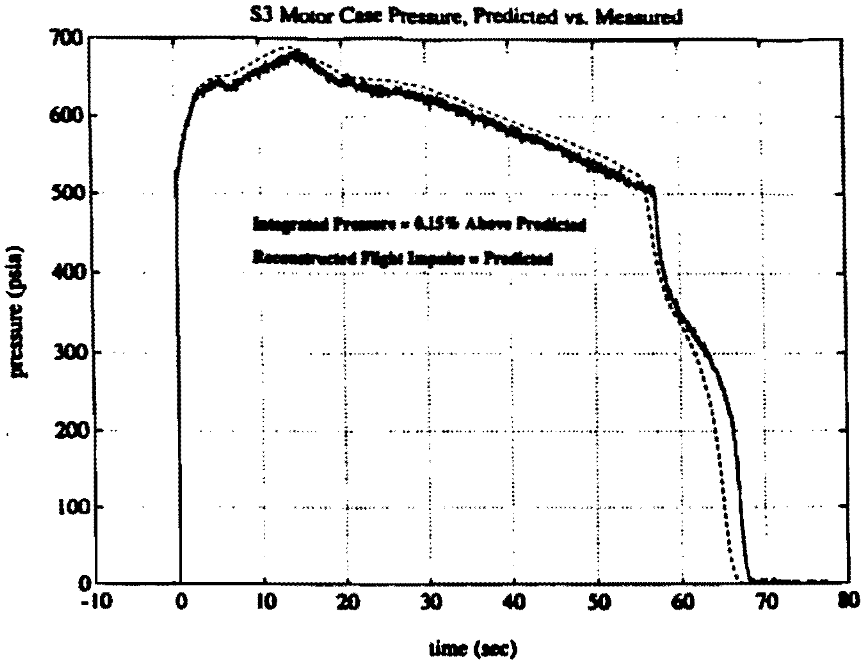

Figure 4.3.1-3

4.4 Aero-thermal Performance

As Pegasus accelerates through Mach 8, the vehicle nose, wing, and fins experience significant aerodynamic heating. The temperature profile for the fairing is shown in Figure 4.4-1. Note that the cap temperature exceeded the 177°C (350°F) telemetry cut-off level and some damage to the external surface of the nose-cap graphite/epoxy composite probably occurred. The point at which this heating is observed is significantly later than the point at which maximum aerodynamic loads occur. On future flights, additional TPS will be added to the nose cap/ogive area.

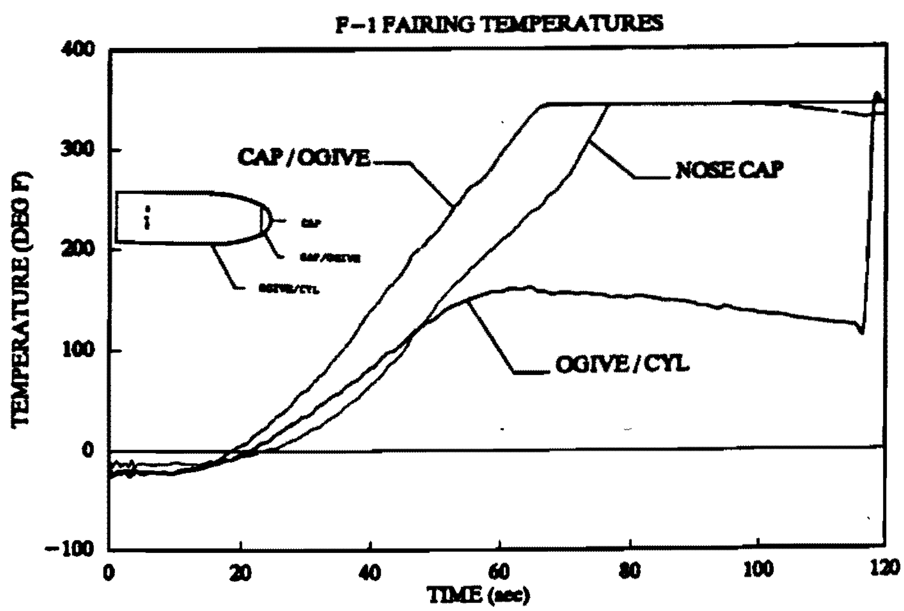

Figure 4.4-1

Thin foil temperature gages were attached to the wing surface beneath the thermal protection coating by NASA DFRF to monitor aerodynamic heating and TPS effectiveness. Figures 4.4-2 and 4.4-3 show the temperature distribution on the lower surface of the wing. As shown in the figures, the temperatures 12-16 inches aft of the leading edge exceeded the data range limit. There is evidence to indicate that the TPS thickness immediately over these gages was less than other locations on the wing and therefore makes this data suspect. Other sensors indicate that the TPS design is adequate.

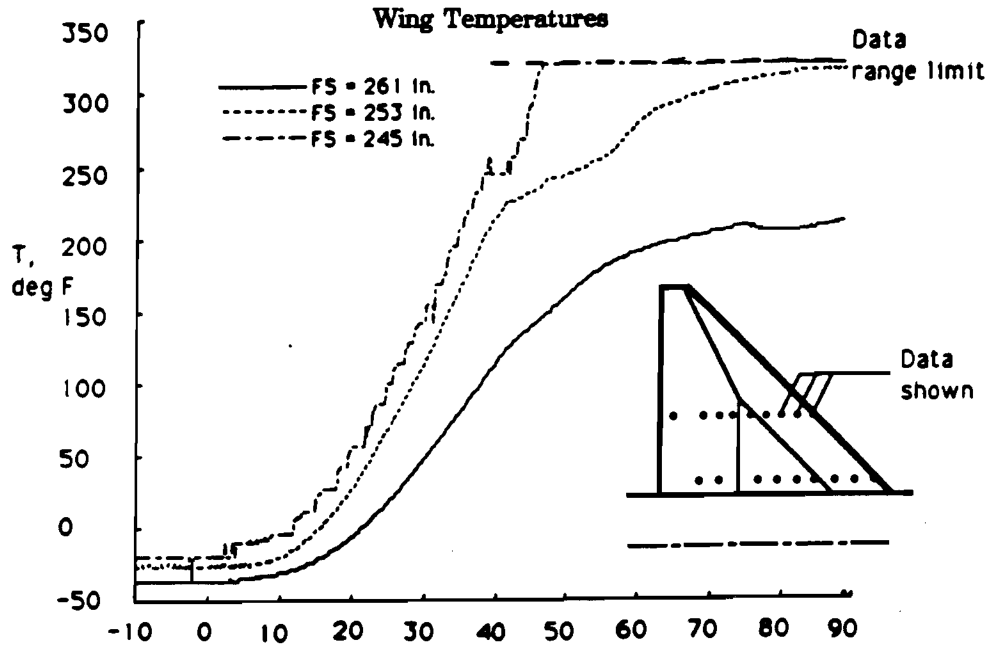

Figure 4.4-2

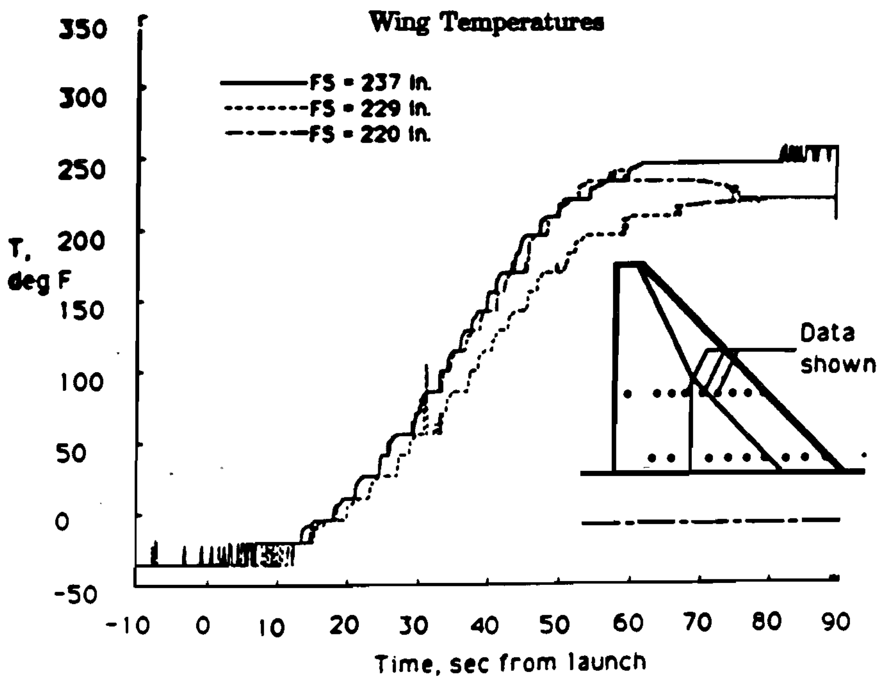

Figure 4.4-3

Rudder and elevon temperature profiles are shown in Figures 4.4-4 and 4.4-5. As shown in the plots, both fins instrumentation wires appear to have become damaged between 65 and 70 seconds into the flight. Since the wires had to be re-routed near the fin rocket nozzles for F-1, it is suspected that the wire insulation was melted as the fin rockets heated the area. Fortunately, the data that was taken prior to wire damage is sufficient to evaluate the design.

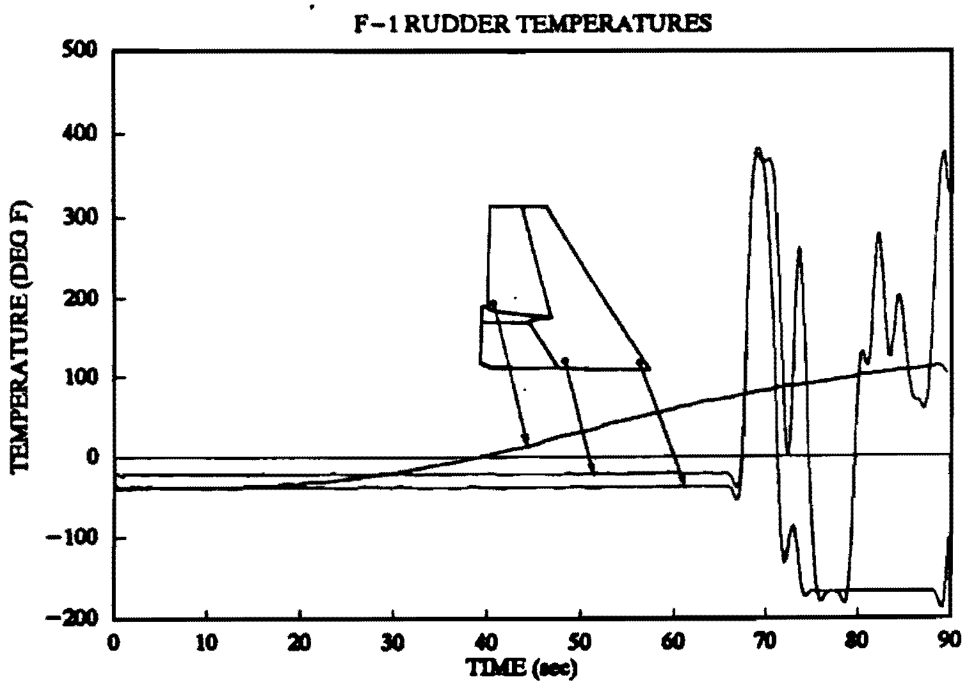

Figure 4.4-4

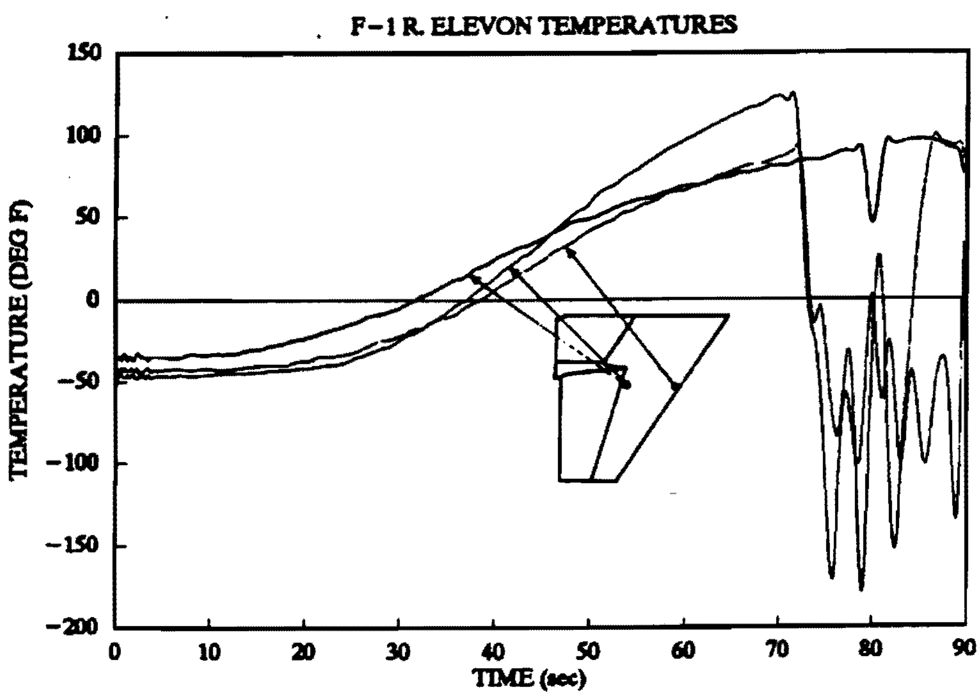

Figure 4.4-5

4.5 Structural Performance

Measured strain on the avionics structure near the payload shows the acceleration profile as well as the previously discussed oscillation at the end of stage 1 burn. The worst case line load on the structure reached about 50% of design yield. The predicted vs. actual bending and axial strains are shown in Figure 4.5-1.

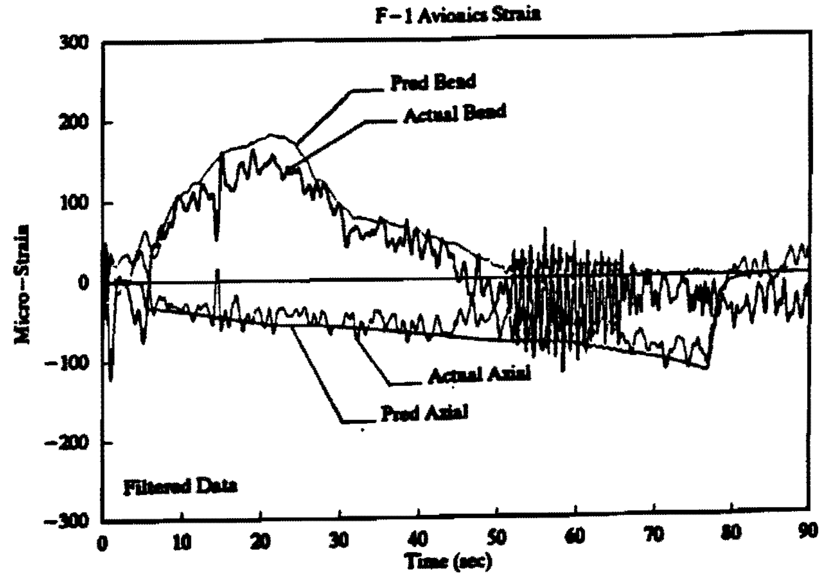

Figure 4.5-1

The wing main spar strain level was about 7% below predicted as shown in Figure 4.5-2. The lower than predicted peak strain (and lift) may be attributed to wing bending (about 8" at the tips) under load.

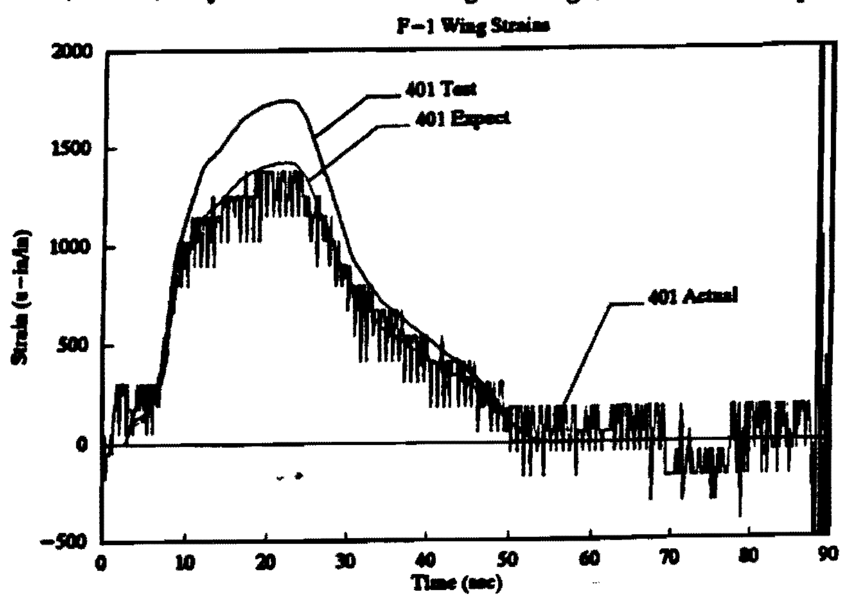

Figure 4.5-2

Peak strains on the rudder and elevons were 59% and 90% of predicted. Strain gages on the graphite composite fairing showed axial compression and bending near the predicted levels. However, the strain levels at the aft aluminum attach ring reached only 70% of predicted for bending. We speculate that the strain readings for the attach ring were affected by local thermal distortions which were not included in the preflight analysis and test. Measured strains on the interstages were all less than 5% of design limit.

4.6 Aerodynamic Performance

Pegasus was designed without wind tunnel or scale model testing. The vehicle's design was solely based on analytical methods by Nielson Engineering and Research (NEAR), using several different established missile design codes as well as computational fluid dynamic (CFD) techniques. It was felt that these techniques would produce results with uncertainties on the order of 20%. Figures 4.6-1 and 4.6-2 present the predicted vs. reconstructed drag and lift coefficients for the F1 flight. As can be seen, agreement is excellent and well within the anticipated uncertainty margins.

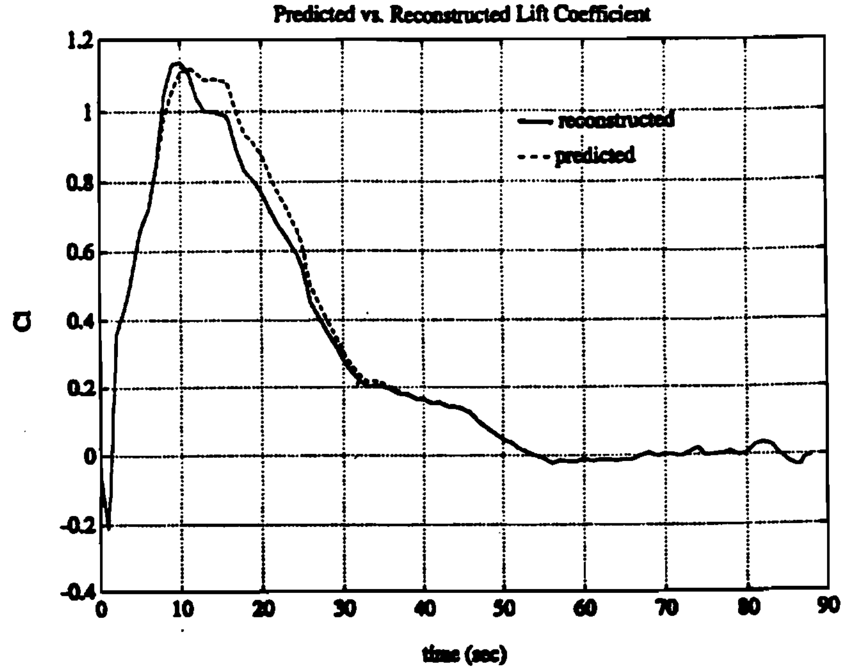

Figure 4.6-1

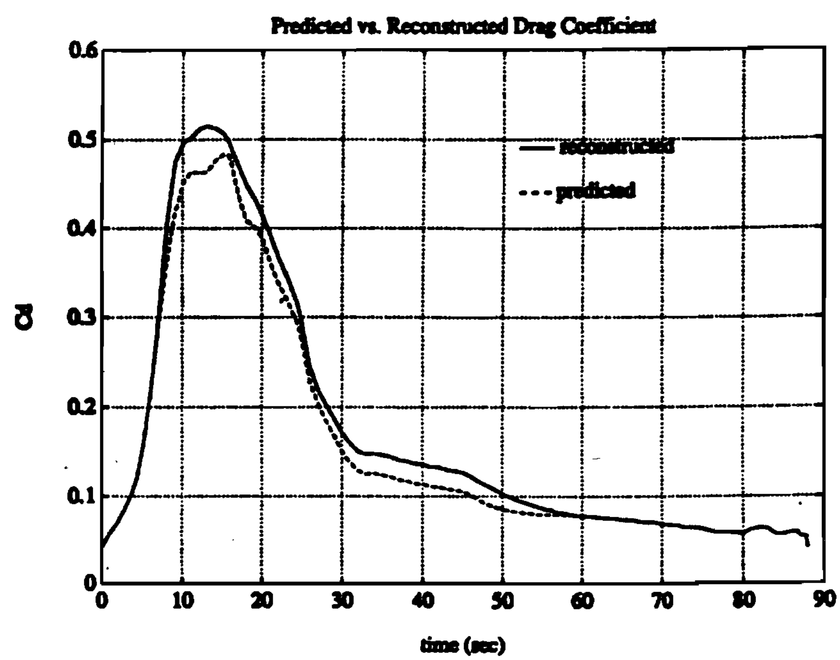

Figure 4.6-2

5.0 PAYLOAD ENVIRONMENTS

The first launch of Pegasus was instrumented to provide a detailed understanding of payload environments. The environmental design and test criteria presented below includes reduced flight 1 data along with a combination of captive carry tests, ground tests and analyses performed in the Pegasus program to date. The environmental data presented here will be augmented by a report from NASA Goddard Flight Research Center that details the data and conclusions from the flight 1 Pegsat instrumentation satellite.

5.1 Acceleration Loads

5.1.1 Captive Carry Accelerations

All captive flights, landings, and taxi accelerations at the payload interface were well within the predicted worst case bounds. Figure 5.1.1-1 shows the shock response spectrum and acceleration time history at the payload interface during landing (Taken during the first inert flight on Nov. 9, 1989). Adding 1 g to the .55 peak time history acceleration is .65 g less than the original 2.2 g worst case prediction.

Landing Acceleration

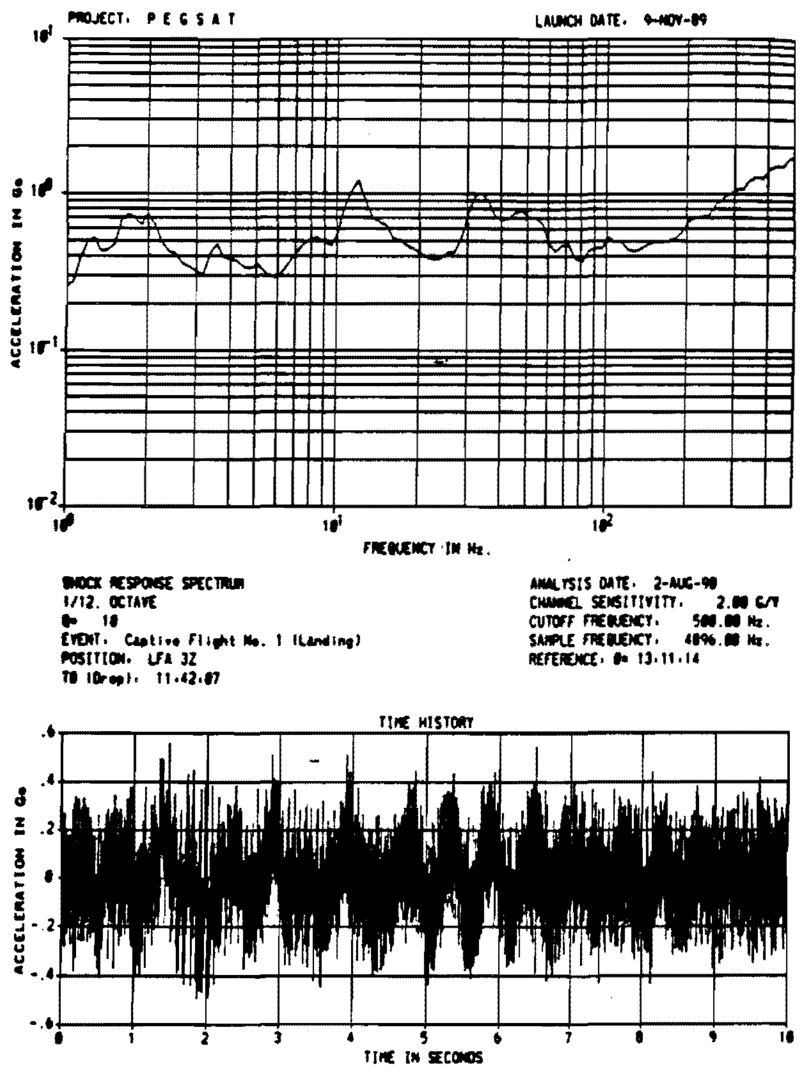

Figure 5.1.1-1

5.1.2 Powered Flight Accelerations

Figure 5.1.2-1 shows the actual lateral (Z direction) and axial acceleration as sensed by the vehicle IMU during Stage 1 burn. The oscillation seen from t = 55 seconds to burnout is due to the fin buzz problem explained in section 4.2.3. Note that the oscillation of the axial curve is probably due to cross-coupling of the lateral buzz at the IMU location and is not manifest at other sensor locations. The peak axial acceleration was approximately 7.7g's and occurred near the end of the Stage 3 burn.

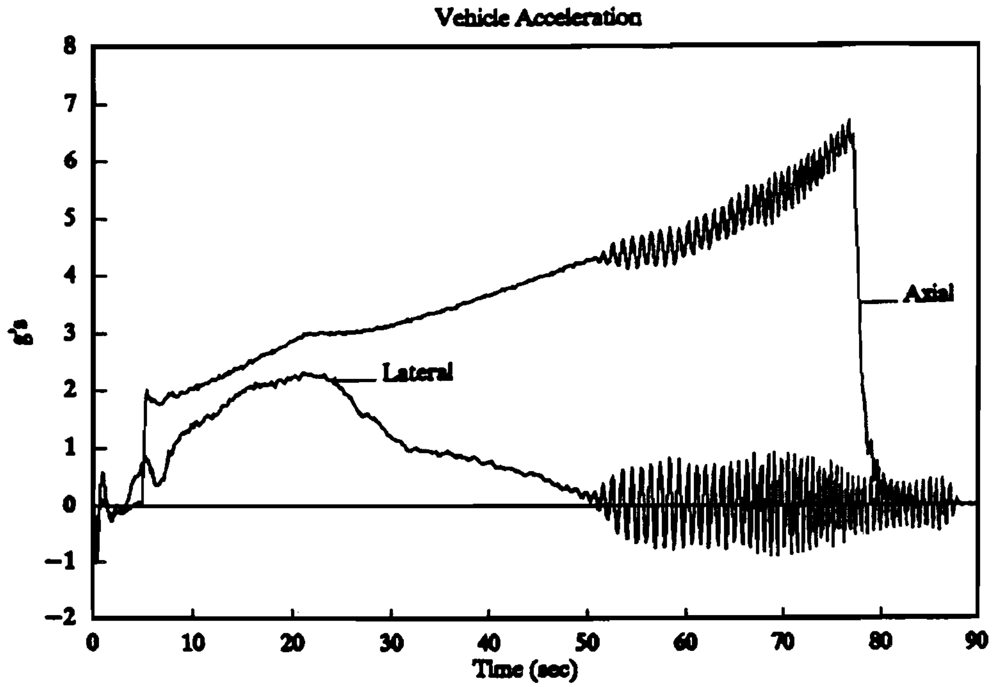

Figure 5.1.2-1

5.2 Payload Thermal Environment

On the ground, the environment inside the fairing is controlled to 21 ± 5.5°C (72.8 ± 10°F) and less than 50% relative humidity by a filtered air conditioning system. The payload temperature during flight 1, with the payload dissipating 60 W, reached a minimum temperature of -22°C (-8°F). Fairing heaters were not used for the F-1 mission. The temperature then climbed to about 5°C (41°F) at fairing separation, at which time payload temperature became dominated by solar exposure as is common in the space environment. Internal air temperature varies depending on the dissipation of the payload and usually is calculated on a mission specific basis. During captive carry, 140 W of power is available to the payload for heaters to maintain temperature limits on sensitive components.

5.3 Random Vibration

The random vibration environment at the payload interface during flight 1 is shown in Figure 5.3-1 (note the 60 and 180 Hz ground instrumentation noise spikes). This spectrum, measured at the base of the payload varied little through the three stages of powered flight. Based on ground firings, substantially higher random vibration levels were predicted from 500 to 2000 Hz for the third stage firing. The lack of ground reflection and vacuum environment significantly reduced the motor vibration in this frequency range. The flight and ground test stage 3 random vibration PSD is shown in Figure 5.3-2.

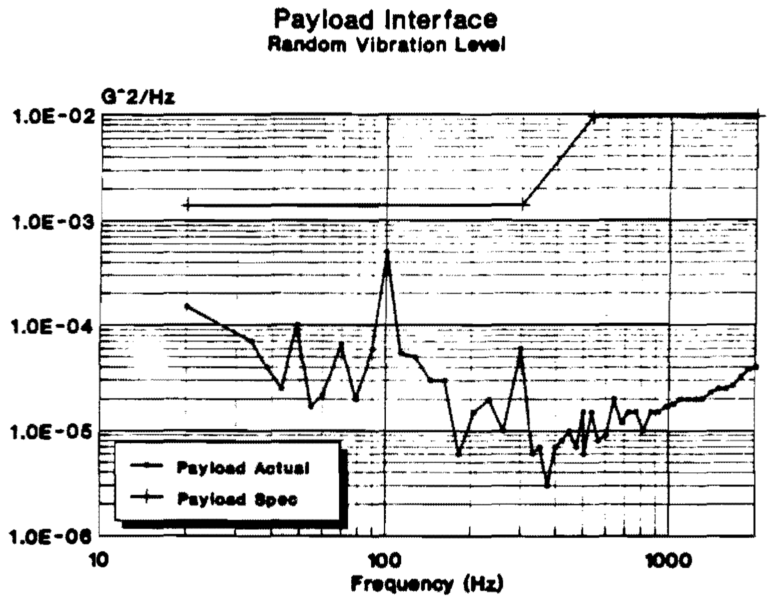

Figure 5.3-1

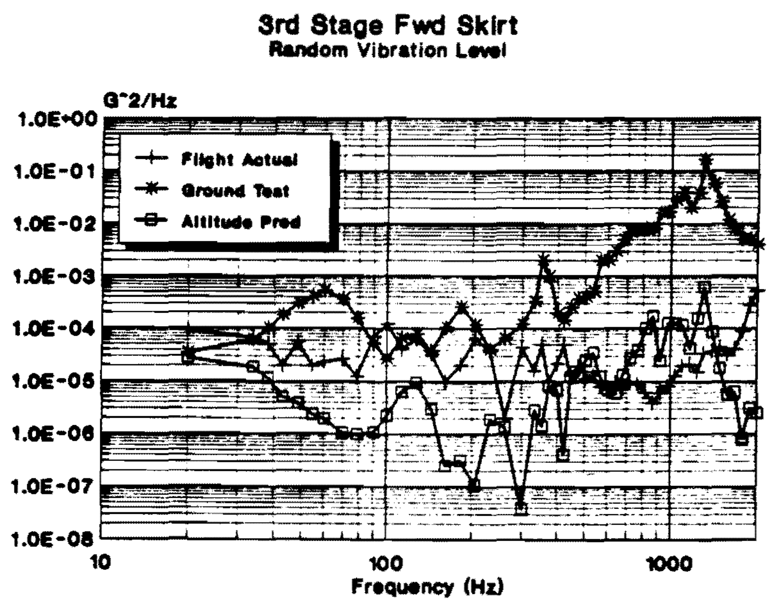

Figure 5.3-2

5.4 Pyro Shock Environment

Shock events on the Pegasus vehicle include: Drop, Stage-1 separation, interstage separation, fairing separation, and Stage-2 separation. The Pyro Shock experienced by the payload is dominated by Stage 2/3 separation. The shock response spectrum and time history shown in Figure 5.4-1 was measured during flight 1 and represents the pyro-shock at the payload side of the payload interface. The maximum SRS acceleration is 120 g's which is significantly less than the 430 g specification used for design.

Stage-2 Separation Shock

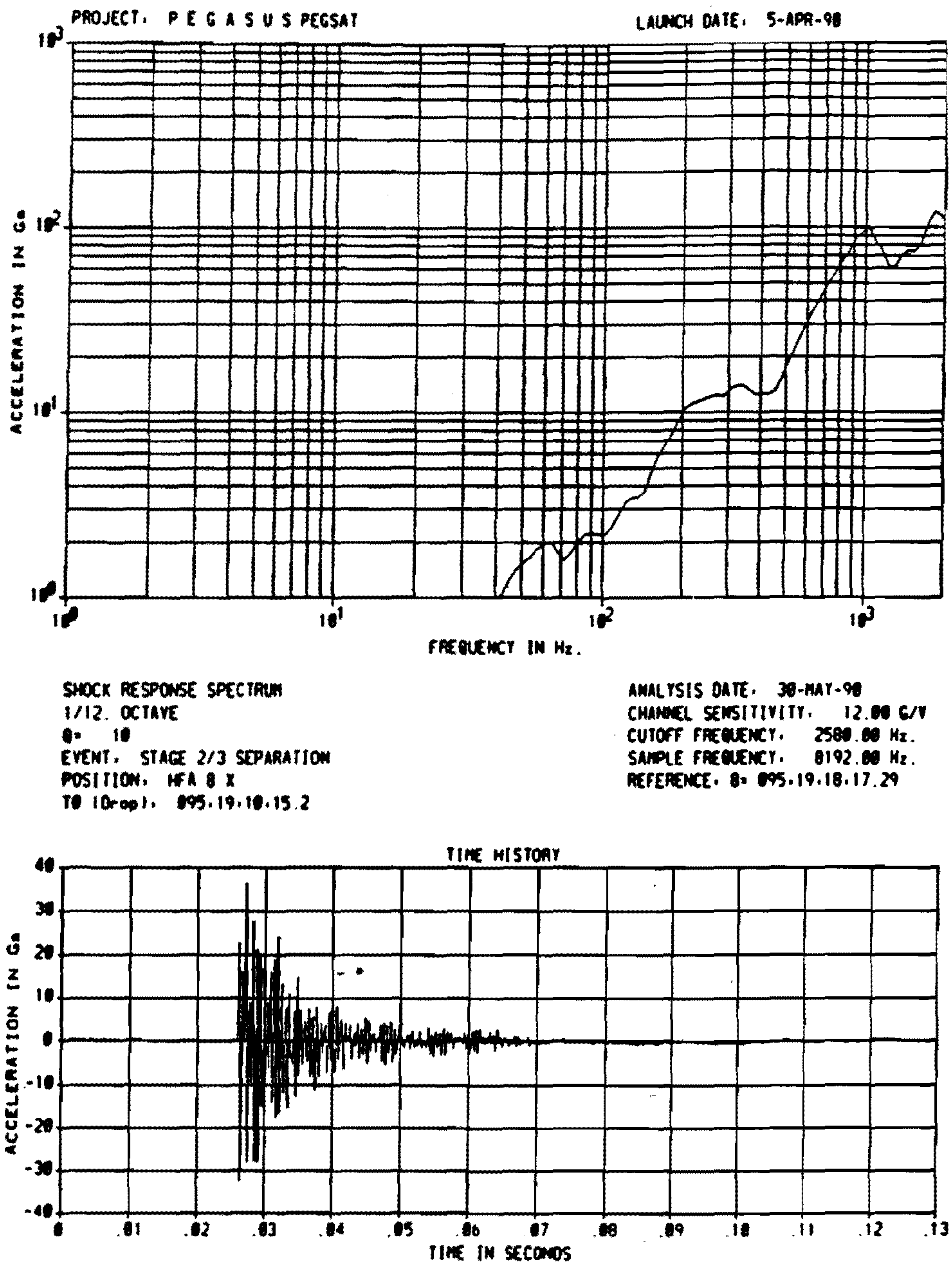

Figure 5.4-1

5.5 Acoustic

Acoustic levels inside the fairing during the Pegasus second inert captive carry tests were measured to be 132 dB during the B-52 takeoff. The levels then dropped to 124 dB or less for the remainder of the captive flight. Acoustic levels during Pegasus F-1 powered flight are below captive carry levels. Figure 5.5-1 depicts the pressure/time history for Pegasus captive and powered flight. Observed acoustic levels for Pegasus are significantly less than normally encountered with ground launch vehicles.

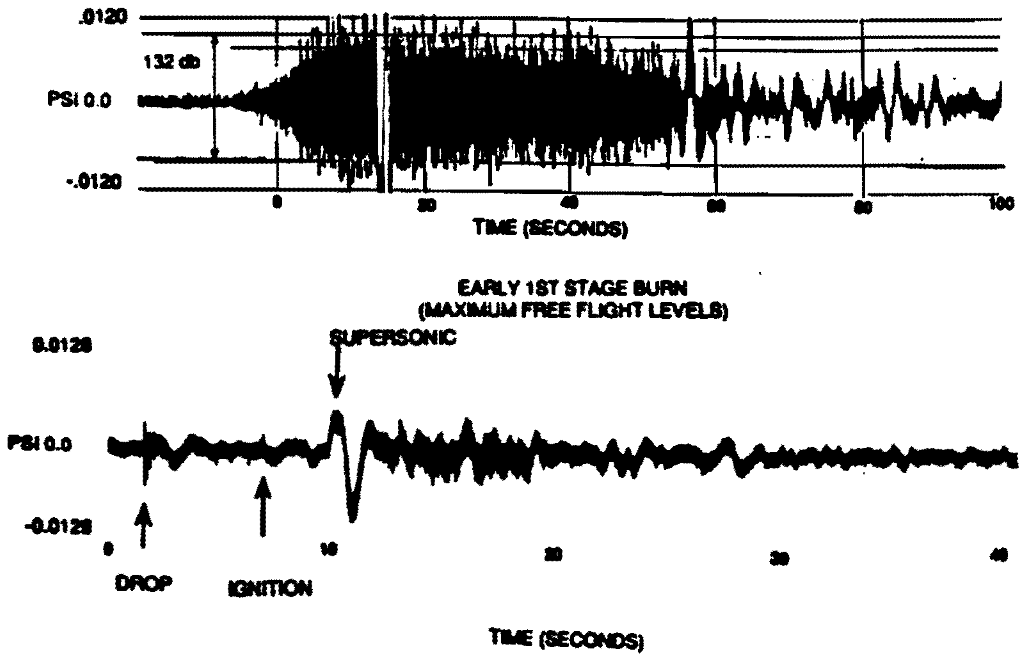

Figure 5.5-1

6.0 Conclusion

Overall, the first flight of the Pegasus launch vehicle was a great success. All aspects of the integration and launch process, from delivery of the motors and avionics to the actual launch operations went smoothly. The design of Pegasus as well as the concept of an air launched vehicle have been fully validated. Observed anomalies, included: 1) A performance shortfall of 265 fps due to lower than expected stage 1 and 2 total motor impulse and higher than predicted aerodynamic drag; 2) An IMU error that exceeded specifications caused by an error in the LR-81 IMU navigation equations.; 3) Two autopilot induced instabilities due to unexpected interaction between vehicle structural modes, the IMU and the autopilot and some limit cycling of the Stage 2 thrust vector control (TVC) actuator. On the positive side, the payload environment was better than predicted payload environments. Payload environmental data is being reviewed and reductions in the worst case random vibration and shock load specifications to levels reflecting those observed are being considered. The navigation and autopilot problems are being evaluated and will be corrected and verified during the DARPA F2 flight. A program is underway to recover the payload performance deficit.

Acknowledgment

The Pegasus team at Orbital Sciences Corporation and Hercules Aerospace would like to acknowledge the support received from NASA GSFC and NASA Ames DFRF in gathering and analyzing a portion of the data presented in this report.

Note

A complete Pegasus first flight technical report is being prepared by DARPA with support from the Air Force, NASA, Orbital Sciences and Hercules Aerospace, and should be available in the near future.

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