DEPLOYMENT SYSTEM FOR THE PEGASUS METEOROID TECHNOLOGY SATELLITE

Eugene R. Ganssle
Space Systems Division
Fairchild Hiller Corp.
Rockville, Maryland

MISSION DEFINITION

At the present time, a large gap exists in the statistical knowledge of meteoroid flux, composition, mass, and velocity. The primary mission of the Pegasus satellite is to narrow the range of uncertainty regarding flux and directionality of meteoroids in a near-earth orbit and to evaluate their hazard to manned spacecraft.

This is accomplished by deploying over 1000 square feet of capacitor-type meteoroid detector panels which include .016 inch, .008 inch, and .0015 inch thick aluminum target sheets. The meteoroid detectors are double sided to produce a total active area of over 2000 square feet and are deployed in a set of wings having a span of 96 feet. "Hit" data is stored for readout by ground command as well as being transmitted continuously by the beacon. Although no attitude control is provided on Pegasus, an attitude sensing system, consisting of five solar sensors and six earth sensors, provides attitude data on the randomly oriented satellite.

Spacecraft power is provided by solar cells equally distributed on four panels which when deployed are angularly arranged like the surfaces of a tetrahedron.

The first of the three Pegasus Meteoroid Technology Satellites was successfully launched on February 16, 1965, into an orbit having an apogee of 740 km and a perigee of 500 km. The booster was a Saturn I utilizing the Apollo configuration. This consists of a boilerplate Command Module, Service Module, and Service Module Adapter located above the Instrument Unit. During launch, the Pegasus is located within the Service Module which is separated in a longitudinal direction after orbit has been achieved (see Figure 1).

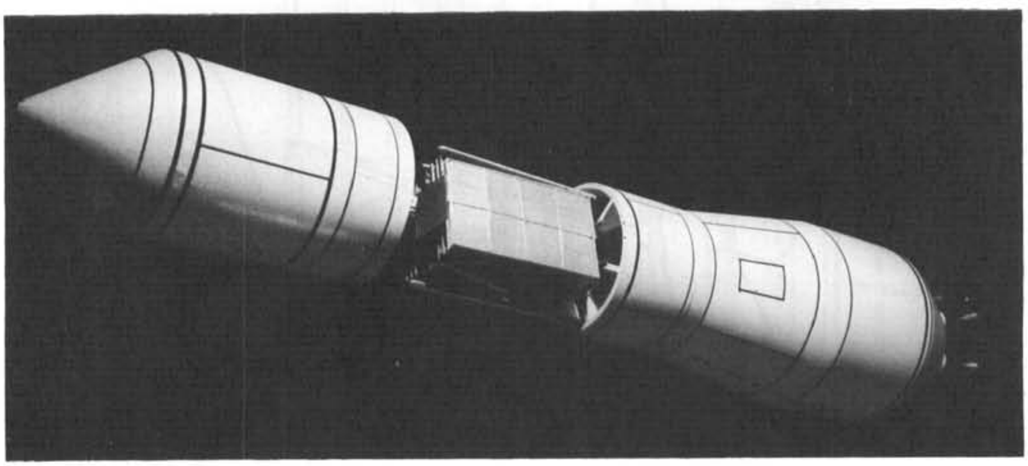

Figure 1 Apollo Service Module Separation

CONFIGURATION

The Pegasus configuration was dictated by five basic requirements:

- 2150 square feet of meteoroid detector panels
- The use of capacitor-type meteoroid detector panels
- Envelope during launch limited to nine-foot diameter and a length above the Service Module separation plane of 15 1/4 feet
- Pegasus to remain attached to the S-IV stage to insure a long-life orbit
- Pegasus weight not to exceed 3307 pounds.

Consideration of several possible configurations led to the selection of "wings" as best satisfying the requirements of the mission. The 96-foot span places much of the meteoroid detector area remote from the attached S-IV stage thereby minimizing its shadowing effect. In addition, the number of folded sections or wing frames is relatively small and the deployment system is simple and straightforward.

WING SYSTEM

The structural arrangement was governed primarily by the area requirement of the meteoroid detector panels (Figure 2). The area was distributed into 208 detector panels approximately 20 x 40 inches in area with 16 panels to a full wing frame. Twelve full wing frames and two "half" frames (eight panels each) permit the entire area to be folded within the nine-foot diameter launch envelope. Three sets of hinge fit-

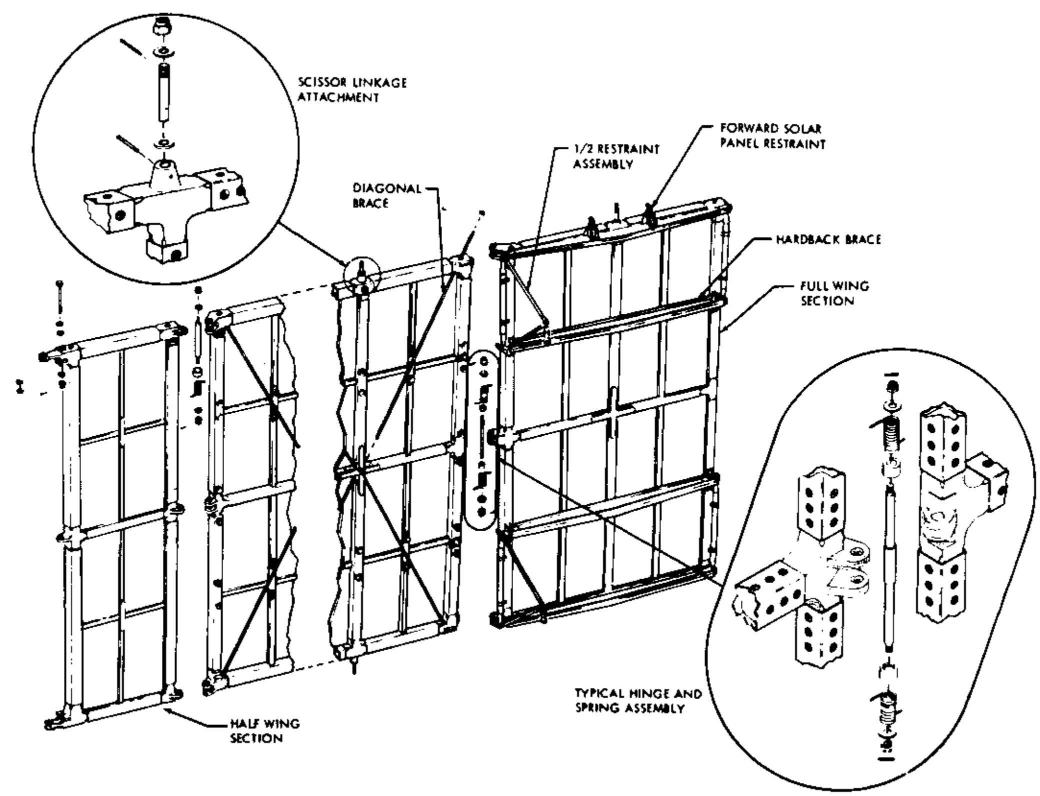

Figure 2 Wing Frames

tings join each wing frame to its neighbor and/or to the center section. The hinges alone are inadequate to support the wings during the rigors of the boost phase, so tongue-and-groove fittings were added to transmit the axial loads from wing frame to wing frame and thence to the center section. The tongues and grooves disengage immediately after initiation of wing deployment. Diagonal tension rods on each wing frame transmit shear loads across the frame and directly into the hinge fittings.

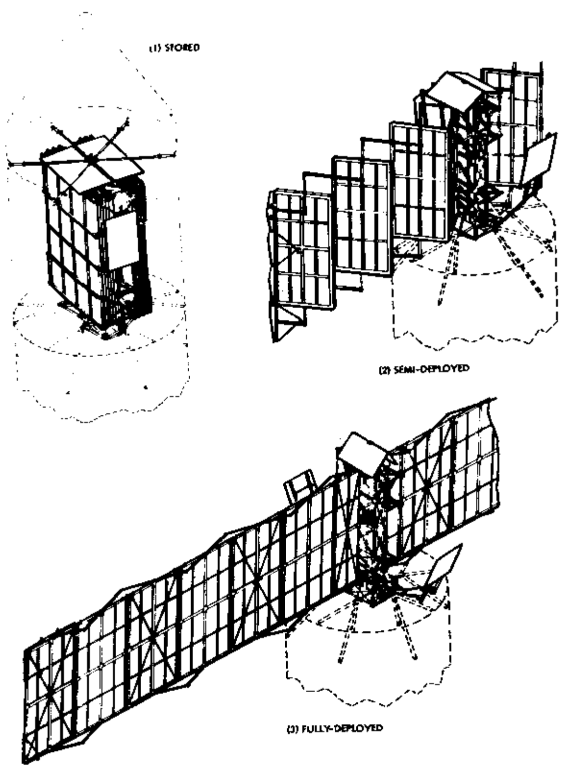

Figure 3 Deployment Sequence

The wing frames are held in a snugly stacked assembly by four sets of restraint trusses, each of which is preloaded against the center section with a load of 6000 pounds. Each pair of trusses is "hooked" together by a tongue-and-groove fitting which is held in the engaged position by a bolt and separation nut. The pretensioning is accomplished by a turnbuckle, and the load application is monitored by strain gauges on the trusses. Firing of the separation nuts allows the tongue-and-groove fittings to rotate apart, and the wings are then free to deploy (Figure 3).

Wing deployment geometry is controlled by a series of scissor links pivoted at the top and bottom centers of each wing frame. The inboard link and inboard wing frames are "geared" by pushrods and bellcranks to a torque shaft in the center section. Full wing deployment is accomplished with 63° rotation of the shaft. The primary deployment energy is provided by torsion springs

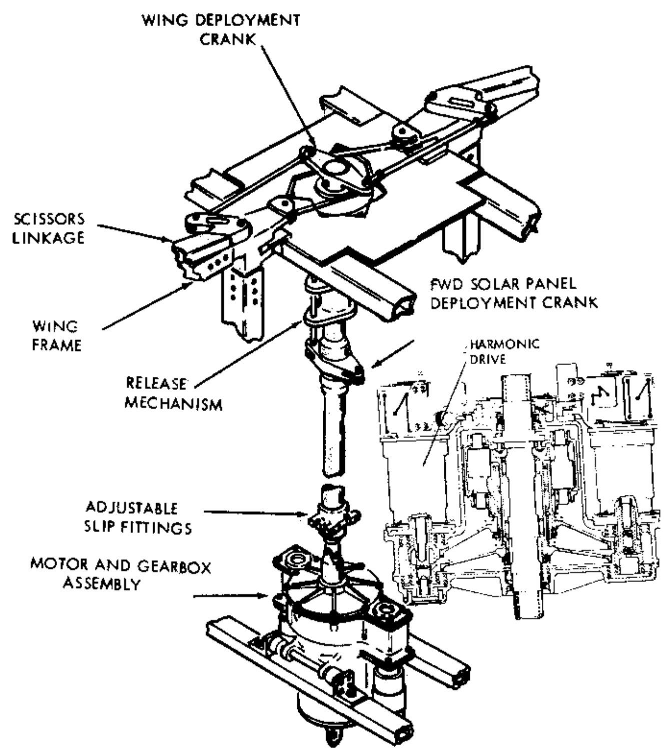

Figure 4 Deployment Drive Mechanism

located at each wing frame hinge. A backup mode is provided by a motor-driven gearbox located at the center of the torque shaft (Figure 4). Redundant motors are coupled directly to a pair of harmonic drives which deliver the power to the double ended planetary gearbox. The harmonic drive reduction is 3400 to 1 and the gearbox reduction is 6 to 1 for a total reduction of almost 21,000 to 1. The differential action of the gearbox permits deployment even with a single jammed motor or harmonic drive, and the wing spring energy is sufficient to accomplish full deployment with no motors operating. The motors contain magnetic brakes thereby providing a damping action which establishes a nominal deployment time of approximately 45 seconds.

CENTER SECTION

The function of the center section is to attach the Pegasus spacecraft to the booster and to provide a support for the wings, solar panels, and electronics (Figure 5). Its lower end is joined to the Service Module Adapter by six bolts and six shear pins equally spaced on an 80-inch diameter bolt circle. Since the Pegasus remains attached to the upper stage, no separation provisions are required at this interface.

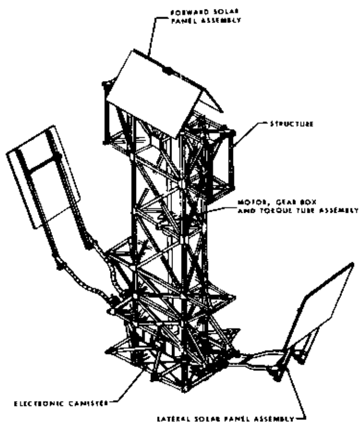

Figure 5 Center Section

The upper end of the center section requires lateral support from the Service Module. This is accomplished by six equally spaced tension rods attached to a centrally located spider fitting (Figure 6). The spider fitting has a conical lower end which engages matching fittings on the top solar panels and the center section, and is held in place by a single bolt and separation nut. Firing of this nut frees the engaged fittings and permits the tension rods and spider fitting to leave the Service Module.

SOLAR PANELS

The solar array consists of four, single faced panels which when deployed lie in the planes defined by the surfaces of a tetrahedron. They were initially intended to be deployed by linkages driven by the torque shaft thereby relating solar panel position to wing position. However, it was felt that although some degree of mission success could be achieved with partial wing deployment, full solar panel deployment is always necessary to assure adequate power. Therefore, it was determined to make the solar panel deployment as independent of the wing deployment as possible.

The top solar panels presented a special

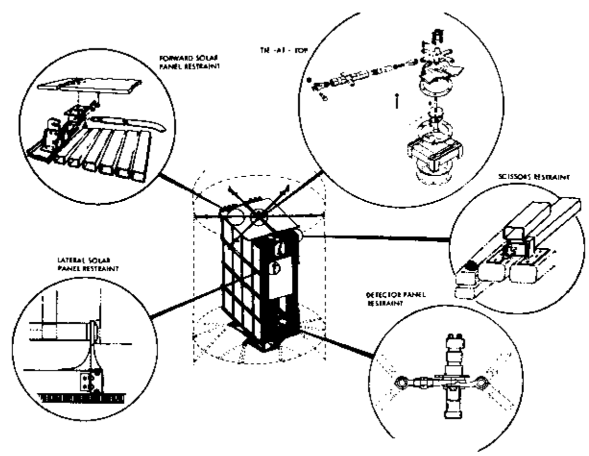

Figure 6 Restraint Systems

case because of their method of restraint and their proximity to the tension rods, which provide lateral support to the center section. During boost, these panels are restrained by the same bolt which attaches the center section to the tension rod spider fitting. Since the panels are spring-loaded at their hinges to deploy, it was necessary that their deployment be delayed after release of this bolt to preclude any possibility of a damaging impact against the tension rods. A cam-controlled locking pin mechanism was provided to key the panel support linkage to the torque shaft for the first 10° of shaft motion (Figure 7). The cam then permits the locking pin to disengage, releasing the solar panel linkage from the torque shaft and allowing spring force alone to deploy the panels to their final position. The delay between Service Module separation and initiation of torque shaft motion allows more than ample time for safe solar panel deployment.

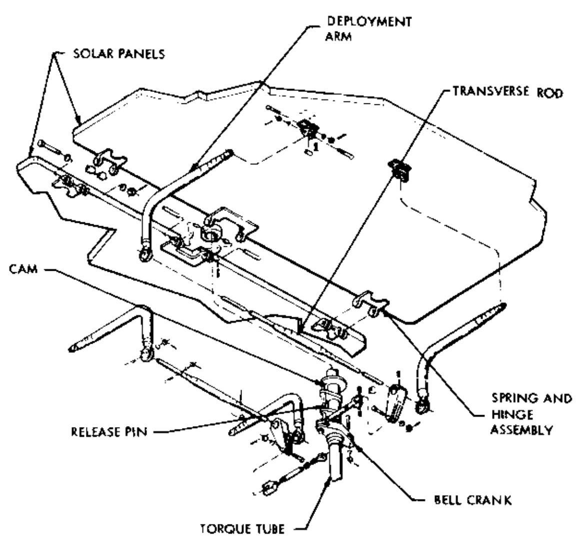

Figure 7 Top Solar Panel Assembly

Each of the lateral solar panels is restrained during the boost phase by four conical fittings which are pinched against the center section by the preloaded stack of wing frames. The initial motion of the deploying wings releases these fittings and allows the panels to be spring-driven to their final position. The panels are mounted on double-jointed arms in order to achieve the desired angular relationship and at the same time eliminate shadowing by the S-IV stage. Over-center links attached to the arms prevent the panels from rebounding and also lock them at their correct angles. The drive springs are located at these links.

SERVICE MODULE SEPARATION

Structural separation of the Service Module must occur at the plane of attachment to the Service Module Adapter and at the spider fitting which offers lateral support to the upper end of Pegasus. The Adapter attachment consists of six axial bolts and separation nuts, while the spider fitting attachment is a single bolt and separation nut as previously described. A pair of short-stroke "starter" springs surrounds each of the six bolts at the Adapter interface, but the principal ejection energy is provided by four Negator springs each having a constant load of 40 pounds and a stroke of 15 feet. Exit velocity of the Service Module is approximately five feet per second. In order to preclude any possibility of the Service Module's contacting the Pegasus during its separation, two guide rails are cantilevered from the Service Module Adapter and extend alongside the Pegasus for its full length. The rails are located to avoid subsequent deployment of wings and solar panels.

ORBITAL DEPLOYMENT SEQUENCE

The complete deployment sequence is as follows:

1. Three minutes after orbit injection, the six nuts at the base of the Service Module and the one nut tieing the top of the Pegasus to the Service Module are simultaneously fired. The 12 starter springs and the four Negator springs are then free to propel the Command Module/Service Module forward on the guide rails.

2. One minute later, a signal fires the four wing restraint separation nuts and simultaneously energizes the deployment motors.

3. Almost instantaneously, the side solar panels are released by the wings and are free to fully deploy.

4. After 10^{o} of torque shaft rotation, the top solar panels are released and are free to fully deploy.

5. Approximately 45 seconds after energizing the motors, the wings are fully deployed.

TESTING

Although the primary area of interest here is the Pegasus deployment testing, it should be mentioned that full-scale vacuum testing and vibration testing of the spacecraft in three axes was also performed. Figure 8 shows the full-scale Dynamic Test Model (DTM) on the vertical and lateral vibration test fixtures. The inputs were applied at the lower mounting plane and the stiffness of the upper lateral support was simulated by a stationary fixture. A dual C-210 shaker system was required to deliver the inputs.

The need for performing deployment tests on a 96-foot span structure which could not sustain one g loads presented obvious difficulties. In addition, deployment capability was required at three sites, the final assembly areu in Hagerstown, Maryland, the environmental testing facility in Pennsylvania, and the preparation hangar at Cape Kennedy. The early delivery required on a deployable mockup and the DTM ruled out the more exotic techniques such as air bearings, and an overhead rail system was selected (Figure 9).

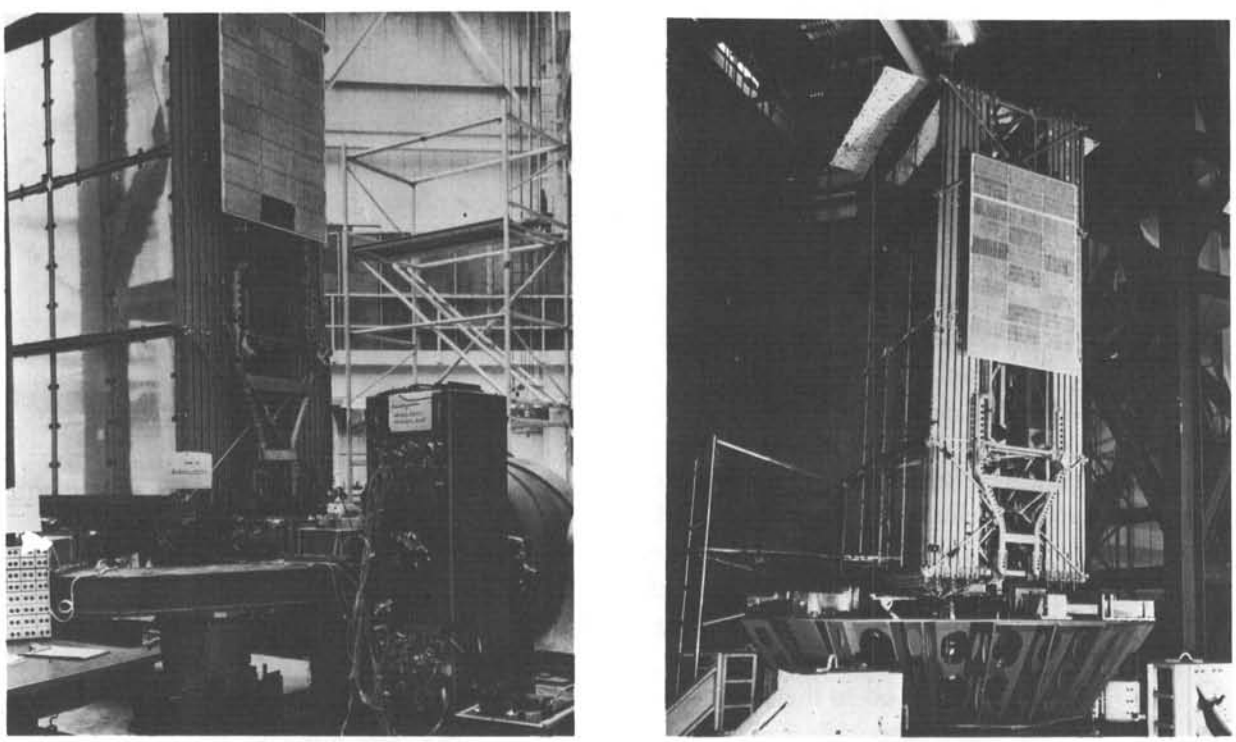

Figure 8 Dynamic Test Model on Vibration Fixture

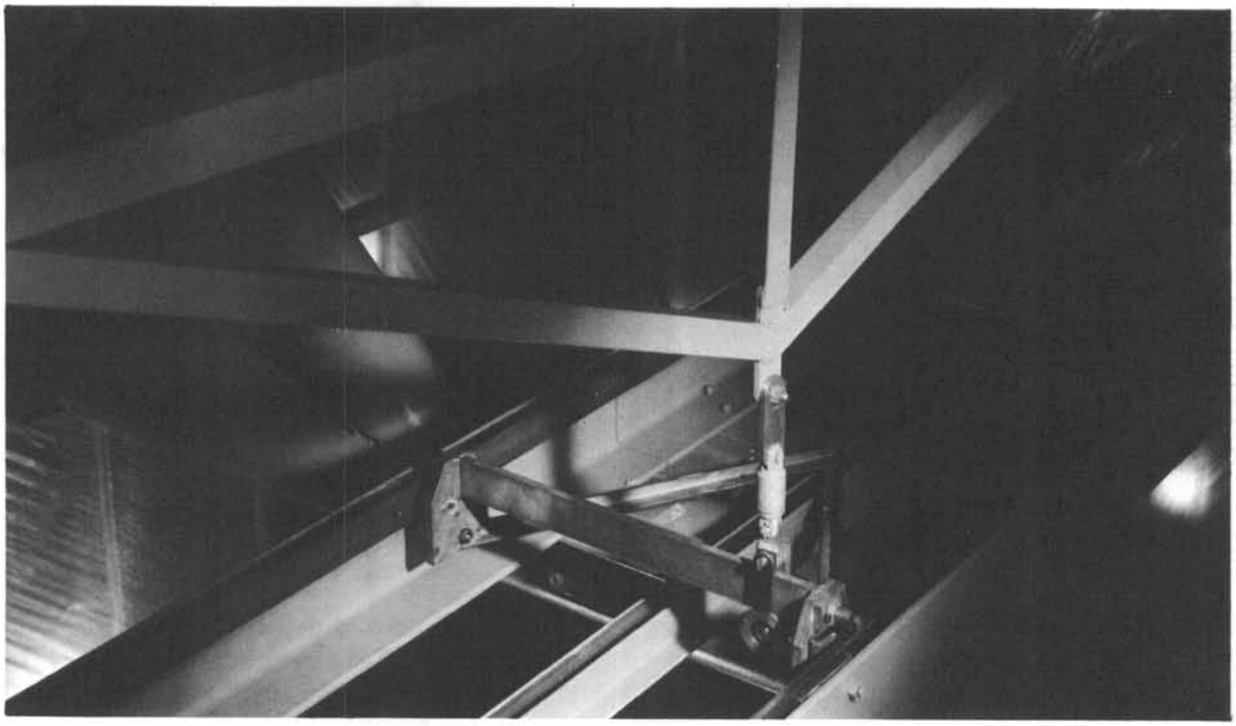

Figure 9 Photograph of Portion of Deployment Fixture - Close up of Hangers

Alternate wing frames are hung from light-weight trusses having transverse rollers at their apexes. This transverse capability is necessary because the wings do not follow a purely radial line when being deployed. The transverse roller is hung on a carriage whose rollers ride on a pair of parallel rails. Rail friction is compensated for by carefully selected weights which exert a load on the carriages in the direction of deployment by a cable/pulley system. This fixture is used for final assembly of the spacecraft as well as for deployment testing (Figures 10 and 11).

The flexibility of the Pegasus wing structure and the variations between different components call for an extensive rigging operation on the deployment fixture. The rigging includes alignment of the gearbox with the torque shaft, adjustment of the pushrods at both ends of the torque shaft, adjustment of the fixture hangers supporting the wing frames (in order to properly align the tongue-and-groove fittings), and adjustment of the brake to achieve a properly deployed wing under the various modes of motor operation. These operations account for the several deployments required even on flight spacecraft.

The DTM was subjected to over 150 deployment tests. This was a developmental phase intended to evaluate such criteria as the

- number of wing drive springs to be used
- optimum amount of preload in the wing restraint system
- effects of varying brake settings
- effects of track condition
- performance variations under simulated motor failure conditions.

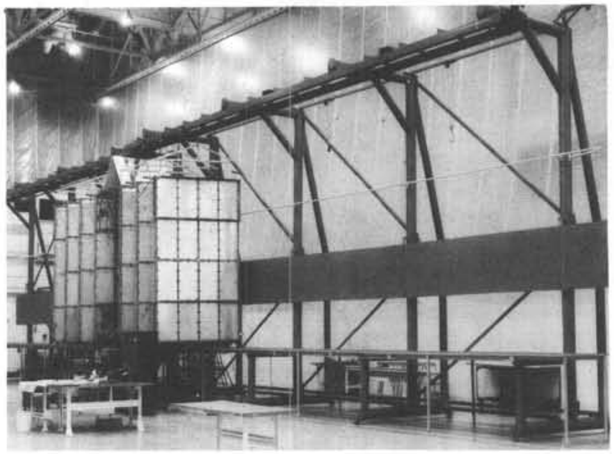

Figure 10 Pegasus Spacecraft Deploying on Zero g Fixture

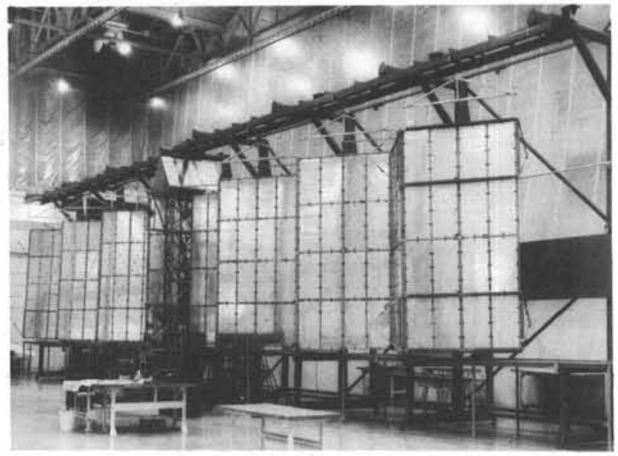

Figure 11 Pegasus Spacecraft Deploying on Zero g Fixture

Provision had been made in the design for up to four wing springs per hinge line. Deployment tests were run which included spring installations of none, one, two, three, and four springs per hinge line. Separately, and in combination with spring variation, deployments were run utilizing preloads in each of the wing restraint trusses of 0, 2000, 4000, and 6000 pounds. Hand smoothing of local rough spots in the track was required, and the technique was developed for balancing the track friction in order to more nearly simulate zero g deployment. Strain gauge monitoring of the torque shaft provided continuous values of driving and/or restraint torque in the gearbox.

Certain characteristics of the deployment system could not be determined experimentally. Response of the system to the rotational velocity of the Saturn stage could not be simulated, for example. However, analysis showed that the calculated Saturn spin rate would be a maximum of about 1.0° per second during deployment and 6.0° per second after deployment. The zero margin rates for the structure are calculated to be 3.2° and 14.15° per second, respectively. Acceleration rates show about the same safety margins.

Simulated zero g deployment of the side solar array on the spacecraft structure in "normal" position could only have been simulated by the use of a very complex, very costly, and probably unreliable support and follower system. As a reasonable compromise, the panels were tested horizontally from a simulated center section.

The panels were supported in one series of tests by an air-bearing arrangement and in another by a simple follower support which effectively canceled the g effect. Since the follower support yielded essentially identical data with that of the air bearing, and since it was more reliable and simpler, it was used for the complete analysis. A series of slow motion photographs of the deployment action shows a very slight rebound and oscillation, which quickly damped out in test. This action is apparently identical with the in-space performance of the solar panel deployment system. Overcenter links hold the panels in the deployed position. Since the forward solar panels form a "pup-tent" shape in opposition to gravity in normal tests, no special tests were required.

The DTM deployment test series proved its value in establishing basic criteria and techniques for subsequent testing on the prototype and flight spacecraft. The first flight spacecraft was deployed 18 times at the Fairchild Hiller plant in Hagerstown, Maryland, and 10 times at Cape Kennedy before delivery to the launch pad. Only two of these utilized live ordnance, the remainder using a solenoid-actuated release designed early in the DTM test series.

A television camera in the first flight Pegasus verified that the deployment system's behavior in space is a precise duplicate of its behavior on the wing deployment fixture. This result justifies the philosophy that any system designed for deployment in space must have a high confidence level established by extensive ground testing.