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Stardust Sample Return Capsule Design Experience

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The Stardust Mission has as its primary objective the acquisition and return of cometary samples, which represents the first sample return from outside of the Earth–moon system. The passage from deep space to the surface of the Earth is accomplished via a sample return capsule (SRC). The design of the capsule required a cooperative effort between Lockheed Martin, the prime contractor, and multiple NASA field centers. The coordinated design process that was undertaken to produce the Stardust SRC, in particular the re-entry-related systems, is described.

Introduction

STARDUST is the first program to attempt sample return from outside of the Earth–moon system. The mission is planned to perform a close flyby of comet Wild-2 with an advanced collection device deployed to gather dust and gasses from the comet's coma. This is accomplished by carrying a sample return capsule (SRC), with its associated particle collection devices, mounted piggyback on a host spacecraft (Fig. 1). The combined craft completes three loops of the sun, including the close flyby of Wild-2. After a seven-year interplanetary mission, the combined spacecraft returns to the vicinity of the Earth, where the SRC is released to enter and land upon the surface of the Earth. Because of the interplanetary trajectory, the SRC arrives at the Earth's atmosphere with the highest entry velocity of any Earth-entry vehicle to date, 12.9 km/s. This is compared to the Apollo lunar missions of the 1960s, which had velocities of 11.0 km/s. Combining the high entry velocity with its small size, the Stardust SRC has the highest heating of any Earth return vehicle to date, approximately 1100 W/cm^2 at the stagnation point of the heatshield. The vehicle is being built by Lockheed Martin Astronautics (LMA), in partnership with the University of Washington and the Jet Propulsion Laboratory (JPL), as part of NASA's low-cost initiative program Discovery.

Origins of Stardust

LMA (formerly Martin Marietta Astronautics) sustained a long-standing participation in planetary exploration. The successful landing of the Martin-built Viking Mars landers in 1976 provided a lead position in planetary work, which was maintained by an advanced missions core technical group. In 1994 this group investigated a large number of potential Discovery-class missions, including comet and asteroid flybys, as well as orbiters and landers to Venus and Mars. The Discovery program is an ongoing effort to produce missions every two years with a development cost of \$15 $\times$ 10^7 or less (fiscal year 1992 dollars). Reducing the overall cost of missions in a faster, cheaper, better approach paves the way to performing a larger number of varied missions than had been possible in the past. A relatively modest study led by Benton Clark (LMA) and Donald Brownlee (University of Washington) proposed a close flyby of comet Wild-2 with a low-cost spacecraft bus. Attached to the bus would be a return capsule with sample collector using advanced aerogel to retrieve cometary particles and return them to the Earth. The proposal leveraged off data from the other in-house Discovery efforts to produce a winning proposal for Stardust in the initial concept screening February 1995. A more intensive phase A feasibility study resulted in a joint proposal by the University of Washington, Martin Marietta Astronautics, and JPL led by the Principal Investigator, Brownlee, with selection of Stardust for development in November 1995.

Sample Return System Concept

Because of the strenuous cost limitations, a low-cost capsule design was proposed, which had a bare bones approach (Fig. 2). Essentially, the SRC is a thermal protection system (TPS) bound together by lightweight composite structure that surrounds the all-important sample canister. Only the simplest of electronics is required to deploy a parachute system and start a locator beacon for recovery. Stabilization of the capsule is entirely passive, relying on a spin rate induced at separation from the host spacecraft bus. One of the challenges was to assure the overall dynamic stability of the entry capsule across a wide range of flight conditions, ranging from vacuum through rarified flow, high heating and loading hypersonic regime, transonic regime, and ground impact. In addition, a new revolutionary TPS system was proposed for the heatshield of the capsule. Traditional TPSs for this application are represented by carbon phenolic, as used on the high-energy Pioneer Venus and Galileo flight systems. This form of heatshield is very effective but also quite heavy, with a density of 90 lb/ft^3. As part of the overall Discovery program technology thrust, a new heatshield material was proposed, which had been conceived at NASA Ames Research Center, phenolic-impregnated carbon ablator (PICA).[1] This ablator has a heat flux capability in excess of 1800 W/cm^2 and has a density of about 17 lb/ft^3.

Technical Partnership

Because of the wide challenges associated with the sample return phase of this mission, a NASA/contractor technical partnership was formed to tackle key issues (Fig. 3). The aerodynamic database, including overall aeroparameters and dynamic stability parameters, was tasked to NASA Langley Research Center.[2] Computation of aeroheating for the forebody and afterbody of the capsule was performed by NASA Ames Research Center[3] with additional work by NASA Langley Research Center and Lockheed Martin. The TPS development and certification included several series of arcjet tests at NASA Ames Research Center high-energy facilities. Finally, the overall stability and dynamic performance of the system was confirmed by six-degree-of-freedom (DOF) Monte Carlo simulations by NASA Langley Research Center.[4] Development of the PICA heatshield material was shared between Lockheed Martin and NASA Ames Research Center, with Fiber Materials, Inc. (FMI), performing the manufacture of the final product. Overall, the capsule design, manufacture, and mission success was the responsibility of LMA in Denver, Colorado.

Landing Site

A primary driver for the re-entry design was the requirements for achieving the desired landing site. Early on, the program baselined a dry-land recovery within the continental United States to allow rapid return of collected samples to the laboratory. This had never been done by the United States before; returning U.S.

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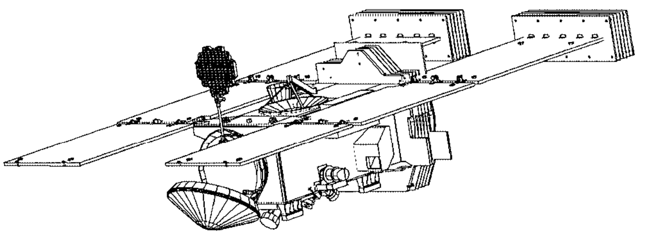

Fig. 1 Stardust flight system, spacecraft bus, and SRC.

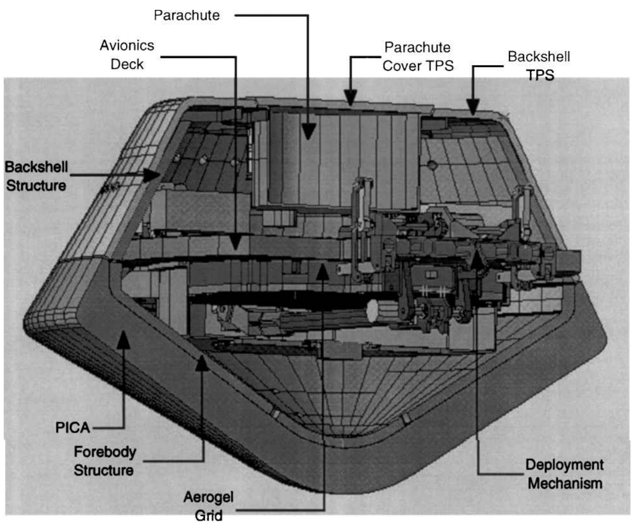

Fig. 2 Stardust SRC.

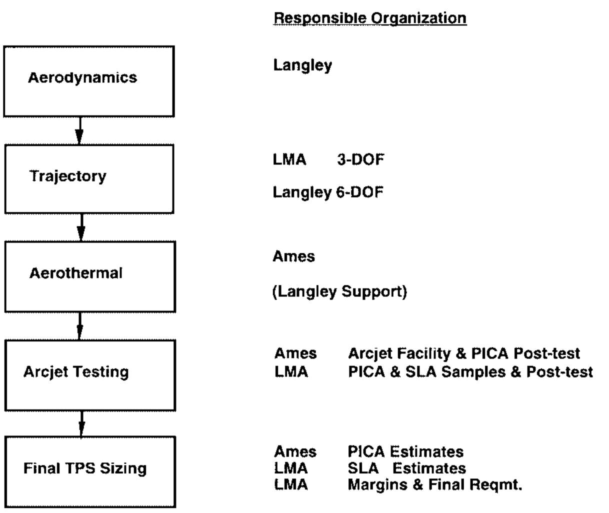

Fig. 3 Stardust SRC entry tasks, technical partnerships.

Earth-entry capsules traditionally come in over the ocean. A remote strip of Department of Defence-owned land in northwest Utah was identified [the Utah Test and Training Range (UTTR); Fig. 4] that had a large enough clear space to permit safe landings. Keeping a tight entry corridor was the key. Entry simulations performed at Denver showed that the mission was possible if the delivery accuracy's could be held to within an 0.08-deg entry flight-path angle. This angle is defined at an altitude of 125 km above the Earth (for the purposes of uniform entry definition, the Earth's equatorial radius is used for all entries). Such a small angle was a challenge that JPL navigation felt could be met. The result of these early design studies indicated that the entry heating could be held to around 1000 W/cm^2, which was well within the expected capability of PICA.

The early design-scoping effort also indicated entry loads of 40 g were required to be handled by the SRC primary structure. As a result of the work performed in designing and building the Mars Pathfinder heatshield and backshell, the re-entry group at Lockheed Martin felt confident in the ability to join the TPS systems to lightweight composite structure. This requires an appreciation of the design parameters, such as flexing, low temperature, seals, etc., as well as careful attention to manufacturing repeatability, quality assurance, and mission success philosophy. In the building of the Stardust SRC, Lockheed Martin had to act as the end user of NASA support products, in effect managing the output of NASA teams, an unusual arrangement for us, but one that worked quite well.

Table 1 Stardust design reference entry timeline

Event	Time	Comments
Capsule separation	−14,400	Entry minus 4.0 h
Entry interface inertial entry velocity	0	125-km reference altitude 12.94 km/s
Peak heating	51.1	1100-W/cm^2 reference heating
Peak loads	61.0	40 g max load
3-g Trigger point	117.1	——
Drogue deploy	132.1	32.6-in. drogue at Mach 1.4
Match 1.0	141.9	——
Main deploy	482.7	26.6-ft main at 10 kft
Touchdown	878.0	4200-ft

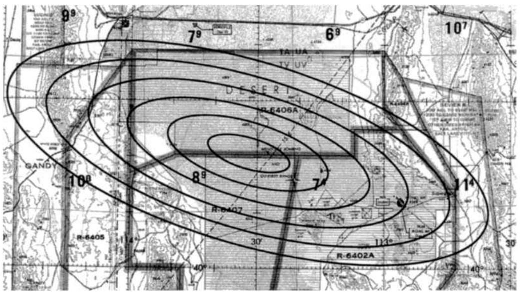

Fig. 4 Target space: UTTR.

Basic Capsule and Mission Description

The Stardust SRC is a 32-in.-diam capsule with a 60-deg half-angle sphere–cone for the forebody heatshield and a 30-deg truncated afterbody cone. The shape of the capsule was driven by packaging and stability considerations. The return payload is the comet sample return canister, a 4 by 21 in. disk, which fits into the interior. The cometary samples are to be collected by hinging open the heatshield, which simultaneously opens the canister. A boom extends a collection plate out into the gas/dust freestream, where particles are to be trapped in a material called aerogel. After collection is complete, the sample will be retracted back into the SRC by reversing the deploy process. All commanding and power is supplied by the spacecraft bus, the SRC's host spacecraft for most of the mission. Separation from the bus is accomplished by firing three pyrotechnic devices at entry minus 4 h.

This event begins the entry sequence (Table 1). At this point a separation and spin mechanism pushes the capsule away from the bus while imparting a nominal 15-rpm spin rate. The combination of spin rate and aerodynamic forces stabilizes the vehicle through the high loads and heating of entry. A drogue/main parachute combination is used to slow the capsule's final descent to UTTR. The parachute is triggered by a very simple set of electronics linked to g-switches and pressure sensors. The capsule's entry groundtrack carries it over northern Nevada and into Utah in mid-January 2006, as shown in Fig. 5. The overall capsule mass is 45.6 kg.

Overall TPS Descriptions

The forebody uses a PICA TPS. This lightweight high-energy ablator is an outgrowth of NASA Ames Research Center technology for impregnating various ceramic and carbon materials to tailor their re-entry performance. In the case of PICA, a preform of carbon foam is impregnated with phenolic to produce favorable pyrolysis and ablation characteristics. Although this material showed great promise, it was essentially a research and development material, which had never flown before and, thus, represented some risk for flight qualification. The afterbody TPS requirements were satisfied by the use of Super Light Ablator (SLA)-561V, a Lockheed Martin product. This venerable ablator was used on the Viking and Pathfinder heatshields. Although developed in the 1960s, this material is still a high performer compared to newer TPS systems and is widely used on multiple Mars missions, as well as on the Space Shuttle external tank. It is simple to apply (requiring only a day for packing), very lightweight (15 lb/ft^3), relatively inexpensive, and comes already flight-certified for Stardust-level backshell environments. Thus, the TPS systems for the proposed capsule included both the old and the new. Both TPS systems are bonded to a carbon composite structure that forms the inner shell of the entry capsule.

Heatshield TPS, PICA

Once the contract began, testing of the new PICA TPS system commenced in earnest (Table 2). To verify the performance of the TPS and structure as a system, various mechanical tests were accomplished.[5] A representative layup was subjected to bend and pull testing, which verified that the bond between the two substances was adequate. Low-temperature testing (the TPS sees minimum temperatures in flight of $-180°C$) showed no degradation of the bond. Finally, a series of impact tests showed that the PICA had shock absorbing capability during the landing event.

The majority of TPS thermal tests were done in the high-energy arcjet test facilities at NASA Ames Research Center. Representative heatshield cross sections, replicating the TPS, bondline adhesive, and composite structure, were manufactured as small (2.5–4 in. diam) test coupons, complete with thermocouple instrumentation. The test model thicknesses were full scale to the intended flight articles. The high stagnation heat flux of 1100 W/cm^2 was replicated in the Interactive Heating Facility. Margin demonstration testing produced successful tests at heating rates as high as 1500 W/cm^2. The test samples were inserted into the high-energy flow for a time sufficient to duplicate the expected flight heat load. Several aspects were looked at in these tests. Erosion of the TPS must be even and stable, and so recession rates were recorded, as well as any spallation behavior noted. Because of test facility limitations, however, the arcjet is not exercised as a TPS sizing machine. The re-entry regime has large time variations in heating, pressure, and density, but the test facility can only accommodate a fixed setting of these parameters. Thus, two separate test levels were used, and the model responses recorded to update computational material response model parameters. These analytical models predict surface recession rate, mass loss, char depth, and temperature rise at depth. Lockheed Martin used a material response code, charring material ablation, to develop a usable PICA response model for TPS sizing and margin estimates.

One of the problems encountered with the arcjet program was with the developmental nature of PICA. Early arcjet tests exhibited high surface erosion, which was attributed to incomplete curing of the material. The manufacturer, FMI, had problems producing consistent preform material in the size and shape required for a single-piece heatshield. This was fundamentally a problem of scaling up from research and development test coupons to full-size flight units. In particular, a repeatable density proved to be difficult to achieve in the scale up of PICA processing from small two-dimensional laboratory coupons to a molded three-dimensional flightworthy article. While the manufacturing problems were in work, extra off-nominal arcjet samples were tested to determine whether the range of acceptance could be expanded. Fortunately, FMI was able to solve the scaling problem and produce an acceptable flight heatshield within the defined Stardust specifications.

Afterbody TPS, SLA-561V

The backshell of Stardust is protected by SLA-561V, which is well within its tested capability. The environmental predictions showed a heating rate of 90 W/cm^2 (including uncertainty margin), where SLA has a tested upper limit in excess of 240 W/cm^2. The new features were as follows: a series of penetrations were required to attach the SRC to the spacecraft bus during flight, an electrical cable had to penetrate the afterbody for interface with the bus avionics, a vent was required to operate at heating greater than Pathfinder's, and various seals were required for the parachute lid and at the interface with PICA. All of these hardware had to be tested in various arcjet tests conducted from 1996 to 1997 (Table 3). Only one serious anomaly resulted from these tests, a burnthrough of a pushoff fitting mounted in a large 8-in.-diam pie pan, whose geometry was an attempt to simulate the corner of the backshell apex. The primary

Table 2 PICA arcjet test series

PICA	Articles	Max heating W/cm^2	Date
PICA/SLA interface test	PICA/SLA seal	35	Sept. 1996
PICA screening	Plain coupons, repairs	1500	Nov. 1996
PICA/SLA interface test 2	PICA/SLA seal	55	April 1997
PICA qualification 1	Plain, repair plugs	1500	April 1997
PICA afterbody	Edge material	56	Sept. 1997
PICA qualification 2	Plain, repair plugs	1500	Nov. 1997

Table 3 SLA-561V arcjet test series

SLA-561V	Articles	Max heating, W/cm^2	Date
Fittings	Separation fitting, chute seal, cable	75	Sept. 1996
Shear tests	Vent fitting	35	Sept. 1996
Qualification 1	Spin fitting, cable, repairs	90+	May 1997
Qualification 2	Separation fitting, spin fitting	90	Sept. 1997

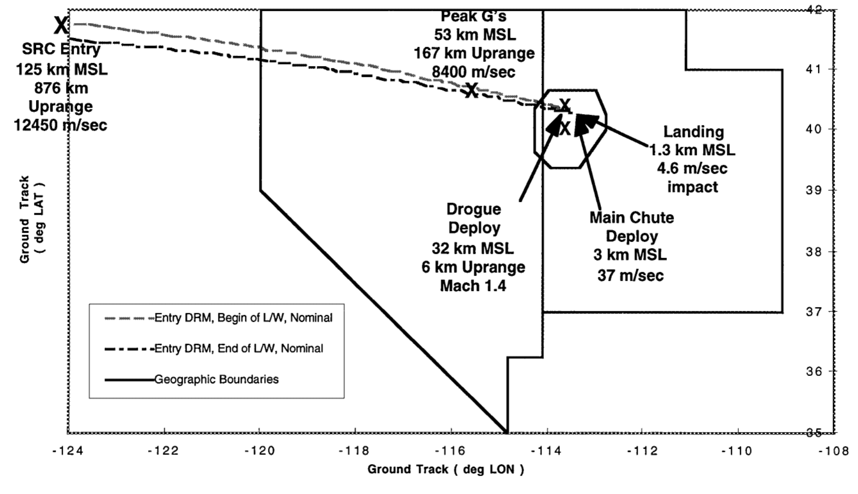

Fig. 5 Stardust entry groundtrack.

reason for this burnthrough was an excessively high test heat flux (about twice the desired rate as observed by titanium melt flow as well as large ablator erosion), caused by difficulty in predicting heat transfer for such a large object in the arcjet flow. Contributing factors to this burnthrough were a lack of fill adhesive between the metal fitting and the ablator and the presence of a large hole to act as a key slot for the SRC spinup mechanism. Minor design modifications were made, and a carefully calibrated retest passed with flying colors.

Entry Aerodynamics and Stability

One of the challenges of the SRC development program was its entry stability. Because of the limited number of components internal to the vehicle, its natural c.g. was fixed and very hard to change. Unfortunately, this c.g. location was much further aft (toward the backshell apex) than previous vehicles, which led to a pair of stability problems. The first had to do with the fairly shallow flight-path angle with which the vehicle kept its overall g-loading down. This shallow approach to the Earth, at a high velocity of 13 km/s, exposed the relatively small capsule to maximal transition flow aerodynamic forces. An assessment by NASA Langley Research Center showed that the SRC was actually unstable for much of this rarified flow region.[6] The presence of this instability region required an increase in the capsule spin rate from an original 5 to 16 rpm. Fortunately, this spin rate change was identified early in the program, when the character of the spin deployer was still being formulated, and so the design impacts were not too severe.

A second stability concern surfaced in the low-speed subsonic region. Original stability parameters for the SRC were extrapolated from dynamic aerotesting in the 1960s. The extrapolation to the fairly high c.g. location (0.35 x/d) was a large one with which there was limited confidence. Consequently a series of low-speed spin tunnel tests were performed at NASA Langley Research Center in late 1996, which confirmed our worst fears.[7] Although the capsule was statically stable, it was dynamically unstable at the desired parachute deployment conditions and would tend to tumble out of control. As a result of these tests, the parachute deploy sequence was modified to target deployment of the drogue at Mach 1.4, well above any dynamic instability region. The drogue then acted as a stabilizer as the vehicle slowed to the subsonic point of main parachute deploy.

Because the transonic database is so sparse for this aerodynamic shape, a shared effort by Lockheed Martin and NASA Langley Research Center in a series of ballistic-range tests was performed. A dozen subscale models of the SRC with the proper mass properties were fired out of a cannon at Eglin Air Force Base's Aeroballistics Research Facility and photographed as they flew down an instrumented test range (Fig. 6). The observed linear and angular motion of the models was fit to provide an improved aerodynamic database from Mach 2.0 to 1.0. In particular, the dynamic damping of the vehicle was found to be in an acceptable range. These tests confirmed the operating envelope.

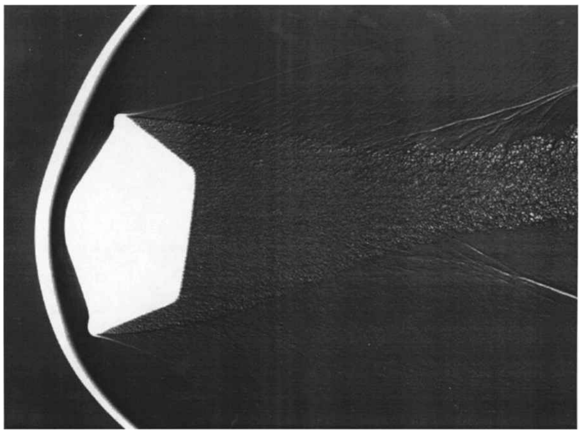

Fig. 6 Eglin ballistic-range shadowgraph image.

Six-DOF Monte Carlo Simulation

The final proof of the end-to-end performance of the entry phase was a six-DOF Monte Carlo simulation performed by NASA Langley Research Center.[4] Many individual pieces had been developed by the Lockheed Martin design team using stand-alone specifications. The deployer dynamics had been developed from computer simulations correlated to test articles. The mass properties and c.g. location of the actual capsule were an ever-evolving process as the design matured and as pieces were produced and weighed. The estimates of navigation accuracy used tracking accuracies and induced Δv from the attitude control system function. Capsule aerodynamics, including errors, came from NASA Langley Research Center computational fluid dynamics and direct simulation Monte Carlo analyses, as well as historical database from the Viking program. Shape change of the capsule due to ablation came from NASA Ames Research Center estimates of PICA surface recession. Effects of density and wind profiles are estimated with a high-fidelity Earth atmosphere model, EarthGram, maintained by NASA Marshall Space Flight Center and based on Shuttle entry data.

All of these pieces were brought together in a six-DOF (attitude and translation dynamics) simulation at NASA Langley Research Center. A total of 48 independent elements were varied in a 3300 set Monte Carlo, which simulated the entire flight of the SRC from release at entry −4 h to touchdown. The key elements are as follows. Because the parachute deploy logic was based on a simple g trigger, it was critical that this event occur on the downleg of the entry g profile. Triggering on the upleg would cause the parachute to deploy near peak heating and, thus, be destroyed. The number of false triggers was recorded and found to be less than 1/4%, a remote occurrence deemed an acceptable risk. The angle of attack at peak heating was required to be less than 10 deg to prevent excessive edge heating. The angle of attack at parachute deploy was required to be less than 30 deg for stable parachute inflation. Additionally, the drogue range was within acceptable stability and load limits by deploying between Mach 1.1 and 1.7. Finally, a most critical parameter for range safety was the landed footprint. The Monte Carlo data currently indicate a total footprint of 61 × 23 km, which fits

Table 4 Key EDL requirements for Stardust

Parameter	Six-DOF results	Requirement	Status
Entry flight-path angle	±.055 deg (3-sigma)	±.08 deg	OK
Angle of attack at max heating	±5.96 deg (max)	±10 deg	OK
Peak entry g	38.1 (max)	40	OK
Max heating	853 W/cm^2	1100 W/cm^2	OK
g-Trigger failure rate	0.13%	——	OK
Drogue deploy Mach	1.09–1.87 (max)	1.2–1.6	OK per Pioneer
Drogue deploy angle of attack	28 deg (max)	30 deg	OK
Main chute min deploy alt (AGL)	4662 ft	3400	OK
Footprint, total downrange	61.2 km (3-sigma)	83 km	OK
Footprint, total crossrange	22.6 km (3-sigma)	29 km	OK

Fig. 7 Flight SRC in final integration; parachute installation.

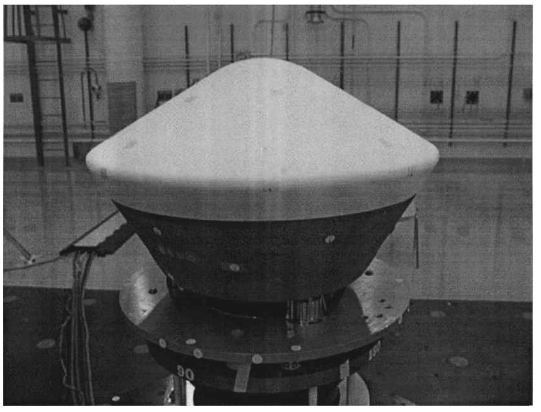

Fig. 8 Flight SRC in spin balance.

comfortably into the ~90 km UTTR range. The key entry, descent, and landing (EDL) requirements are summarized in Table 4.

In May 1998, the flight SRC moved through final integration and assembly. The Stardust spacecraft was successfully launched on Feb. 7, 1999 (Figs. 7 and 8).

Conclusions

Stardust SRC was designed and built by Lockheed Martin Astronautics with critical inputs from at least three NASA centers: NASA Langley Research Center, JPL, and NASA Ames Research Center. Key subcontractor efforts came from FMI for the PICA heatshield production as well as the Pioneer parachute (main drogue chutes). During the design of the flight vehicle, critical technical information was combined into basic design specifications for capsule shape, entry loads, thermal protection sizing, attitude stability, parachute timing, and landed footprint.

Acknowledgments

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References

[1] Tran, H., "Phenolic Impregnated Carbon Ablators (PICA) for Discovery Class Missions," AIAA Paper 96-1911, June 1996.

[2] Mitcheltree, R. A., Wilmoth, R. G., Cheatwood, F. M., Brauckmann, G. J., and Greene, F. A., "Aerodynamics of Stardust Sample Return Capsule," *Journal of Spacecraft and Rockets*, Vol. 36, No. 3, 1999, pp. 429–435.

[3] Olynick, D., Chen, Y.-K., and Tauber, M. E., "Aerothermodynamics of the Stardust Sample Return Capsule," *Journal of Spacecraft and Rockets*, Vol. 36, No. 3, 1999, pp. 442–462.

[4] Desai, P. N., Mitcheltree. R. A., and Cheatwood, F. M., "Entry Dispersion Analysis for the Stardust Comet Sample Return Capsule," *Journal of Spacecraft and Rockets*, Vol. 36, No. 3, 1999, pp. 463–469.

[5] Vellinga, J. M., "Environmental Design Considerations for Stardust," Society of Automotive Engineers, SAE Paper 97-2278, July 14, 1997.

[6] Wilmoth, R. G., Mitcheltree, R. A., and Moss, J. N., "Low-Density Aerodynamics of the Stardust Sample Return Capsule," *Journal of Spacecraft and Rockets*, Vol. 36, No. 3, 1999, pp. 436–441.

[7] Mitcheltree, R. A., "Subsonic Dynamics of the Stardust Sample Return Capsule," NASA TM-110329, March 1997.

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