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STRUCTURAL DESIGN OF THE TITAN AEROCAPTURE MISSION

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Abstract

A major goal of NASA's planetary exploration efforts is to create affordable spacecraft capable of delivering science experiments for long duration periods. To help achieve this goal the aerocapture technique for slowing a spacecraft has been investigated and appears to produce less vehicle mass then an all-propulsive mission. A conceptual spacecraft was designed and studied for an aerocapture mission to Titan, Saturn's largest moon. The spacecraft is an Orbiter/Lander combination that separates prior to aerocapture at Titan. The structural challenges faced in the design will be discussed as well as optimization sizing techniques used in the Orbiter's aeroshell structure. Design trades required to optimize the structural mass will be presented. Member sizes, concepts and material selections will be presented with descriptions of load cases and spacecraft structural configurations. Areas of concern will be highlighted for further investigation. This study involved the colaberation efforts of NASA representatives from Langley Research Center (LARC), Jet Propulsion Lab (JPL) and Ames Research Center (ARC). The concept design borrowed from existing flight hardware as much as possible.

Introduction

The structural sizing for a conceptual aerocapture spacecraft to Titan was required to obtain mass estimates based on current sizing methods. Finite element analysis (FEA) and HyperSizer™ sizing software was used to model the launch stack assemble that included the Propulsion Module (PM), the Orbiter and the Lander. The Orbiter spacecraft performs aerocapture at Titan and is designed to withstand atmospheric heating. The Lander is a sphere-cone and was considered as a concentrated mass. No aeroshell design analysis was performed on the Lander. The launch vehicle used was a Boeing Delta IV heavy with a 4 meter fairing. The spacecraft integration into a Delta IV heavy launch vehicle was achieved through trade studies focusing on mission performance necessary for an aerocapture mission. Three primary design objectives were: minimum structural mass, dynamic modes at launch were meet or exceeded and stress levels were within margin with minimal deflections. Load cases and frequency minimums at launch came from the Boeing Payload Planners Guide. Maximum loading at launch and during entry at Titan was used to design the spacecraft structure. The lowest predicted natural dynamic modes were investigated to identify any low frequency problems with the spacecraft.

The structural design used composites for the Orbiter aeroshell and a truss system to join the stack components. Modeling efforts were kept as simple as possible to shorten modifications occurring as the design progressed. HyperSizer™ sizing software was found beneficial in sizing the Orbiter's aeroshell. The software's ability to optimize composite sections without refining mesh densities and geometry was demonstrated throughout the design's progress.

Nomenclature/Abbreviations

CG	Center of Gravity
FEA	Finite Element Analysis
FEM	Finite Element Model
HGA	High Gain Antenna
NSM	Non-structural mass
PM	Propulsion Module
TPS	Thermal Protection System

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Conceptual Titan Design

The Titan aerocapture spacecraft is a stacked configuration requiring a three component stack consisting of a Propulsion Module (PM), an Orbiter and Lander. The three separate spacecraft were combined to form a launch stack capable of fitting into a 4 meter Delta IV fairing. Each vehicle must be able to separate during the Titan aerocapture mission sequence. The PM provides all thrust maneuvering to get the Orbiter and Lander near Titan. The Orbiter and Lander will then separate from the PM and then from each other. The Lander descends to the Titan surface in a Huygens type aeroshell. The Orbiter continues through the thin Titan atmosphere and begins aerocapture until achieving its mission orbit. The Orbiter was the only vehicle designed to take advantage of aerocapture. Once in orbit around Titan, the Orbiter will support an on orbit relay station for the Lander.

The conceptual spacecrafts were used as a baseline to test design and analysis methods used among the various NASA centers involved with aerocapture vehicle designs. This study focused on the weight reduction and strength requirements of the major load carrying structural members. The design attempts to maintain an axial load path direction starting with the Lander, into the Orbiter through its payload pallet and heatshield and final into the PM.

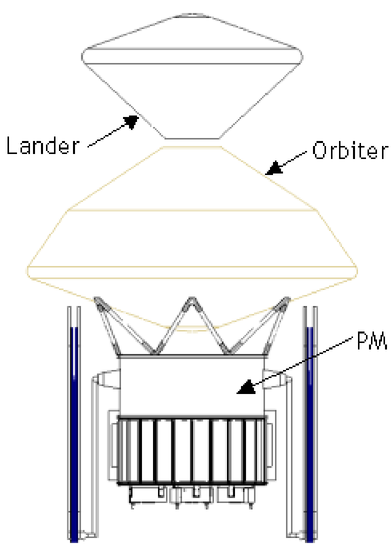

Figure 1. Launch Stack

The structural analysis performed in this study helped verify the stack arrangement and size the Orbiter aeroshell and support structure for the generic Lander. Investigating various stack arrangements showed that a truss would provide the lightest structure for supporting the Lander. A truss was also used for the PM to Orbiter adapter structure. The study used a launch load envelop for the Delta IV heavy. The Orbiter maximum diameter was set to 3.75 meters and used a heatshield cone and biconic backshell as shown in figure 1. The PM was modeled to include its stiffness contribution in determining overall stack frequency during launch. Launch loads were taken from the Boeing Delta IV Payload Designers manual (ref. 1).

Stack Configurations

The stacking sequence of the PM, Orbiter and Lander was decided upon after several trials of the three components arranged in different configurations. Each configuration had its abilities compared with each other until the stack shown in figure 1 was chosen. This arrangement was used after various stack sequences were attempted to find a stack able to meet strength, dynamics and center-of-gravity (CG) requirements. The diagrams in figure 2 represent a sample of the many stack sequences of the PM, Orbiter and Lander attempted during the design trade studies. The final stack configuration used in the design placed the Lander on top of the Orbiter.

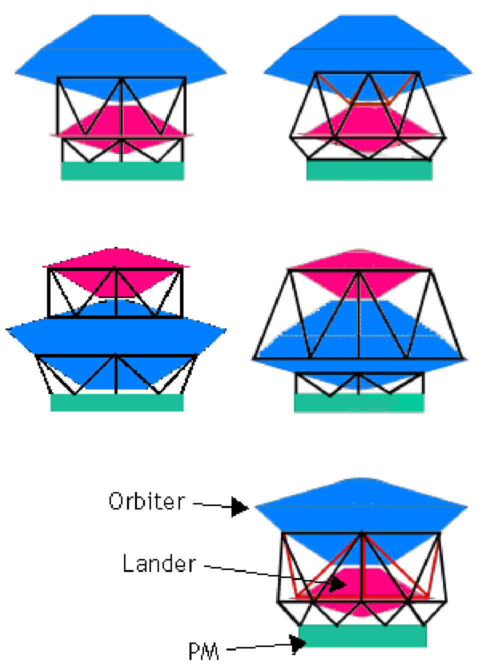

Figure 2. Trial Launch Stack Configurations

The configurations placing the Orbiter on top were dismissed because of the large mass of the Orbiter and low lateral stack bending frequency that is created during launch. The configurations produced heavy structures due to the increased stiffness requirements necessary for raising the lateral bending frequency above 10 Hz. One of the design goals was to minimize structural mass and maintain a minimum frequency of 10Hz lateral and 27Hz axial. These values were taken from the design guide in reference 3. The final configuration produced the minimal structural mass and maintained design stiffness requirements.

A generic sphere-cone Lander with a mass of 400 kg was assumed in the study and was modeled as a lumped mass with rigid connections to its outer diameter. A truss is used to create a load path from the Lander, through the Orbiter and into the PM. During the mission the Orbiter and Lander separate from the PM. The truss adapter to the PM is jettisoned with the PM and the six attachment points to the Orbiter heatshield are plugged. The method for plugging the heatshield penetrations will require further study. One possibility is to mechanically activate panels to cover the attach points. The Lander will separate from the Orbiter and head directly for the Titan surface. The upper truss supporting the Lander will then separate from the Orbiter. Aerocapture of the Orbiter will then commence at Titan.

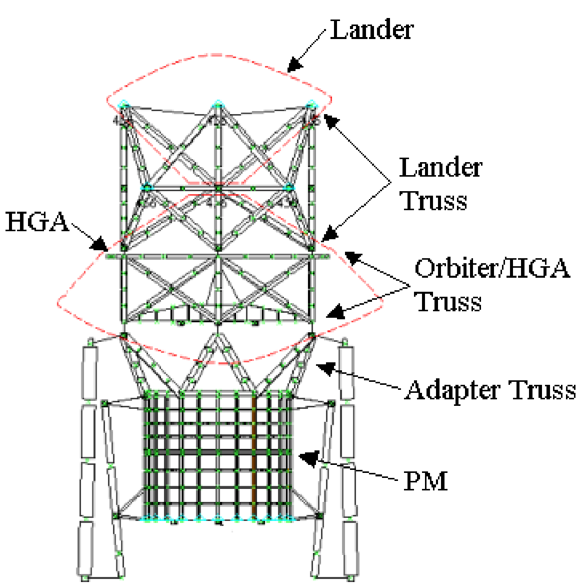

Figure 3. Stack Truss

Orbiter Aeroshell Design

The Orbiter spacecraft structure consists of the fore body heatshield, biconic backshell, cap plate and internal support structure supporting the Orbiter's payload. The largest payload components in the Orbiter are the 2.75m High Gain Antenna (HGA) and the spherical hydrazine tank. Figure 4 shows these components plus the arrangement of other internal components carried by the Orbiter. The payload deck is a hexagonal shaped aluminum honeycomb panel that extends to the aeroshell at six separation points. All payload deck items are modeled as concentrated masses with their CG offset made using rigid elements as required. The Orbiter aeroshell is supported by the adapter truss attached to the heatshield. The Lander is supported by a tube truss system that penetrates the Orbiter backshell at four locations. The load path continues straight down through an internal structure that also supports the HGA.

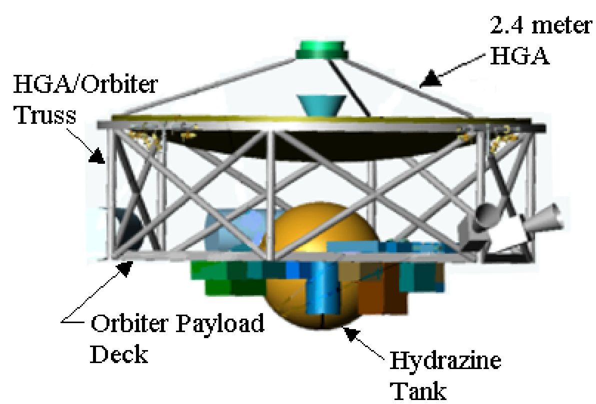

Figure 4. Orbiter HGA and payload deck
(Lander not shown)

A payload pallet ring is used to transfer the loads coming from the Lander as well as the payload deck to six hard points on the Orbiter's heatshield. The six hard points are equally spaced around the perimeter of the payload ring and represent penetrations through the heatshield. The Orbiter aeroshell FEM is pinned at the six hard points. Concentrated masses were used to model the internal payloads along with rigid elements to properly locate CG's.

The load contribution from the Lander and Orbiter is carried into an adapter truss through the six points on the heat shield. The adapter truss tapers down to fit the front of the PM completing the load path. The choice of allowing the load path to continue through the heat shield raises obvious concerns with the thermal protection system (TPS) being compromised. The mass of the Orbiter was reduced by not using its aeroshell to support the Lander mass. The stack concept relied on keeping load paths running axially through a tubular space truss. The six penetrations in the heatshield were accepted in this study and referred to as a detail requiring further investigation.

The Orbiter was analyzed using a combination of nastran finite element analysis (fea) and Hypersizer™ commercial sizing and optimization software. In order to utilize Hypersizer™, a coarse grid nastran fea was created with all non-structural masses (NSM) and mission loads of interest. The major NSM contribution was the heatshield and backshell TPS. Other NSM included Orbiter payload and aeroshell separation mechanisms as well as allocations for the six attach points through the heatshield. Figure 5 shows an exploded view of the FEM used to create a HyperSizer™ model of the Orbiter's aeroshell.

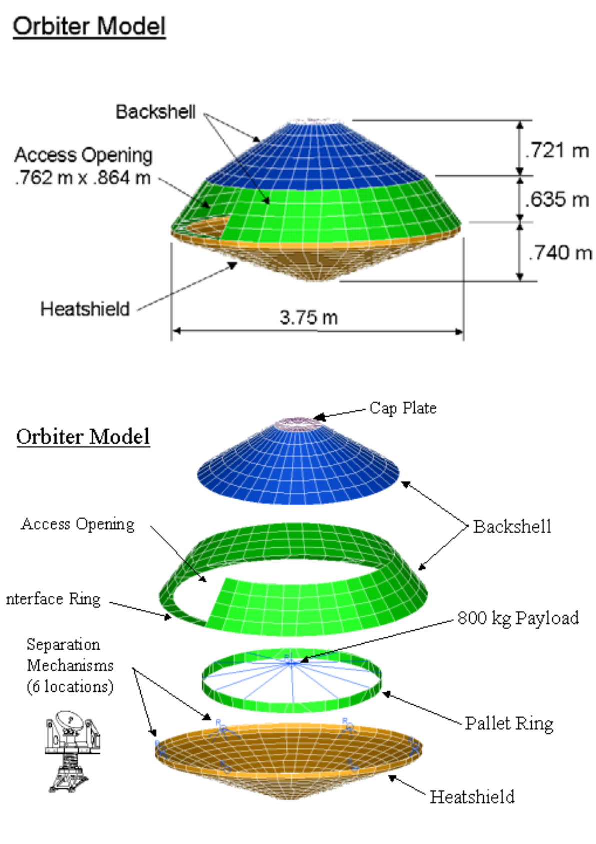

Figure 5. Orbiter Aeroshell FEM

The mesh size was kept coarse, however included enough detailing of the aeroshell geometry to accurately calculate the element forces required in Hypersizer™. An opening in the backshell was modeled to represent an access panel to the Orbiter payload. No attempt was made to stiffen the opening by modeling the door or method of attachment to the backshell. The mass of the door was treated as a NSM with smearing at the nodes. The nastran finite element model (FEM) of the Orbiter was created only with basic nastran elements: quad4, tria3, conm2 and rbe2's. These elements are easily supported by Hypersizer™ and were imported to form a Hypersizer™ model of the Orbiter aeroshell.

Several model configurations and load cases were used to find worse case conditions on the Orbiter aeroshell. The first configuration studied was the Orbiter in the launch mode with accelerations based on the Delta IV payload guide (ref. 3). The loads used were 3 g's lateral and 7 g's axial. The combined loads were the absolute maximums in the Delta IV launch load envelope. No assessment of acoustic energy and shock spectra on the total payload was attempted during the design.

The next load cases investigated were aerocapture entry loads of the Orbiter through the Titan atmosphere. Two load cases were investigated for different entry velocities. A 6.5 km/s and 10 km/s entry load cases were analyzed. The peak aero loads were obtained from CFD analysis based on the two trajectory cases. The loads were assumed to act normal to the heatshield and evenly distributed. The following loads were used:
6.5 km/s Entry loads: 4 G axial with 3146 Pa on heatshield
10 km/s Entry loads: 10.3 G axial with 8997 Pa on heatshield

The launch load forces were imported into Hypersizer™ to start sizing of the aeroshell. The Orbiter's Hypersizer™ model was divided into different components for sizing. The approach was to size the heatshield as one uniform thickness as well as the lower and upper backshell and cap plate. The optimization concepts used were: honeycomb core with face sheets, blade stiffened panels and isogrids. Each concept had dimension variables that were used to find the optimal aeroshell geometry such as: blade separation distances, core and face sheet thickness, blade depths and thickness.

The TPS non-structural mass was added inside Hypersizer™. The mass could easily be changed on one of Hypersizer's™ user input screens. This feature of the software was helpful for modifying the model to suit different TPS trial materials and thickness. The final TPS material used in the design of the Orbiter was TUFROC on the heatshield with an aerial density of 1.181 g/cm². This density was held constant over the heatshield. The backshell and cap plate used SLA with an aerial density of .187 g/cm². The density was also constant over both surfaces. The TPS masses were exported as nastran conm2's and evenly distributed at the element nodes.

Why use HyperSizer?

Spacecraft structures contain complex geometry and load distributions that are highly indeterminate and historically demanded finite element analysis (FEA) to solve. Performing structural analysis and sizing optimization has required large degree-of-freedom models with long solution run times. A software product called HyperSizer™ can help simplify structural sizing and reduce design analysis time. HyperSizer™ helps to automate the sizing of structures by reducing launch acceleration and entry loads into force and moment components on panels and beams throughout the spacecraft. The sizing includes finding the optimal material combinations, panel and beam dimensions such as thickness, depths and spacing. The code is not a finite element analysis or computer aided design package. HyperSizer™ adds to the capabilities of these tools to allow the engineer to design, size and perform detailed failure analysis on a complete vehicle.

The Orbiter's aeroshell design was used to demonstrate the software's composite design capability and use in conceptual designs. A new mass-sizing tool is under development for planetary spacecraft at LARC. The tool will have the ability to link spreadsheet user inputs into HyperSizer™ for composite structure sizing. This will greatly improve structural mass estimates and lessen analysis time usually dominated by FEM creation and modifications.

Optimization Capabilities

Optimization capabilities within HyperSizer™ include finding minimum weight panel or beam concepts, material selections, cross sectional dimensions, thickness and lay-ups from a database of 50 different stiffened and sandwich panel designs as well as a database of composite, metallic, honeycomb and foam materials. The database is used to define structural families inside HyperSizer™. The structural families include definitions for panels and beams such as the "uniaxial stiffened family", the "unstiffened plate/sandwich family" and the "open beam family".

The panels shown in figure 6 below represent some of the typical families of structural panels available in HyperSizer™. The panels may be stiffened with typical aerospace shapes or corrugated. The grid-stiffened family of panels has recently been added to HyperSizer™. This allows for the sizing optimization of isogrids, orthogrids and general grid rib-stiffened panel concepts with either isotropic or composite materials

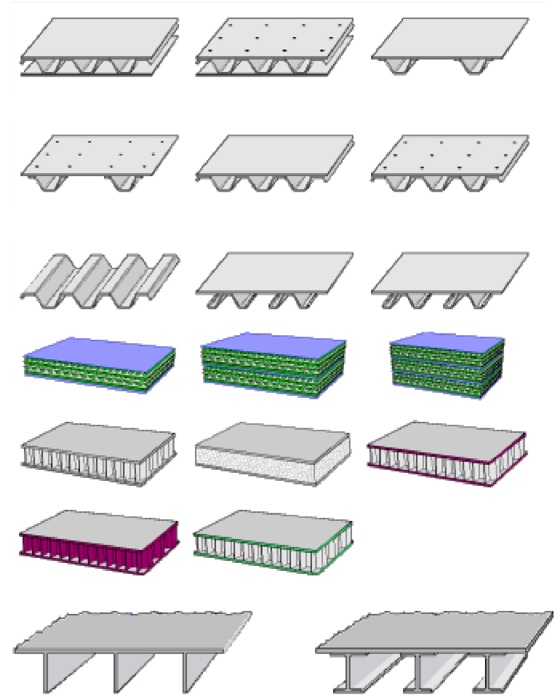

Figure 6. Typical HyperSizer™ panels

Orbiter payload deck and HGA support

The Orbiter payload deck and HGA support were modeled with plate and beam elements. The payload deck was a flat hexagonal shaped plate with a large hole cutout for a hydrazine tank carried into aerocapture orbit. The six corners of the hexagon platform extend to the outer diameter of the Orbiter. The platform lies in the same plane as the backshell/heatshield separation plane. The HGA is supported by an internal truss that also connects to the Lander truss. Loads from the Lander travel through the HGA support and into the payload ring located below the payload deck. Loads are then transferred through the heatshield structure and into the Orbiter/PM adapter.

Propulsion Module

The PM was modeled as shown in figure 7. The bulk of the module was made of aluminum channels and distributed lumped masses. Two solar arrays were also modeled and appear on the module sides. The solar arrays were modeled as beams having an approximate stiffness of the array panels. The modeling effort attempted to accurately capture the correct stiffness and mass of the module without fine detailing of the meshes. A concentrated mass with rigid elements was used for the propellant tank. Support structure for the tank was also provided with a truss system tying into a ring frame. A cylindrical wall stiffened with beam elements form the main thrust tube. The aft end of the tube was pinned with the forward end attaching to the Orbiter/PM adapter truss. The PM was modeled to help determine overall stack frequency at launch. By including the stiffness from the PM in the dynamic analysis, better determinations of the lowest modes were found.

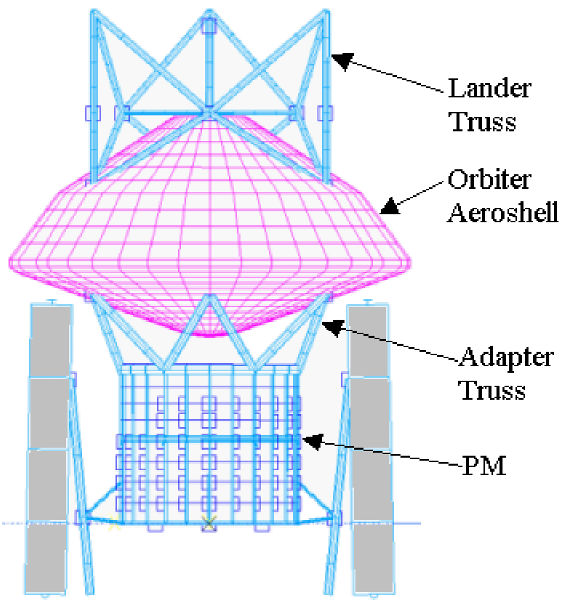

Figure 7. Orbiter FEM Components

Results

A dynamic analysis was performed on the Orbiter and launch stack to check for low natural frequencies. The suggested launch frequency minimums from the Boeing design guide (ref. 3) of 10 Hz lateral and 27 Hz axial were used. The launch stack minimum modes, shown in figures 8 and 9, were 10.5 Hz lateral and 27.8 Hz axial respectively. Figure 10 is the first mode shape of the Orbiter aeroshell at 54.6 Hz. No dynamic magnification factors were considered during launch.

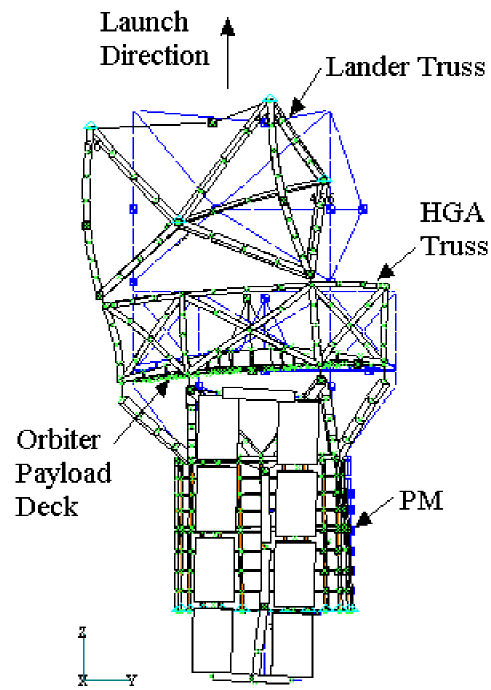

Figure 8. Lateral 10.5 Hz Launch Stack Mode

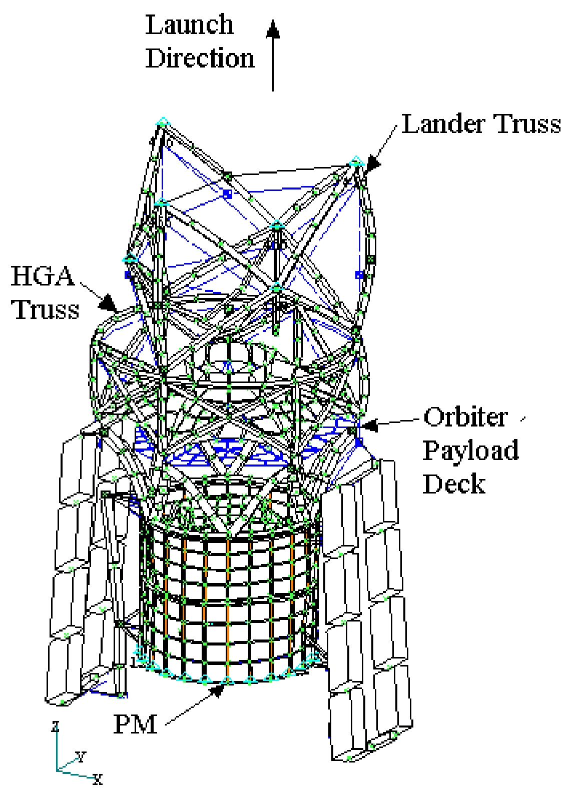

Figure 9. Axial 27.8 Hz Launch Stack Mode

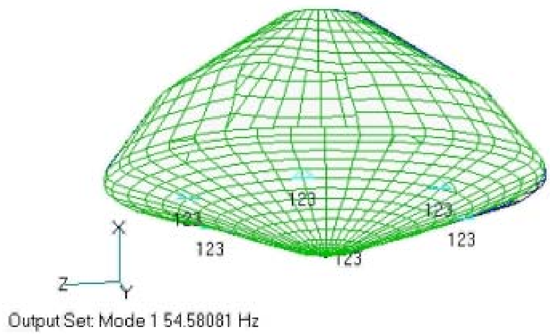

Figure 10. Orbiter 1st mode

The overall maximum deflections on the heatsheild were checked at launch and during entry. The largest deflections occurred on the heatshield during launch and were less than 1 mm. The exaggerated deflected shape of the heatshield is shown in figure 11. Double curvature exists where the six adapter truss points attach through the heatshield. The deflections were small and not considered a concern for TPS bonding.

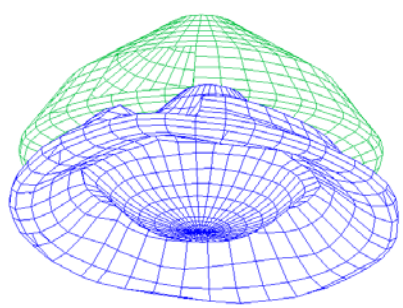

Figure 11. Orbiter Deflections at Launch

The Orbiter's aeroshell was sized after several iterations between nastran and Hypersizer™. The dominant load case was found to be during launch. The process optimized the aeroshell structure and indicated which materials and structural concept would produce the lightest aeroshell. Honeycomb core with face sheets were shown to be the best structural concept. The final core material for the heatshield was a 25.4mm thick Hexcell 5052 alloy hexagonal aluminum honeycomb with 1.7mm graphite polyimide face sheets. This design was similar to the Mars Exploration Rover (MER) heatshield except for the six hard points used in attaching the Orbiter heatshield to the PM. The backshell sizing done within HyperSizer™ showed a honeycomb core face sheet concept produced the minimum structural mass. The core was a 12.7mm thick Hexcell 5052 alloy hexagonal aluminum honeycomb and graphite polyimide face sheets of varying thickness. The cap plate design was similar to the backshell.

Results from analyzing the HGA and Lander trusses showed the optimal material was 2" OD, 0.12" wall M55J/954 tubes. The Orbiter/PM adapter truss was similarly made with 3.2" OD, 0.2" wall M55J/954 tubes. The sizes were driven by finding sections large enough to prevent buckling.

A summary of the final Orbiter aeroshell mass is given in table 1 and the total launch stack mass summary is shown in table 2. The total spacecraft mass for the launch configuration was 3173.2 kg and included the Lander, Orbiter and PM. TPS and non-structural masses were included plus allowances for miscellaneous items such as heatshield to backshell separation components. The design relies on a system of composite M55J trusses that form a load path into the PM. This system produced minimal displacements during launch and held stresses within safety limits that were: 1.4 on ultimate, 1.25 on yield limit and 1.5 for buckling.

Part	Area	Structure Mass	TPS Mass	NSM
Heatshield (TufRoc)	12.58 m²	41.58 kg	148.62 kg	0
Backshell (SLA-561V)	15.01m²	43.27 kg	28.69 kg	2.38 kg
Pallet Ring	1.20m²	42.47 kg	0	1.20 kg
Separation Ring	1.79m²	11.35 kg	0	.89 kg
Separation Ring Attachments	.45m²	2.85 kg	0	4.50 kg
Totals		**141.52 kg**	**177.31kg**	**8.97 kg**

Total Aeroshell (structure + TPS + NSM) = **327.80 kg**

Table 1. Orbiter Aeroshell Mass

Lander	400 kg
Lander Truss	61.8 kg
Orbiter Aeroshell + Payload	1200 kg
Orbiter/PM Truss	61.4 kg
PM	1450 kg
Total spacecraft	**3173.2 kg**

Table 2. Launch Stack Mass

Conclusion

The success of an aerocapture mission at Titan greatly depends on the mass reduction of the structure and the configuration of the launch stack. The design efforts encountered during this conceptual study showed the importance of defining the configuration in reducing spacecraft mass. The final launch configuration used an unconventional method of attaching the Orbiter heatshield to the PM. This method allowed a continuous load path from a 400kg Lander, into the Orbiter, through the Orbiter heatshield and into the PM. Maintaining a load path through the trusses that avoided the Orbiter aeroshell from supporting the Lander minimized the Orbiter aeroshell. The stacking arrangement also minimized the buckling lengths of the truss members as well as the number of required members. HyperSizer™ was used to perform optimization sizing of the Orbiter aeroshell without a detailed mesh and extensive remodeling effort. The results indicated a honeycomb face sheet composite could produce a light structure while providing the necessary stiffness to meet minimum dynamic frequency requirements at launch. The results from this study have established a starting point for a detailed fea of the Orbiter aeroshell. Such an analysis could include varying core and lay-up thickness and detailed analysis of attachment connections and separation mechanisms. The structural mass for this design was within the mass margin estimated for a successful Titan mission.

Acknowledgements

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