References

large structures and liquid fuel storage tanks, the $K_{II}$ is larger than $K_{III}$. The large value of $K_{II}$ for thin panels can be attributed to a dominant free boundary layer effect in thin structures [20,21]. For very thick specimens, $K_{II}$ is almost zero and only $K_{III}$ exists. This value is same as the antiplane solution reported in the stress-intensity-factor handbook [22]. Based on these results, a mixed mode test specimen is being developed at NASA Langley Research Center to characterize the residual strength and fatigue crack growth rate under a mixed mode stress field.

**Advanced Analysis Methods**

At M^2RC a number of advanced structural analysis methods are being developed by utilizing modern computers to address current and future needs for aerospace structures. These include the development of shell and tube theories [23,24] and new algorithms for vector and massively parallel computers [25-27]. Details of these research activities are given in references 23-27. Only the anisotropic shell theory [23,24] for analysis of composite tubes is described here.

As mentioned previously, composite tubes are being used in many space structures, including space station and aerobrakes. A fundamental understanding of tubular structures under thermo-mechanical loads is indispensable in designing light weight structures and in predicting failure. Therefore a closed form stress field solution was developed for cylindrical anisotropic tubes using the Lekhnitskii's stress function approach. Figure 12 shows the configuration of a tube, loading, and the six stress components. The stress field for various types of mechanical loading was calculated and reported in reference 23. Figure 13 shows the axial $(A_3)$ expansion and torsional distortion $(A_4)$ displacements of a symmetric angle ply $(\alpha/\alpha)_{S}$ laminated tube subjected to an incremental thermal load $T$. The angle between the fiber and axis of the tube is represented by $\alpha$. The tube axially contracts for $\alpha$ ranging between 0 and 42.5° and expands for $\alpha$ ranging between 42.5° and 90°. For $\alpha = 42.5°$, the tube's expansion is zero and the torsional distortion is small. Zero expansion tubes are very useful in structures like antennas to reduce thermal stresses and maintain configuration stability.

**Controls Research**

Research in the controls area was focused on the development of algorithms and in their implementation on experimental test articles representative of a variety of spacecraft and robots (for example, lightly damped platforms, satellites, antennas, aerobrakes, and robots). The algorithms being developed fall into two distinct areas: fueltime optimal control and adaptive compensation. In the first area, the spacecraft can perform a reorientation maneuver using minimum fuel or in minimum time. The algorithms govern the switching of valves in the propulsive control system. In the area of adaptive compensation, the dynamics of the spacecraft or the space robotics system is partially unknown (model uncertainty) and environmental disturbances can also be unknown. Besides developing control algorithms, a new concept called "Electrodynamic Structures" is also being researched at M^2RC.

**Fuel/Time Optimal Control of Structures**

Spacecraft reorientation maneuvers are generally performed by propulsive control systems. The "Impulse Damping Control system" is one method that governs bang-off-bang switching of jet valves. This method appears to be well suited to control multi degree-of-freedom systems like spacecraft and other continuous structures. Impulse damping control uses properties associated with the fuel optimal control of a harmonic oscillator to formulate a near fuel optimal closed loop control strategy for spacecraft damping. In this scheme, recursive calculations of standard deviations of spacecraft displacements and velocities regulate on-off actuators [28].

Figure 14 shows a 16 ft hinged-free beam fitted with on-off air fed impulsive actuators. Strain gages and angular rate transducers are mounted at the hinged end to measure displacements and rotations. A two-mode fuel optimal reorientation maneuver of the cantilever beam was successfully conducted [29]. The fuel optimal control technique was also applied to demonstrate the maneuverability of a Mars aerocapture test model. Figure 15 shows a one-tenth scale lunar aerobrake test model having three rotational degrees-of-freedom is suspended at its center. The base diameter of the model is five feet. High pressure jets are mounted at the four quadrants of the aerobrake to orient the aerocapture to a defined direction.

A variety of test studies have shown that the total fuel consumption for the Impulsive Control system is about one-third less than that for a continuous control system [30].

**Adaptive Compensation Control Algorithms and Applications**

Classical compensation techniques used in robot control are based on a complete knowledge of all pertinent system and environmental characteristics. It requires only re-tuning controller gains when parameters change. Thus unknown disturbances or changing environmental conditions may result in performance degradation. To address the issue of uncertainty or time-varying conditions we are developing real-time control and signal processing algorithms. The three methodologies that are being investigated are: self-tuning adaptive controllers, virtual passive compensators, and fuzzy logic controllers.

**Self-tuning controller** - This approach consists of system identification and control strategy. In system identification, several algorithms have been devel-
aerobrake surface and \( p_0 \) is the stagnation pressure. The angle \( \phi \) is measured as shown in figure 2. Aerodynamic heating for a typical atmospheric excursion would raise the surface temperature of the aerobrake at stagnation point to about 2500° to 3000° F. Computed stagnation pressure is about 0.74 psi for 4.18 g deceleration of lunar transfer returning vehicle entering the Earth's atmosphere and is about 2.0 psi for 6 g deceleration of a Mars transfer vehicle entering the Martian atmosphere. The design, analysis, optimization, and mass estimation of a number of aerobrake structural concepts proposed for Mars and lunar reentry vehicles were conducted. In all these studies, a hot-cold structural design concept was used. That is, the aerobrake was assumed to be made up of two layers: an outer hot layer and inner cold layer. The hot layer is the thermal protection system to provide thermal insulation to the inner cold load-bearing structure. The TPS was designed such that the backface temperature is low enough that general aerospace materials could be used to build the cold structure. Design and analysis of the inner cold structure were conducted at M²RC and the details of the thermal protection system was taken from the reference 2.

Figure 3 shows the spheroid [3-6] and sphere-cone [7] aerobrake configurations analyzed for Mars and lunar missions. Aerobrake diameters for two Mars missions were 120 ft and 135 ft and for the lunar mission was 50 ft. Composite tube trusses [3,4] and frames [7], stringer-panels (semi-monocoque) [5], and sandwich honeycomb [6] constructions were chosen in various design studies. Aluminum alloy, titanium alloy, graphite/epoxy, and graphite/polyimide materials were used by various investigators. In references 3-5, and 7 an equivalent uniform load was used, whereas in reference 6 both uniform and aerodynamic loads were considered.

Cross section of a typical sandwich honeycomb aerobrake wall is shown in figure 4. The outer layer is the TPS and the inner layer is the stiff honeycomb structure. In the sandwich shell study, the effect of TPS deformation was included in the analysis. The mass of the TPS was about 0.956 lb/sq. ft [2] and it was assumed to be same for all material cases considered. The design criteria used for aerobrake structure were:

1. factor of safety is 1.4 for stress limits in tension and compression
2. the global buckling load factor must exceed 1.4
3. the maximum compressive load in the struts must not exceed 71% of the Euler buckling load
4. the maximum local deflection of the shell must be less than the TPS tip-off deflection.

In references 4, 5, and 7 the first three design criteria were used where as in reference 6 all four criteria were used. A summary of aerobrake configurations, loading, assumptions, and the design criteria used, and the predicted aerobrake masses are given in table 1. The aerobrake mass is normalized by its spacecraft mass (see table 1). Although the truss aerobrake appears to be the lightest (about eight percent of the spacecraft mass), there are number of uncertainties in those studies that would impact the final results. Some of the uncertainties are the joints' masses are approximate, analyses do not include the TPS tip-off deformation criterion (which was found to be most critical for rigid TPS, refer to [8]), and the depth of the structure may interfere with pay load support. The honeycomb sandwich shell analysis considered all four design criteria and the calculated mass of the aerobrake is about 13.8% of the spacecraft. Therefore, it is possible to design and construct an aerobrake with a mass less than fifteen percent of the spacecraft to provide aerobraking as a viable alternate to retrorockets.

**Design Guidelines for Aerospace Structures**

At M²RC a number of analytical studies are being conducted to develop design guidelines for many aerospace structures. The two such areas being pursued are: buckling of partially restrained composite panels with circular cutouts and global-local buckling of space truss structures.

**Buckling of partially restrained panels with cutouts**

A wealth of literature is available on buckling of panels with and without cutouts subjected to either fixed, hinged, or free supports. Real structures are built-up of structural sections, the edge restraint in these cases is neither hinged nor fixed, but lies in between the two. Currently there are no buckling solutions for partially restrained composite panels. Klang and his students [8,9] developed buckling solutions for partially restrained composite rectangular panels with circular cutouts subjected to normal and shear loading.

Figure 5 shows the configuration and loading of a rectangular panel with a circular cutout. The analysis was based on Ritz and collocation methods using global trigonometric functions developed for partially restrained beams by Razzaq et al. [10]. Linear buckling solutions were generated by considering forty-nine terms in the resulting series equations. Figure 6 shows the variation of normalized buckling load with the hole size (d/W) for various edge restraint conditions (e). The edge restraint factor, \( e \), is zero for simply supported edges and infinity for fixed edges. As expected the buckling load increased with the degree of edge restraint. The solutions for simply supported and fixed edge cases agreed within five percent of the theoretical solutions reported in the literature. The panel used in reference 9 was a (45/-45/0/2)\( _{s} \) graphite/epoxy composite laminate; the subscript \( s \) represents the lay-up symmetry. Buckling loads for the shear loading are shown in figure 7.