Bonded Aluminum Honeycomb—Aircraft Flight Surface Primary Structure Application

T. R. Logan*
Boeing Commercial Airplane Company, Seattle, Washington
and
U. Soudak†
Israel Aircraft Industries, Ltd., Ben Gurion International Airport, Israel

Adhesive bonded aluminum construction has a demonstrated capability of providing the high-level, long-term durability required for commercial transport primary structure. A bonded aluminum honeycomb empennage structural box was designed that shows a 9.9-14.7% weight reduction and a potential 40% fabrication cost reduction compared to conventional design. Anticipating new lightweight aluminums such as lithium/aluminum, this design is proposed for near-term fuel efficiency and cost saving and as an updated metal baseline for realistic evaluation of advanced designs.

Nomenclature

APU = auxiliary power unit
BAC = Boeing Aerospace Company
FPL = Forest Products Laboratory
PABST = Primary Adhesively Bonded Structure Technology Program

Introduction

DEVELOPMENT of advanced, updated metal structural design is a continuing requirement for direct application to production aircraft and as a basis for evaluation of advanced composite designs. This study encompasses design and preliminary development of bonded aluminum honeycomb horizontal and vertical tail structural boxes for a large commercial transport airplane. Primary motivation for development of honeycomb structure is the significant decrease in the number of parts, with the associated decreased fabrication cost, over that of conventional mechanically fastened construction, as illustrated in Fig. 1. This cost advantage, when coupled with significant weight savings, provides an important concept for primary aircraft structure application.

Viability of bonded aluminum honeycomb as a primary aircraft structural concept is underscored by the extensive, successful flight service of secondary structural applications and designs such as the Boeing YC-14 horizontal and vertical tails and the Fokker F-28 vertical tail. Successful application of bonded primary structure on commercial transports has been demonstrated, for example, by the long-term, highly favorable experience of Fokker.[1] In addition, the PABST program[2] has provided a broad understanding and the technology base for metal-to-metal bonding of large-scale primary structure using the latest generation of structural adhesive systems.

This paper addresses the first phase of a detail design and testing program undertaken to establish cost and weight data for a point-design application and to provide a basis for full-scale development and certification of a bonded aluminum honeycomb empennage structural box.

Service Experience

Application of adhesive bonding to metal airplane structures increased steadily through the 1950-60 time period, reaching a peak (for Boeing airplanes) of about one-third the wetted area on the 747 airplane. In recent years, advanced composites have replaced many of the traditional flight control surface bonding applications, so that the amount is now reduced to about 15%, dominated by more highly loaded secondary structure (such as empennage leading edges and flap structure).

Early experience with bonding earned, with some, a poor reputation for service reliability, primarily because of corrosion and related disbonding. However, we now have an increased knowledge of bonding mechanisms, and the critical role of surface preparation is clearly understood.[3,4] Also, sufficient improvements in bonding techniques, processes, and laboratory test procedures correlated to service experience[5,6] have been made, so that these past problems are no longer a significant hinderance to expanded applications of adhesive bonding.

To verify these improvements, a recent in-depth survey was made of 63 different operators of Boeing airplanes. The survey was limited to aircraft with delivery dates starting in 1974, to isolate those airplanes for which either the optimized FPL etch or BAC 5555 phosphoric acid anodize process was used. Fifty-four operators reported no bond-line delamination or corrosion; delaminations reported by the remaining operators were either random single occurrences or specifically involved a wing trailing-edge panel exposed to APU exhaust. This component has subsequently been reconfigured and is currently trouble free. An extensive parallel survey was also made of spare parts usage and significant service item reports of Boeing components using the BAC 5555 process. This survey confirmed conclusions from the in-depth survey. Figure 2 summarizes the service experience with bonded aluminum structure using the improved processes.

A similar trend of successful experience with improved bonding systems has been reported for USAF aircraft,[7] which includes bonded aluminum honeycomb underfloor bulkheads in the C-5. This improved service experience, together with the extensive positive service experience reported in Ref. 8,

---

Presented as Paper 82-0676 at the AIAA/ASME/ASCE/AHS 23rd Structures, Structural Dynamics and Materials Conference, New Orleans, La., May 10-12, 1982; submitted May 28, 1982; revision received Feb. 1, 1983.

*Manager, Structures Design Development. Member AIAA.

†Staff Member, Boeing. Currently, Manager, Airframe Design, Israel Aircraft Industries, Ltd.

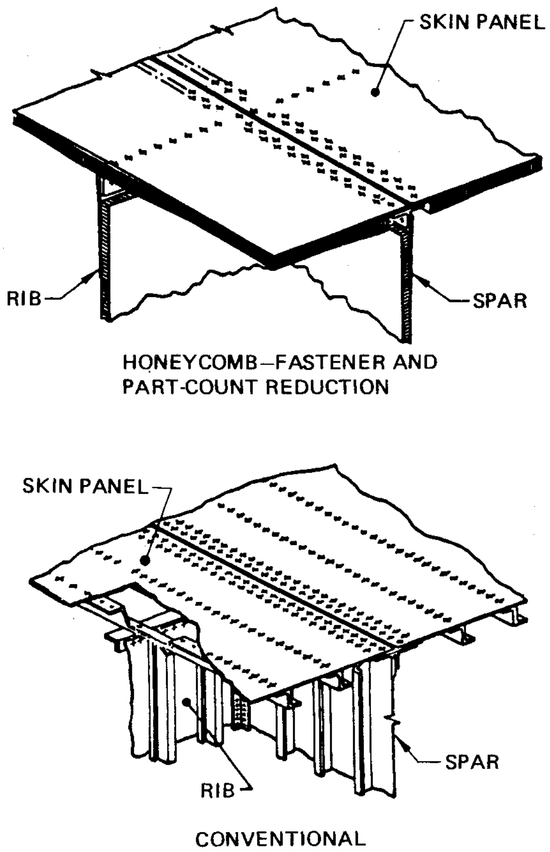


Fig. 1 Bonded aluminum honeycomb vs conventional construction.

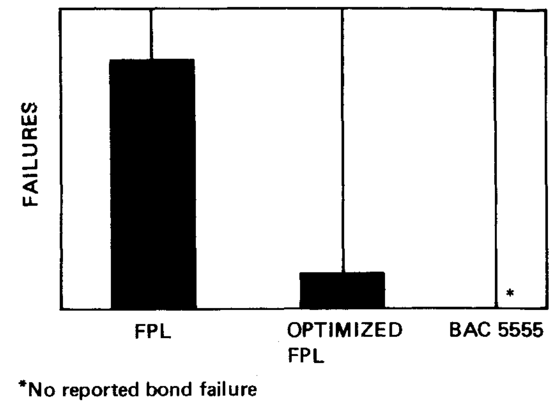


Fig. 2 Metal-metal and/or honeycomb-metal bond-line delamination.

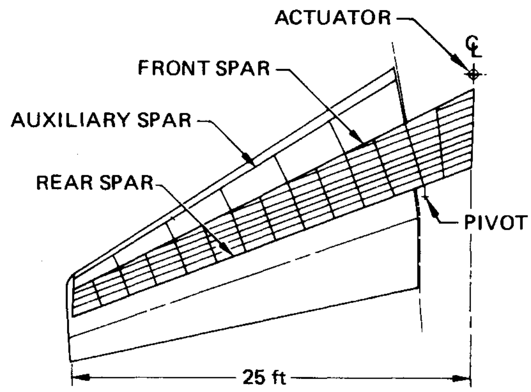


Fig. 3 Horizontal stabilizer box.

demonstrates that the current avoidance of adhesive bonded primary structure on transport aircraft is unwarranted.

Structural Design

Detail design studies centered around the horizontal tail structural box are shown in Fig. 3. The baseline design is a built-up, mechanically fastened skin-stringer configuration. Upper skin panels are basically composed of 2024 aluminum, with the remainder of the box being 7075 aluminum. It is a three-spar conventional design, where only the box between the front and the rear spar is considered to be carrying bending loads. The front area, between the auxiliary spar and the front spar, is made of bonded aluminum honeycomb panels for rigidity and for local loads. This optimized baseline design is improved over previous Boeing designs, offering a 10% cost reduction and a 3% weight reduction compared to existing concepts in which the carry-through structure is a separate subassembly.

Design Goals and Constraints

The overall goals of the proposed bonded design are to 1) decrease weight by 10%; 2) significantly decrease total cost by decreasing part count; and 3) design for inspection intervals equal to the baseline.

Inherent structural advantages of the bonded design are 1) improved fatigue strength (less attachments with related stress concentration, and 2) increased torsional stiffness efficiency of panel material.

The following principal constraints were imposed on the advanced bonded design:

1) Match baseline bending and torsional stiffness (aeroelastic redesign not considered).

2) Avoid use of bonding for major assembly to avoid related development costs and increased inspection requirements.

3) Match geometry and attachment points of baseline stabilizer to provide interchangeability (the new bonded concept could be improved by repositioning the jack attachment).

4) Avoid attachment through the core (where unavoidable, use potting compound).

5) Fail-safe design required limit-load capability with failure (streamwise) of a complete panel or a spar chord.

6) Design for ease of repair (on the airplane wherever possible).

Design Allowables

Buckling allowables for honeycomb panels were taken from YC-14 empennage tests and were 55 ksi for 7075 aluminum and 40 ksi for 2024 aluminum. These same buckling cutoff allowables were used for this study, for compression and for combined compression and shear loads. Preliminary results from small specimen tests clearly indicate that these are conservative allowables because the detail design for buckling strength is now improved. Reduction of the difference between the inner and outer skin gages (to a ratio of 2:1 or less) show significant increases in buckling strength; equal thickness sheets provide the highest buckling strength. All of the small coupon specimens and the large panels tested to date buckled the inner smaller gage skin and support this theoretical result.

Because of the lack of test data regarding damage tolerance in large bonded honeycomb panels, maximum tension stress in 2024 aluminum tension skins was limited to 45 ksi, although 50 ksi is used for conventional skin-stringer designs and therefore represents a conservative approach.

Optimum Design

Minimum weight can be obtained by a linear combination of skin gages and chord areas for maximum allowable stress. This combination is limited by the fail-safe requirement, which is a function of panel width. As panel width increases, skin area can be added (for a given number of spars), thereby improving load distribution. On the other hand, more spar chord area would be required to meet fail-safe requirements, and a failed spar chord would have a larger effect. Optimum design points are obtained by developing iterations on these variables.

For each skin panel, an optimum must be sought between two conflicting requirements: similar skin gages necessary for maximum buckling strength vs a required 2:1 to 3:1 ratio of outer-to-inner skin to meet exterior minimum gage and producibility requirements. Optimum design was done analytically, and large panel tests, now underway, should support the use of increased buckling allowables and result in additional weight saving. Instability modes of shear crimping and face wrinkling were completely avoided, based on conventional analysis and specimen tests.

A flowchart for the optimization process used for the skin panels is shown in Fig. 4. The analysis used three computer programs: The first provided overall preliminary optimization of the box using a finite element analysis; the second provided detail buckling optimization for the compression structure based on linear behavior; and the third provided fail-safe capability according to the following criteria:

1) $F_{F.S.} = 2/3\ F_{ult}$, where $F_{F.S.}$ is the allowable fail-safe stress and F_{ult} the ultimate tension stress.

2) Any crack in the honeycomb panel was assumed to cause the entire panel to fail; spar chords are fully effective in carrying all cracked panel loads, distributed proportionately to their areas.

3) Allowable chord stresses for 7075 aluminum, considering stress concentration effect, was 66 ksi.

These criteria, along with a safety factor, led to the following formula:

$$\frac{66}{[2/3 + .697(wt/A)]S} > 1.02$$

where w is the panel width, t the combined panel skin thickness, A the spar chord area, and S the stress in panel.

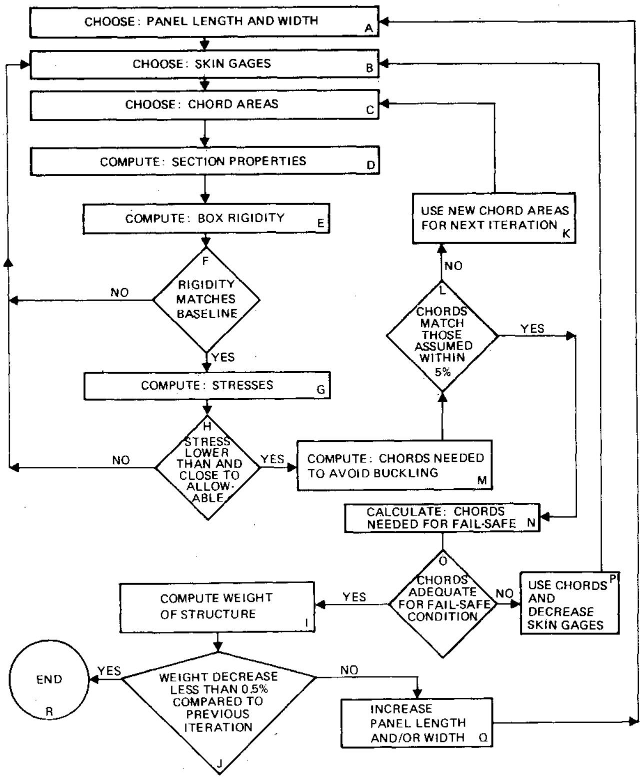


Fig. 4 Flowchart for optimization process.

Critical design conditions involving fail-safe and stability factors interact as shown in Fig. 5. The iterative process outlined in Fig. 4 was used to find the optimum point for each bay.

"Core effect," a phenomenon of face skin relief,[9] was not taken into account, as it is a recent finding and proven only for small specimens. This effect accounts for the load actually carried by the core and can have a significant practical value, depending on the detail configuration. Studies indicate that future large panel tests and more experience with this concept will provide additional weight savings.

Substructure Arrangement

A variety of structural arrangements was evaluated to ascertain the optimum layout; principal configurations are summarized in Fig. 6. It should be noted that some of these arrangements are made practical only by the inherent simplicity of honeycomb design. For example, the fanned spar arrangements, such as 6c and 6d, are economically attractive with a honeycomb design, but are usually not economical when using conventionally stiffened skin construction in the gages being considered.

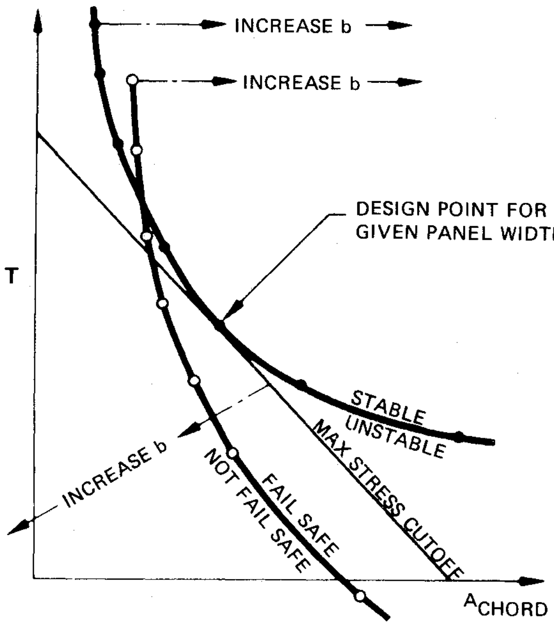


Fig. 5 Design condition interaction.

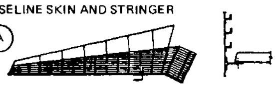
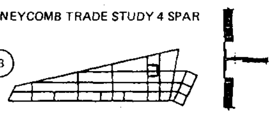
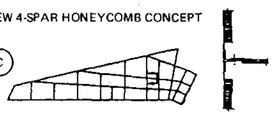
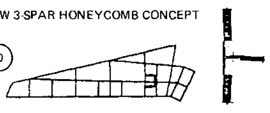
Concept	Weight (lb)	Part count
BASELINE SKIN AND STRINGER (A) 	2,490	Parts: 1,135
HONEYCOMB TRADE STUDY 4 SPAR (B) 	-150 (-5.9%)	Parts: -200 (-18%)
NEW 4-SPAR HONEYCOMB CONCEPT (C) 	-300 (-12%)	Parts: -230 (-20%)
NEW 3-SPAR HONEYCOMB CONCEPT (D) 	-170 (-6.8%)	Parts: -320 (-28%)

Fig. 6 Structural arrangement trade study.

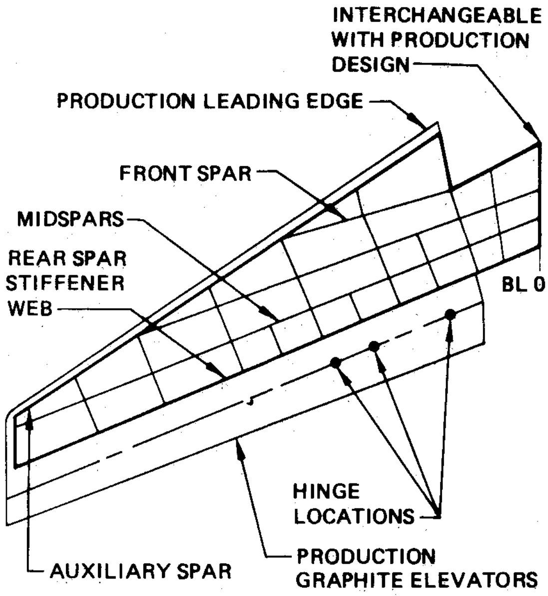


Fig. 7 Selected structural arrangement.

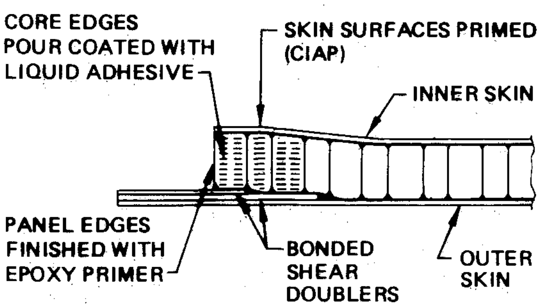


Fig. 8 Square-edge joint concept.

Table 1 Weight summary

	Baseline weight, lb	Low development weight, lb	Potential weight, lb
Horizontal stabilizer box	1790	1607 (−10.2%)	1520 (−15.1%)
Vertical stabilizer box	1216	1100 (−9.5%)	1044 (−14.1%)
Total	3006	2707 (−9.9%)	2564 (−14.7%)

The ribs were positioned in accordance with local elevator hinge loads and optimum panel sizes. Most of the hinge-load ribs are only partial, extending to the midspar. Using a conventional structural arrangement, as shown in Fig. 3, two additional intermediate spars were required for compliance with fail-safe requirements. However, these additional spars can add a significant weight and cost penalty. A completely new concept is suggested, as shown in Fig. 7, which provides a solution to these concerns. In this arrangement, the front and intermediate spars are swept forward to minimize their length. Thus a more efficient design is achieved for less weight, as the primary torque box is now increased in area and its loads are carried through directly to the centerline, with no dead ends, except for a small triangle at the front where loads are very low. Note that the angle in the forward spar must use the same body cutout as the baseline design, but since the loads are relatively low in this area, the weight penalty is small. This design also features access panels in the skin for assembly and inspection, increased fatigue strength at the centerline at-

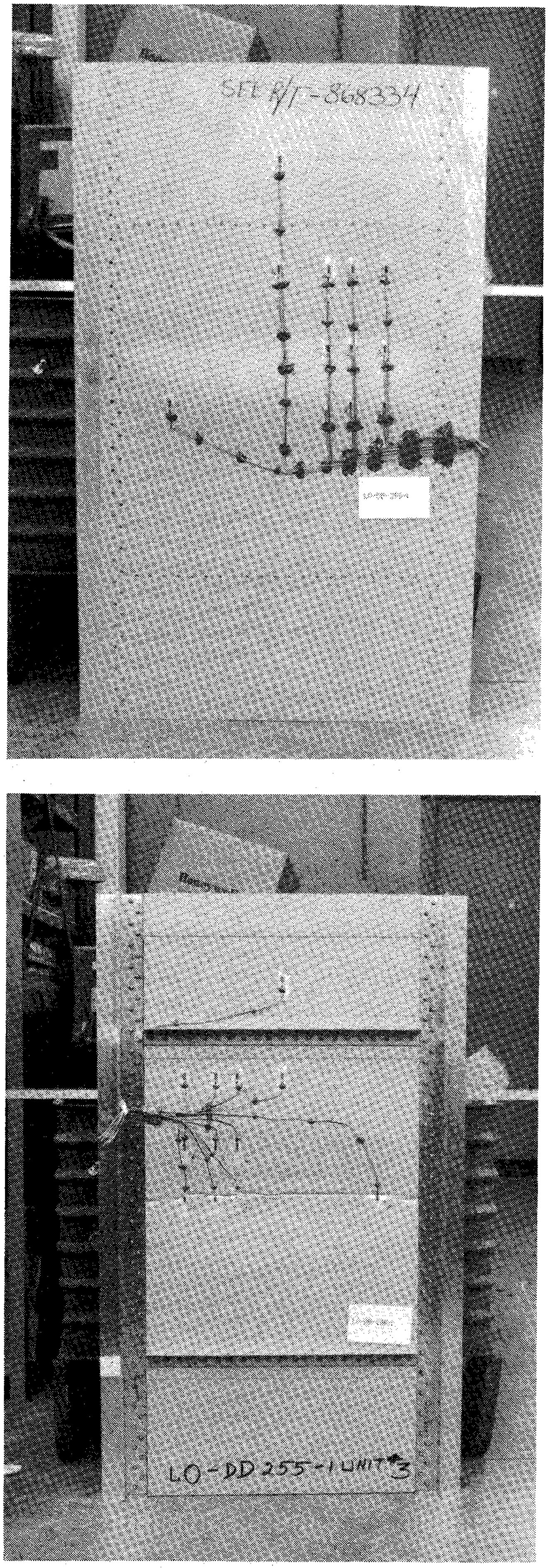


Fig. 9 Honeycomb test panel.

tachments because of the decreased attachment angle, and a decrease in the carry-through area and weight. To summarize, this arrangement will produce a minimum weight design with low fabrication costs.

The YC-14 empennage was estimated to have a 40% cost reduction over conventional skin and stringer design. The design in this study is expected to achieve the same or greater cost reduction using the square-edge and fanned spar concepts.

For improved internal access and further decrease in fabrication and inspection costs, a refined configuration is being developed that deletes the spar web from the aft midspar. The spar chord is then sized to provide a spanwise buckling node along the panel edge. This study is being conducted with Stanford University and will be reported separately when complete. The advantages include greatly improved internal access, which is a benefit during assembly and for ease of service inspection. This concept also allows elimination of a large number of holes required for access, providing an incremental cost and weight reduction.

Edge closeouts in sandwich panels add significant production costs and can be a problem in service. The structural concept chosen for this design uses a "square-edge joint," shown in Fig. 8. The concept has seen extensive service in Boeing airplanes and is discussed in Ref. 9. The main advantage of this concept is that it significantly reduces fabrication cost, as compared to more traditional closeout techniques. Square-edge joints can be analyzed and designed to carry compression and tension loads with the same effectivity as a continuous core joint, except for instability modes. Thus all spar joints are designed square edged, as are rib joints toward the tip, where the compression load is sufficiently low (less than 2000 lb/in.).

Weight Summary

Based on this study summary, weight savings were developed as shown in Table 1. Vertical stabilizer weights were developed by comparison of areas and other factors with the detailed analysis for the horizontal stabilizer. The difference between these weights and those in Fig. 6 is due to the inclusion of some installation items and a more refined baseline design. For the configuration studied, the weight saving was estimated at 9.9%. Additional development, including more extensive bonding for assembly of major box subcomponents and the use of dense core (55 lb/ft^3) for fastener installation, rather than potting compound, will increase this saving as shown. An additional equivalent weight saving of 50 lb is gained by the honeycomb design because of the decreased drag of smoother skins compared to the mechanically fastened design.

Test Program

A test program is currently underway to validate the allowables used and to verify the design configuration. The present phase of the program includes testing of a variety of joint concepts and a number of compression panels (shown in Fig. 9). This particular panel includes an access hole that uses a square-edge concept and a local reinforcing ring. The next test phase will include a series of stub boxes to verify panel design and substructure details.

Conclusions

A preliminary study of a bonded aluminum honeycomb horizontal stabilizer box demonstrates a significant 9.9-14.7% weight saving. In addition, a 20% part-count reduction and simpler design can potentially reduce overall manufacturing costs by at least 40%. Service experience with improved bonding systems has shown that structural adhesive bonding, as a primary fastening technique, is as reliable as conventional mechanical attachment. Anticipating new lightweight metals (such as lithium/aluminum), updated metal designs have a definite place in airplane primary structural design studies, providing an improved basis for comparison and improving near-term fuel efficiency while awaiting the greater potential of advanced composites.

Acknowledgment

The authors gratefully acknowledge the work performed by the BCAC Structures Design Development group.

References

[1] Schliekelmann, R.J., "Adhesive Bonding in the Fokker-VFW F-28 Fellowship," Royal Netherlands Aircraft Factories, Fokker, Feb. 1973.

[2] Thrall, E.W., "PABST Program Test Results," *Adhesives Age*, Vol. 22, Oct. 1979, pp. 22-33.

[3] McMillan, J.C., "Surface Preparation—The Key to Bondment Durability," *Bonded Joints and Preparation for Bonding*, AGARD Lecture Series No. 102, April 1979.

[4] Minford, J.D., "Effect of Surface Preparation on Stressed Aluminum Joints in Corrosive Saltwater Exposure," *Adhesives Age*, Vol. 23, Oct. 1980, pp. 36-41.

[5] Marceau, J.A. and McMillan, J.C., "Exploratory Development on Durability of Adhesive Bonded Joints," Final Report AFML-TR-76-173; (Contract AF 33615-74-C-5065, Oct. 1976.

[6] Benoliel, R.W., "Durability of Adhesive Bonded FPL Etch and Phosphoric Acid Anodized Test Specimens," Society for the Advancement of Material and Process Engineering, Azusa, Calif., V10, 1978.

[7] Reinhart, T.J., "Use of Structural Adhesives by the U.S. Air Force," *Adhesives Age*, Vol. 24, Aug. 1981, pp. 20-23.

[8] Schliekelmann, R.J., "Operational Experience With Adhesive Bonded Structures," *Bonded Joints and Preparation for Bonding*, AGARD Lecture Series No. 102, April 1979.

[9] Soudak, U. and Hamm, R.A., "The Design and Analysis of Aluminum Honeycomb Sandwich Panel Joints for Primary Structure," *Journal of Aircraft*, Vol. 18, Sept. 1981, pp. 770-774.