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THE PROMISE OF LAMINATED METALS IN AIRCRAFT DESIGN

D. H. Petersen, L. E. Sloter II, W. A. Poindexter, J. L. Maris, and G. E. Kuhn
Vought Corporation
Dallas, Texas

ABSTRACT

Laminated metallics have been shown to enhance the fracture and fatigue resistance of structural components and offer the added advantages of weight and cost reduction in aircraft structures. Vought Corporation has been a leader in the development of new laminated structural materials concepts in the area of both adhesively bonded aluminum and titanium and metallurgically bonded steel, aluminum and titanium. Fracture toughness improvements of 200 percent have been achieved in the all-metal type of metal-to-metal laminated materials developed under Naval Air Systems Command sponsorship. The enhanced durability that results can be effectively utilized to provide increased life as well as to reduce weight of aircraft structures. Similar results were also obtained in adhesively bonded metal laminate systems.

A significant advancement demonstrating the state-of-the-art of laminated metals was accomplished in the Advanced Technology Wing Program (ATW) sponsored by the AFWAL Flight Dynamics Laboratory. A wing section was designed and built such that the lower skin consisted of aluminum layers adhesively bonded together. Specifically, the entire lower skin of the new 10-feet long test article was constructed from 0.080 inch 2024 aluminum. The sheets were stacked to varying thicknesses, integrated with the "T" spars, and bonded together. No fasteners penetrated the lower wing cover. By eliminating these fasteners in the lower skins, it was possible to rid the structure of corrosion intrusion sites and potential locations for structural cracking . The reduced number of fasteners also helps to decrease manufacturing and assembly costs. The wing suffered no damage during two lifetimes of spectrum fatigue testing. It also survived an additional 1.8 lifetimes of damage tolerance testing. The testing included exposure to sump water and simulated JP-4 fuel.

The relative merits of monolithic metals, adhesively bonded sheet metal and a new family of metallurgically bonded laminar alloys are presented and discussed.

I. INTRODUCTION

Ownership costs of operational aircraft continue to increase at an alarming rate. Although the key cost elements can be categorized simply as acquisition/amortization expense, fuel, and maintenance, their relative importances can vary widely with the nature of the aircraft, e.g., cargo vs. defense, and the service environment. Obviously, the effects of interest rates, fuel price fluctuation, changing service, safety and noise control requirements, and manpower costs all affect the complex equation for computing life cycle costs. New materials systems and design concepts are being continuously sought to reduce or control life cycle costs and improve system efficiency.

Aerospace designers who use classical materials systems and fabrication processes are finding that they are reaching the limits of optimization of their design approaches. The designer who understands the overall impact of materials on performance and works hand-in-hand with the materials specialist to cleverly incorporate new concepts will be the originator of improved aircraft. This paper will focus on two Vought Corporation technologies based on laminated metals which show great promise in current and future aircraft design by virtue of their improved durability and reduced weight at acceptable manufacturing costs.

Even though many materials concepts such as adhesive bonding, superplastic forming, and graphite/epoxy composites, are commonly called "advanced" they are actually state-of-the-art since these processes and materials are being incorporated into aircraft design by industry leaders whenever they are considered cost effective. Adhesive bonding is one of those technologies which is applicable to both defense and commercial aircraft. This will be discussed using as an example the Advanced Technology Wing wing box design developed under contract with the Air Force.(1)

The second subject that will be discussed is truly an emerging technology. It is based on metallurgically bonded laminates which have the potential to improve airframe component durability and life by factors of two or more while at the same time reducing structural weight.(2-8)

II. DURABILITY OF AIRCRAFT STRUCTURES

The relationship shown in equation 1,

$$K = \sigma\sqrt{a}Y \tag{1}$$

where "K" is the fracture toughness, "σ" is the design stress, "a" is the crack length and "Y" is a geometric factor, illustrates the need to incorporate fracture mechanics into aircraft design philosophy. For a thick plate with a thru-crack far from the plate edges (Figure 1) equation 1 becomes

$$K_{Ic} = \sigma\sqrt{\pi a_c} \tag{2}$$

where "K_{Ic}" is the plane strain fracture toughness and "a_c" is the critical crack length at which failure will occur. Table I lists "a_c" for a variety of high performance alloys.

The technical significance of this relationship is that, for a constant flaw size, doubling the fracture toughness provides a potential doubling of allowable design stress. Conversely, a 100 percent improvement in toughness produces a 400 percent improvement in damage tolerance at the same design load by increasing the critical flaw size four fold.

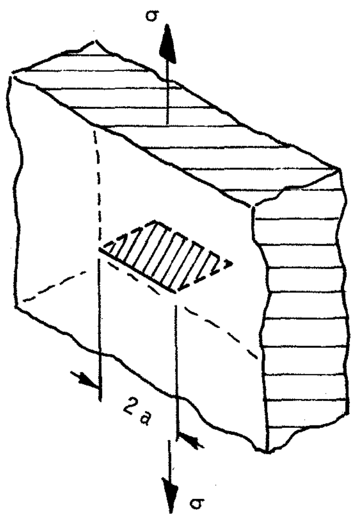

FIGURE 1. CLASSICAL THRU-CRACK IN STRUCTURAL PLATE

Alloy	K_{Ic} (ksi $\sqrt{in.}$)	σ_{ys} (ksi)	a_c (in.)
2024-T851	24	66	0.17
7075-T651	22	72	0.12
Ti-6Al-4V	105	132	0.81
Ti-6Al-4V	50	150	0.14
4340	90	125	0.66
4340	55	220	0.08
52100	13	300	0.002
300M	61	265	0.067
H-11	35	260	0.020
H-11	25	300	0.009
AF1410	135	220	0.382

TABLE 1. ENGINEERING ALLOY PLANE STRAIN TOUGHNESS

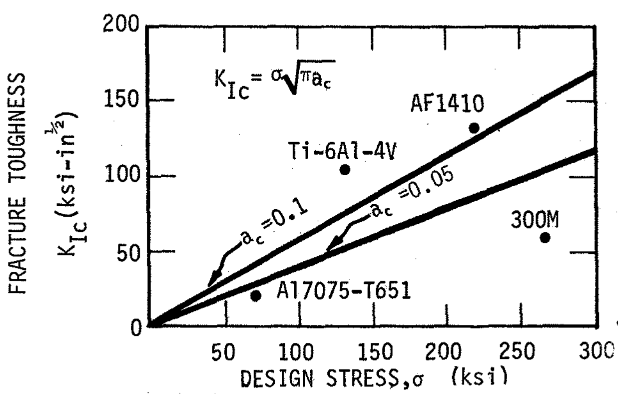

FIGURE 2. DESIGN STRESS VS. TOUGHNESS

Figure 2 graphically shows the allowable design stress vs. K_{Ic} for values of 0.05 and 0.1 inch for "a_c". This figure is valid for any alloy whose behavior can be treated by linear elastic fracture mechanics. Remember that the value of "a_c", Equation 2, is one-half the structural crack length shown in Figure 1.

The selection of a value for "a_c" will depend on one's design philosophy and, more particularly, his ability to detect flaws with confidence. A realistic value of "a_c" probably lies between 0.1 and 0.2 inches when considering the field inspection flaw size detection confidence limits of state-of-the-art techniques. What is really being said is that any flaw which exists will grow and that flaw length must not reach a value of "$2a_c$" between inspection intervals. Consequently, knowledge of the fatigue crack growth rate, the fracture toughness of the material and the stress field within the structure must be used to establish a safe design.

Incorporating actual materials into Figure 2, e.g., 300M steel and titanium-6,4 alloys, it is readily seen that certain materials are inefficiently utilized in many designs. Optimal properties will call for materials to fall on the curve through the origin. Figure 2 shows that the usable strength of 300M steel can not be fully employed if we assume that a crack of 0.2 inches may exist prior to detection at a periodic Nondestructive Inspection (NDI) interval. The titanium alloy has more toughness than needed to fulfill its needs on the basis of usable strength.

The opportunity then lies in making the structure more durable by more efficient utilization of strength for alloys such as aluminum and steel by increasing toughness and by increasing strength without loss of toughness for titanium. Vought Corporation has successfully demonstrated the ability to improve the durability of alloys through use of a "soft-interleaf" concept (Figure 3). If the interleaf is an adhesive, the concept is classical adhesive bonding. When the interleaf is metallic, it is a metallurgically bonded laminate.

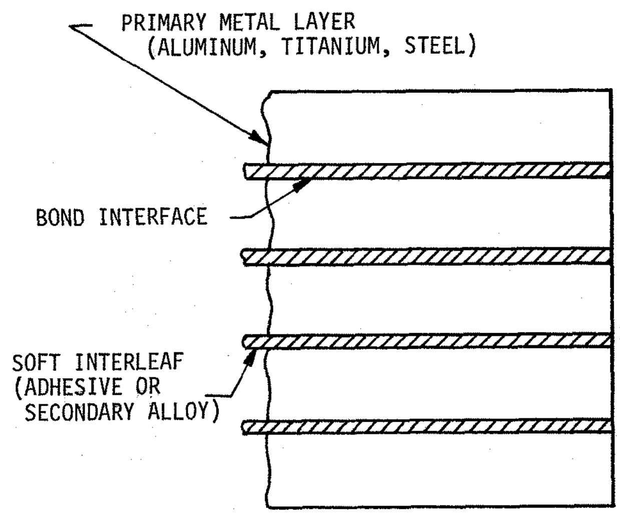

FIGURE 3. SOFT INTERLEAF LAMINATE CONCEPT

The toughness of metals increases as plate thickness decreases (Figure 4). Vought has shown that if one uses the optimum thickness for high toughness as a layer thickness, the laminate will have the same high toughness even in thick section lay-ups. The key to this phenomenon is to keep the individual layers acting independently of one another at a critical stress level. The soft interleaf, e.g., adhesive or secondary alloy, provides that condition and prevents the bonded layers from being just another monolithic plate such as occurs when titanium-6,4 layers are diffusion bonded together. Both these soft interleaf laminate concepts are discussed here.

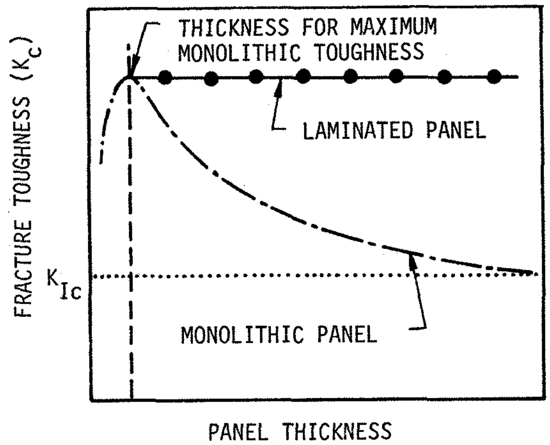

FIGURE 4. EFFECT OF LAMINATION ON CRACK DIVIDER FRACTURE TOUGHNESS

III. ADVANCED TECHNOLOGY WING (ATW)

The Air Force has sponsored a number of programs on adhesively bonded structures of both small and large aircraft including primary structures. Two of these programs performed at Vought,

(1) "Bonded Multilayer F-104 Aft Fuselage Ring Fittings," F33615-72-C-1618, and

(2) "Advanced Technology Wing", F33615-76-C-3138,

as well as cooperative programs with Air Force Flight Dynamics Laboratory and Naval Air Development Center on adhesively bonded and rivet bonded box beams exemplify the types of benefits which can be derived by diverging from the classical bolted and riveted designs to new state-of-the-art fabrication methodologies. The Advanced Technology Wing is especially illustrative of the kinds of benefits that can be derived when materials engineers and designers work together in a creative environment. The demonstration article was a full chord wing box similar to the inboard ten feet of the FB-111 wing main box just outboard of the pivot fitting area (Figures 5 and 6).

The upper skin is a one piece 2024-T851 machine-tapered plate that is mechanically fastened to the substructure by screws and nut plates. The front and rear beam are one-piece, integrally stiffened, machined 2024-T851 aluminum. Fuel sealing is achieved by centerline channel sealing at the 1/4 inch screws and nutplates. The lower cap is integral with the lower skin assembly.

Intermediate spars are formed sine wave sheet metal of 2024-T62 aluminum. The spars are 0.063 inch thick with two 0.063 inch bonded and nested aluminum doublers at the flange that supports the compression skin. The spars are attached to the

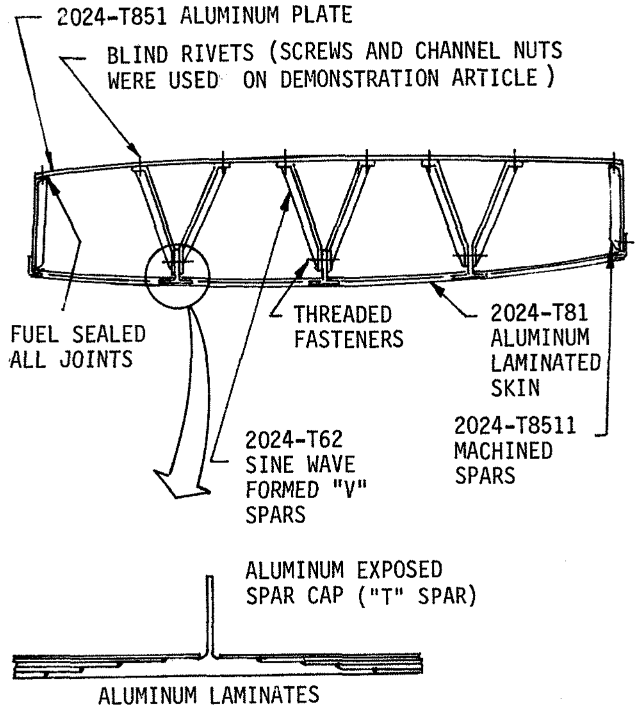

FIGURE 5. ATW LAMINATED DESIGN CONCEPT

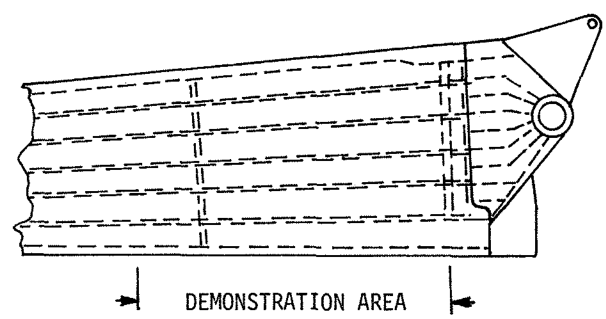

FIGURE 6. F-111 MAIN WING BOX DEMONSTRATION SECTION

upper skin by 1/4 inch screws and to the upstanding leg of the lower tee cap by 3/16 inch Hi Lok fasteners. The intermediate spars are paired to form a Vee at the lower cap, thus eliminating skin/spar joints in the lower skin.

The lower skin assembly is all bonded with 0.08 inch laminated skin planks of 2024-T81 aluminum bonded to integral extruded spar caps of 2024-T851 aluminum. The adhesive system is BR-127 bond primer and FM 300 film adhesive. Bond line uniformity was excellent, ranging from 0.010 to 0.012 inch.

All results indicated that this skin configuration can be fabricated without difficulty. In order to avoid defects due to the air entrapment sometimes experienced with wide area bonds, a thin layer of non-woven Dacron positioning cloth (AF-3306) was placed over the adhesive prior to assembling the next lamina, forming an escape path for any trapped gases. The panels were prepared and autoclave cured conventionally.

The exposed spar cap design (planked laminated aluminum skins) is a multi-load path fail safe structure because the planked configuration restricts any fracture to a local area (one bay). This category allows for smaller initial flaw assumptions (0.10 inch) and a minimum period of unrepaired service usage of one design lifetime. This allowable was verified by Vought crack growth tests and analyses as presented below. The damage tolerance evaluation of the aluminum lower skin joint consisted of an analytical prediction of the crack growth behavior of the design with verification testing using a 48-inch tension panel. The crack-growth data from this test was compared with Vought's analytical program. The standard equation for surface flaws,

$$K = 1.1\,\sigma \sqrt{\frac{\pi a}{Q}}\, M_K \tag{3}$$

where "Q" is the flaw shape parameter found in Figure 7 and M_K is the elastic sheet magnification factor shown in Figure 8.

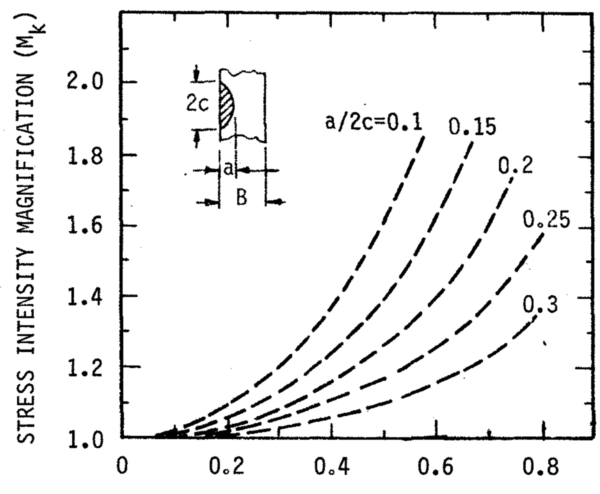

FIGURE 7. ELASTIC STRESS MAGNIFICATION FACTORS FOR DEEP SURFACE DISCOUNTINUITIES - IN TENSION

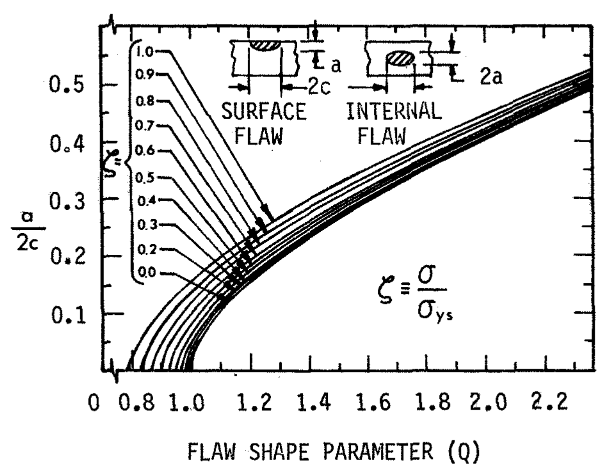

FIGURE 8. SHAPE PARAMETER CURVES FOR SURFACE AND INTERNAL FLAWS

In general, the comparison of test data to predicted data is very good. Some of the difference, however, may be explained by the fact that the analytical prediction did not account for any crack growth retardation which may exist in the actual test data.

IV. ATW TEST RESULTS

The wing box successfully met the design goal of completing two lifetimes of spectrum loading with no apparent damage. Design stress levels were achieved in the test article as verified by the strain gauge readings. Superior fuel containment was demonstrated with only minor leaks identified in the upper cover, which were easily repaired by reinjection into the sealant groove.

Upon successful completion of the fatigue test program damage tolerance testing was initiated. Damage tolerance testing requirements for the ATW based on MIL-A-83444 were:

(1) The structural elements must sustain one lifetime of spectrum loading with an initial flaw of 0.10 inch length.

(2) The wing box must have a residual strength (P_{XX}) at the end of one lifetime of damage tolerance testing of 1.2 times maximum spectrum loading.

(3) After failure of a structural element, the wing box must sustain a single load path failure load (P_{yy}) of 1.15 P_{XX}.

All these requirements were met.

The testing was initiated by installing thumbnail flaws in the lower cover in critical locations of high stress (Figures 9 and 10). After cracks were installed, the full scale test article was cycled at constant amplitude loading until the first crack reached a length of 0.10 inch when spectrum testing was initiated. The flaws were exposed to 100 percent relative humidity using water saturated wicks. Over the 4-month test period surface width was measured every 100 missions at the four flaw locations.

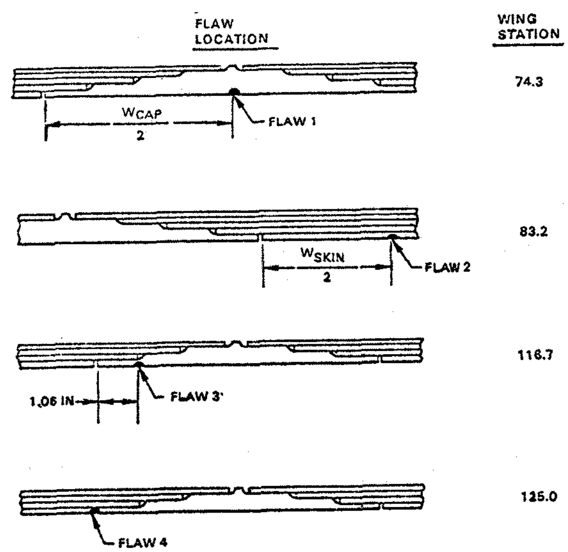

FIGURE 9. DAMAGE TOLERANCE FLAWS

As predicted, crack growth was much greater on flaw number 1 than the other three flaws. The prediction to measured comparison showed excellent correlation in the first lifetime of testing for flaw number 1, but after application of P_{xx} (102 percent LL), the crack growth fell significantly below that predicted. Using the surface crack length (2c) data measured through 1630 missions, an additional crack growth analysis was performed to investigate increasing spectrum loads to insure failure during the second lifetime. That analysis for a 10 percent increase in stress, beginning at the last surface crack reading of 0.307 inch, predicted failure at 1910 missions. To increase the probability of element failure in the second lifetime, the loads were then increased 10 percent after 410 missions of the second lifetime.

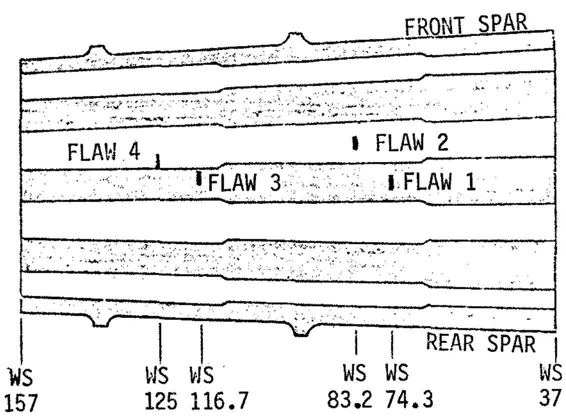

FIGURE 10. LOWER SKIN FLAW LOCATIONS

The ATW damage tolerance testing was successfully completed with the failure of the spar cap element and then the application of P_{yy} (117 percent limit load) without catastropic failure. Failure occurred at 1068 missions of the second lifetime of damage tolerance testing. This concluded eight months of cycling that included two lifetimes of fatigue testing, installing intentional flaws and then the damage tolerance testing. Although the inboard installed flaw, flaw 1, was on the verge of complete element failure, the origination of the failure was 3.5 inches outboard of that point and initiated from inside the box at a fuel passage area. Considering the constant amplitude cycling to grow the crack and the 10 percent increase in loads at 410 missions of the second lifetime of damage tolerance testing, equivalent fatigue life at failure equaled 4.5 lifetimes of cycling. This was more than twice the goal.

V. ATW COST AND WEIGHT PROJECTIONS

To gauge the effectiveness of the ATW wing designs, detailed estimates of projected production and maintenance costs and weights were made. These estimates were then compared to similar projections for the baseline configuration. This task was accomplished at the completion of the demonstration article design effort so that realistic requirements determined during detail design could be reflected in the estimates.

Both cost and weight data were calculated for the full structural box from pivot fitting to wing tip. The comparison showed significant cost and weight savings for the ATW configurations relative to to the conventional baseline article.

Projected production costs for the laminated aluminum ATW box indicated potential savings of up to 17 percent over the baseline wing. These savings resulted from reductions in recurring costs for both materials and labor over a production run of 200 ship sets. Comparisons of maintenance and repair costs for an operational fleet of aircraft resulted in a predicted savings of 45 percent for the ATW designs in maintenance manhours per flight hour.

Recurring production costs were compiled from part by part estimates for materials and labor. Component definition and sizes were taken from full span layouts for both the baseline and ATW configurations, supplemented by details from the demonstration article designs. Actual costs were monitored during fabrication of test article. The estimates are for the full structural wing box including the pivot fitting.

Manufacturing labor estimates were determined for each detail part and assembly sequence using Vought standard hours and estimating factors. The estimates included all essential manufacturing operations for each component and assembly. Definition of these operations was provided by preliminary production planning developed for evaluation of 48 inch span analytical sections during concept selection, and was supplemented by additional information as required to accurately represent the full span structure. Raw material requirements were defined in similar detail.

Appropriate factors were applied to these standard hours to account for realization and learning experiences. Prorations were added for processing, production control, quality assurance, engineering liaison and recurring tooling costs. Labor hours were developed for quantities of 1, 4, 40, 100, and 200 ship sets, produced at a rate of four ship sets per month.

Table 2 compares the total recurring production costs of the baseline and ATW components. The costs presented are cumulative averages for the various quantities. As shown, the projected cost savings achievable with the ATW configuration range from 11.4 percent for the first article to 16.5 percent for a production run of 200.

A detailed comparison of the costs of the various structural components of the wings is presented in Table 3, based on a total production run of 200 ship sets. As shown in the table, the cost of spars systems for the two configurations are significantly less due to the incorporation of the lower cap into the skin which simplifies machining. This savings is offset, however, by a 36 percent increase in interior spar costs. The relatively high cost of these formed sheet metal components was due to present equipment size limitations which necessitate producing them in three 12-foot lengths and splicing to meet span requirements. The use of formed sheet metal ribs in the ATW configuration resulted in a large cost reduction.

Production Quantity Ship Sets	Cumulative Average Recurring Cost - $ per Unit (Variance with Baseline - Percent)	
	Baseline F-111 Construction	ATW 0100 Laminated Aluminum
1	$555,700	$492,200 (-11.4 percent)
4	477,000	418,000 (-12.4 percent)
40	326,9900	278,700 (-14.7 percent)
100	280,600	236,400 (-15.8 percent)
200	250,000	208,600 (-16.5 percent)

TABLE 2 WING BOX COST SUMMARY

Structural Element	Cumulative Average Recurring Cost ($ per Unit) Based on 200 Ship Sets	
	Baseline F-111 Construction	ATW 0100 Laminated Aluminum
Front and rear spars	$11,340	$6,440
Interior spars	17,330	23,600
Ribs	40,090	8,250
Upper Skin	18,740	14,220
Lower Skin	19,990	32,090
Pivot Fitting	68,520	68,520
Assembly	73,990	55,500
TOTAL	$249,990	$208,618

TABLE 3 STRUCTURAL ELEMENT COST SUMMARY

ATW upper skin costs are reduced by approximately 24 percent through savings in material and machining achieved as a result of the Vee spar concept. Costs for the laminated aluminum lower skin are significantly higher than the baseline due to the complexity resulting from integrating the lower spar caps into the skin. This added fabrication cost will still be cost effective in view of reduced lower skin maintenance man-hours.

VI. ATW BONDED WING SUMMARY

The Advanced Technology Wing program successfully demonstrated the feasibility and potential payoffs of a new approach for aircraft wing structure. Its features included: (1) an adhesively-bonded metal laminate lower skin, (2) elimination of all fasteners in the lower skin and reliance on adhesive bonds to carry design loads, (3) a Vee-shaped internal spar arrangement utilizing formed sheet metal components with a sinewave configuration. The payoffs include: (1) lower assembly weight resulting from savings in the lower skin and spars, (2) reduced costs, and (3) improved fuel sealing resulting from elimination of fastener leak paths.

Testing was severe in that 4.5 lifetimes of equivalent fatigue loading were demonstrated by first applying the required two lifetimes and then installing intentional flaws and conducting damage tolerance testing until failure of a structural element. This was accomplished with high design stresses that were verified by strain gauges during the test. In addition, fuel under pressure was maintained in the box during the 8-month cycling. It was concluded that the test article met all the structural requirements of MIL-A-008866 and MIL-A-83444.

Design features that provided high payoffs included:

- Sine Wave Vee Spars
  - Structurally efficient for shear
  - Vee-shape eliminated lower skins joints when compared to conventional thick skin/multiple spar
- All bonded laminated lower skin assembly
  - Elimination of fastener holes increased allowable skin stress
  - Potential fastener leak paths were eliminated
  - Fail-safe multiple load path feature was achieved in the design

Results of the test box fabrication effort demonstrated that structures of this type can be successfully produced using current processes and equipment. Additionally, analysis indicated that it can be done cost effectively on a production basis and that the durability of such structures is enhanced therefore lowering maintenance costs.

VII. METALLURGICALLY BONDED LAMINATES

Although the term "laminated metal structures" usually creates an image of metal layers bonded together with an organic adhesive, there are other forms of lamination. Vought Corporation has pioneered a new technology in which the layers of primary alloy are metallurgicaly bonded together through a soft interleaf of a secondary alloy. This material concept removes many of the objections conservative designers have used to avoid the use of an adhesively bonded design. There is no unstable low temperature moisture sensitive and heat sensitive critical layer. Vought has investigated the use of roll bonding, explosion bonding and diffusion bonding to bond multilayered laminates to form these truly all metal systems.

The key characteristic that a laminate must have in order to show improved durability is that the individual layers exhibit plane stress behavior and not plane strain behavior, as was discussed above. Vought has prepared a variety of materials systems illustrating this concept including those listed in Table 4.

Primary Alloy	Interleaf	Strength (ksi)	K_{Ic} (ksi-in$^{1/2}$)	Laminate Toughness (ksi-in$^{1/2}$)	Improvement
300M Steel (0.16" Layer)	1020 Steel	250	61	135	121 percent
300M Steel (0.061" Layer)	1020 Steel	236	61	209	242 percent
7075 Aluminum	7072Al	76	40	60	50 percent
7475 Aluminum	1100 Al	77	60	90	50 percent
Titanium-10,2,3	Ti-15,3,3,3	187	38	88	114 percent
Titanium-6,4	6061 Al	140	57	124	117 percent
7475 Aluminum	FM-73M Adhesive	71	60	85	41 percent

TABLE 4. FRACTURE TOUGHNESS IMPROVEMENT IN LAMINATED HIGH STRENGTH STRUCTURES

The very high fracture toughness values for the 300M alloy laminates are especially impressive for a low alloy steel. In fact, these values exceed those for AF1410 at equivalent yield strengths. When one considers the high cost of the AF1410 alloy, it is easy to see that it can be cost effective to substitute a laminated material such as the 300M/1010 steel system for AF1410.

The 50 percent improvement in fracture toughness shown for aluminum is technically significant. The achieved fracture toughness is 100 percent of the theoretically possible gain based on the thickness versus toughness relationship shown in Figure 4.

The retention of the already high toughness of titanium at significantly higher strengths allows consideration of titanium for selected application which are now being filled by steel. The 40 percent weight reduction possible when making that substitution offers significant fuel savings as well as performance improvements.

When these toughness values are plotted against yield strength and compared with representative monolithic alloy fracture toughnesses, the benefits of these materials offered to aircraft design are quite clear (Figure 11). As an example, design stress levels can be significantly increased allowing a corresponding weight reduction (Table 5). In addition, the critical flaw size, a_c, is shown to be from 5 to 10-fold larger which can be very cost effective by allowing extension of NDI intervals as well as improving the level of confidence in detecting flaws well before they reach critical dimensions.

The data shown in Table 4 and Figure 11 were measured using ASTM E399 on specimens prepared from flat plates. One could also tailor laminates to meet specific mechanical wear or loading environments. Also, one is not limited to flat plates. Figure 12 shows a roll-bonded laminate which has been forged. There was extensive metal flow without any loss of the metallurgical bond quality. In fact properties of the forged laminates were as good as the roll-bonded blanks.(7)

Design Stress (ksi)	Assumed Weight (lbs)	a_c* (inch)	Weight Reduction
180 (monolith)	100	0.029	---
200 (laminate)	90	0.28	10 percent
220 (laminate)	82	0.23	18 percent
240 (laminate)	75	0.19	25 percent
260 (laminate)	69	0.16	31 percent

*Fracture toughnesses for monolith and laminate from Tables 1 and 4.

TABLE 5. BENEFITS OF SUBSTITUTING 300M STEEL LAMINATE FOR MONOLITHIC STRUCTURES

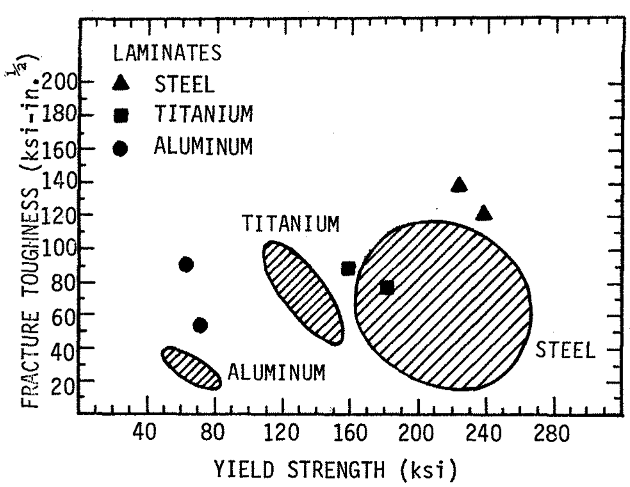

FIGURE 11. TOUGHNESS OF TYPICAL AEROSPACE ENGINEERING ALLOYS

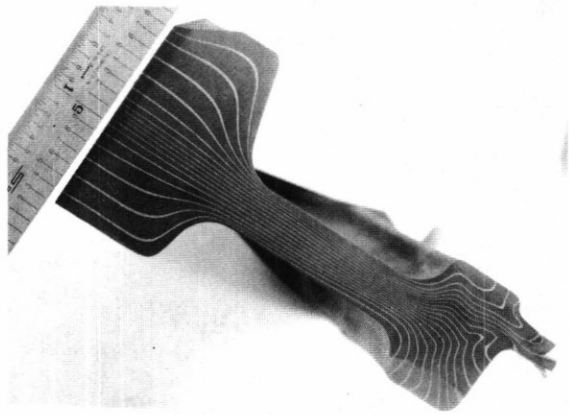

FIGURE 12. 300M/1010 STEEL METAL LAMINATE CLOSED DIE FORGING CROSS SECTION (MACROETCHED IN DILUTE NITRIC ACID)

VIII. CONCLUSIONS

Laminated metal structures can be fabricated cost effectively by adhesively bonding metal sheets or by roll-, diffusion or explosion-bonding metal interleaved stock. Using either the adhesively bonded or metallurgically bonded system, the primary advantages of laminate structures are durability and damage tolerance. The choice of fabrication method is dependent on the final component geometry and the environmental and other service requirements of the structure. In any case, the incorporation of laminates into aircraft design can be cost effective if capital investment advantages, fuel cost savings, improved durability, and reduced maintenance expenses are considered. Developments in adhesives, bonding practices and structural analyses combined with the results of component testing and the establishment of design allowables have made adhesively bonded laminates a state-of-the-art technology. The concept of metallurgically bonded laminates is considered to be an emerging technology that will provide decided improvements in future aircraft durability, economic life, and life cycle costs.

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