Developing the "Backbone" of the F-14

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This paper presents the major features in the development of high-performance, electron beam (EB) welded titanium structures, emphasizing the detailed design of high-efficiency joints, material selection, and processing. The selection of operating weld stresses and solution of associated EB welding problems also are discussed. Related test programs required to support this design effort, such as flaw growth rate testing, photoelastic model testing, and full-size static and fatigue development tests, are discussed at length.

I. Introduction

THE variable-geometry concept requires that a wing center section support the movable outer wing panels (see Fig. 1). These outer panels transmit large, variable bending, and torsional moments as the wing is swept back and forth. The pivot was located so as to minimize the adverse pitching moments associated with swing wings, especially at high altitudes and high turning rates – the dog-fight environment. In addition, since the wing is located at the maximum cross-sectional area of the airplane, aerodynamic considerations (drag, area rule, etc.) required that the wing box follow aerodynamic air passage contours. Thus, the G-14 wing center box became a 20-ft-long, gull-shaped structure (see Fig. 2). Constructing this long, complex "aerodynamicist's dream" was another matter. Since this was to be the structural heart of the aircraft to which the fuselage, nacelles, and outer wings were attached, dimensional stability was of paramount importance. Bolting would have meant heavy weight penalties and space problems, and , prior to the F-14, welding usually meant D6AC steel with its process control and manufacturing difficulties plus cost and weight penalties (see Fig. 3). Clearly, if the F-14 was to be the Navy's air superiority fighter for the 70's and 80's, advanced material and/or construction technology was mandatory.

II. Early Developments in EB Welding of Titanium

In the early 1960's, when electron beam (EB) welding was a scientific curiosity, Grumman personnel began welding most of the commonly available structural materials, particularly titanium alloys, in a small vacuum chamber. It was soon discovered that 6A1-4V titanium had unusually good weld characteristics, including the ability to join the thick sections with a minimum of distortion – an exceedingly important airframe requirement for high-performance aircraft. One-pass EB welding of thick sections also promised to eliminate the time consuming multiple passes and inspections required with conventional welding procedures (see Fig. 4). By the mid-60's FX/VFX (later to become the F-14) trade studies showed that substantial structural weight savings could be achieved by using titanium. A singular effort was focused on the wing center section of the variable-sweep-wing fighter where high-efficiency joints could further reduce weight.

By 1968, Grumman had designed, fabricated, and tested a thick-section EB welded 6A1-4V titanium box beam, representative of the fighter's wing box beam. This was the first time a major modern aircraft structure included welds in tension as part of its basic design. This box was fatigue tested using a severe spectrum. After surviving the desired life, test load levels were increased 12% to cut down the test time. Failure finally initiated at a tool mark in the lower cover; there were no failures in any of the welds.

In its VFX proposal, the company outlined, and subsequently implemented an objective and systematic plan for wing-box development. Material and process control of titanium, including cleaning, EB welding, machining, stress relieving, and descaling, had a profound influence on the basic design.

Detail design, accurate knowledge of internal stresses, good material properties, tight process control, and quality manufacturing are the key factors that control the fatigue life of structures. For conventional structures, these practices are well-known. For a large EB welded wing box, however, these processes required a quantum jump in technology.

III. Development of F-14 Wing Center Section

Getting started in the EB welding business required sizeable dollar investments in special equipment not normally used by airframe manufacturers. About three million dollars were spent for three large welding chambers (two end-loaded tunnels 25×9×11 ft, and one clamshell, 32×10½×8 ft). The powerful electron guns in these chambers are designed to operate at voltages up to 60 kV, current up 500 m, and speeds up to 100 in./min. Prior to welding, the chambers are evacuated to a pressure of 10^{-4} mm of Hg. The evacuation takes less than 20 min. This facility has been operating for over three years, producing five F-14 wing boxes per month with ease.

Stress Relieving

Generally speaking, processes (such as EB welding) that produce narrow fusion zones have the highest residual stress and steepest gradients after cooldown. Residual stresses approaching the yield point have been measured in EB titanium weldments, but they have been stress relieved with minimum distortion. Published data record that residual stresses of annealed Ti-6A1-4V are reduced to negligible values when the material is held at 1200°F for four hours. Accordingly, the complete wing box assembly is stress-relieved to these parameters. During this process, a tenacious oxide is formed which has been found to be detrimental to the fatigue life of the structure. Therefore, after welding, the entire box is descaled by dipping it in an acid solution to restore base metal properties. The box is given then a light shot peening in order to obtain a residual compressive stress in all surfaces.

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Index categories: Aircraft Structural Design (including Loads); Aircraft Structural Materials.

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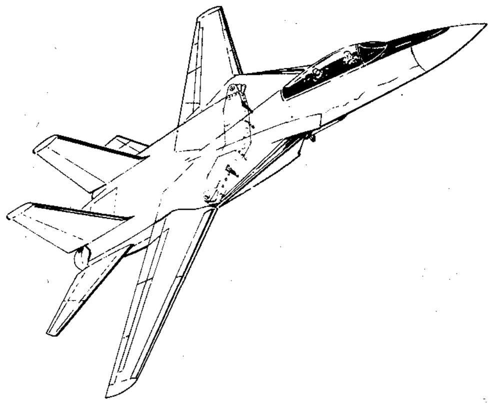

Fig. 1 Location of wing center section in F-14.

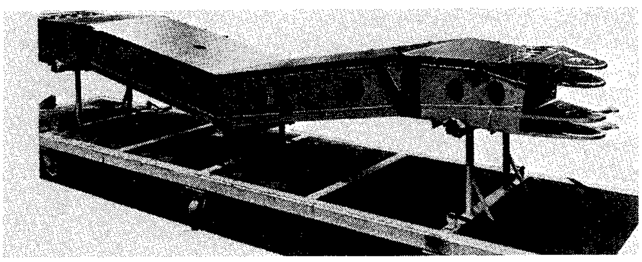

Fig. 2 F-14 wing center section or "backbone."

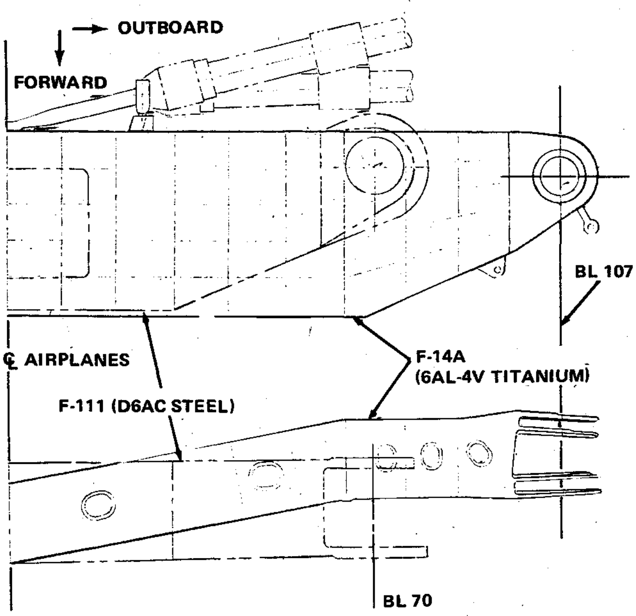

Fig. 3 Comparison of F-14 and F-111 wing center sections.

Nondestructive Testing/Inspection

Nondestructive tests (NDT) with ultrasonics, x-rays, and penetrants were found to be unreliable when used by themselves. Thus, procedures were developed to permit the various NDT techniques to complement each other. A gross inspection technique is applied immediatley after welding. This consists of: a visual inspection for underfill; root side suck-back; penetration; and, most important, witness lines for possible missed seams (see Fig. 5). If negative results are obtained, the part is rewelded; otherwise it is machined to remove surface irregularities and x-rayed for possible internal voids. The assembly then is ultrasonically inspected for missed seams and internal voids. Finally, after stress relieving and descaling, the wing box is die-penetrant inspected for surface flaws.

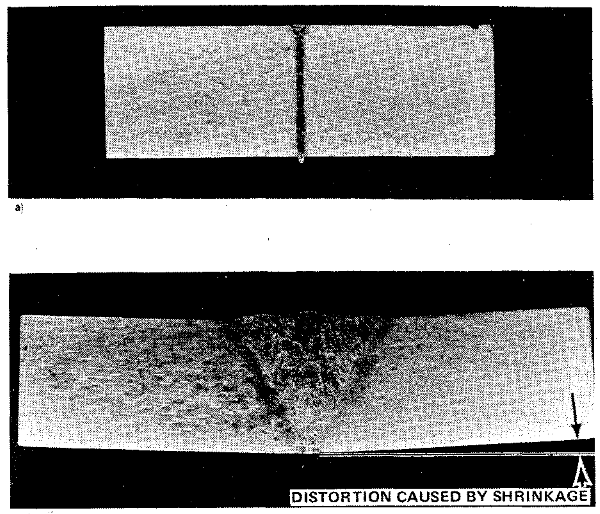

Fig. 4 EB weld vs conventional weld in Ti-6Al-4V: a) EB weld (1 pass); b) gas tungsten arc weld (75 passes).

Material Control and Treatment

A preminum grade of annealed Ti-6Al-4V material is used for the lower covers and all of the pivot platters. This material is purchased to a specification that controls the raw-materials input to the ingot as well as all melting and forging parameters, and has a guaranteed minimum fracture toughness(K_{IC}) of 60 ksi$\sqrt{\text{in}}$.* Early experience in producing thick welds indicated that hydrogen in the titanium out-gassed during the welding (melting) process and created gas bubbles or porosity. Therefore, all detail parts undergo vacuum degassing to reduce the hydrogen level from 130 to 75 ppm.

Fracture Control Program

In order to insure that production processing procedures are equivalent to those used during the development stage, Grumman initiated a self-imposed fracture control program – the first in the aerospace industry. For the first production run, fracture toughness specimens were cut from parent material of the most critical machined parts upon receipt of material. One set of specimens was tested in the as-received condition, and one set accompanied the parent part through all processing, including JEB welding, and then was tested. The results of these tests (see Table 1) provided verification of design data and proved that production processing did not have any adverse effects on the design properties.

In the early development stages, it was recognized that a strong fracture control program was needed to support the design. Not only is it important to control manufacturing processes to minimize flaws and to have adequate NDT procedures to identify them, but it is equally important to determine flaw growth characteristics under in-service conditions of loading and environment, so that intelligent decisions can be made on flaw or defect disposition during fabrication. To this end, a program was funded by Grumman at the Battelle Laboratory to determine the growth characteristics of flaws in EB welded Ti-6Al-4V sheet and plate.

The variables selected in this fracture test program covered the expected range of stresses, stress amplitude spectra, flaw configuration, and environments related to service life. The welds studied included both transverse and transverse with an intersecting longitudinal weld. Flaws were studied both in the weld and in the heat affected zone.

Test loadings at Battelle were varied based on Navy service experience. The variations consisted of aircraft useage, constant and varying minimum load levels, overload effects, and strain rates. These anomalies produced no discernible effect on the basic spectrum. Excellent results were obtained; all test

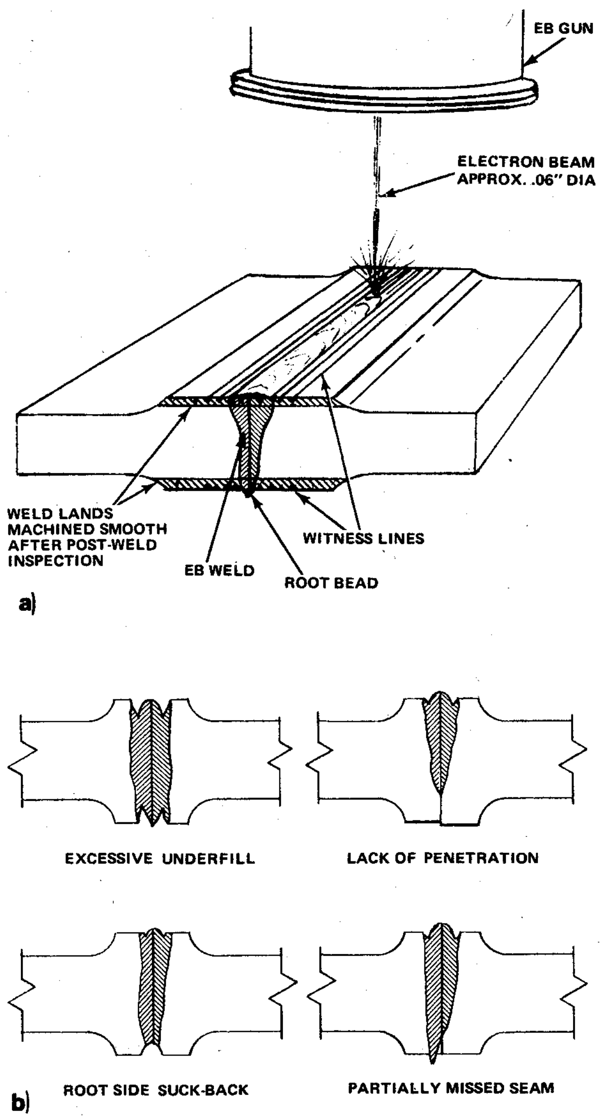

Fig. 5 Details of EB welding and defects screened by visual inspection techniques.

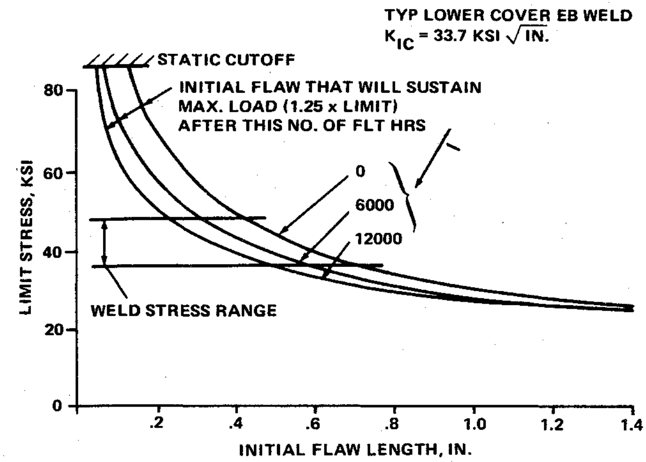

Fig. 6 Life of Ti-6A14V weld as function of stress and initial flaw length: a) details of Eb welding; b) welding defects.

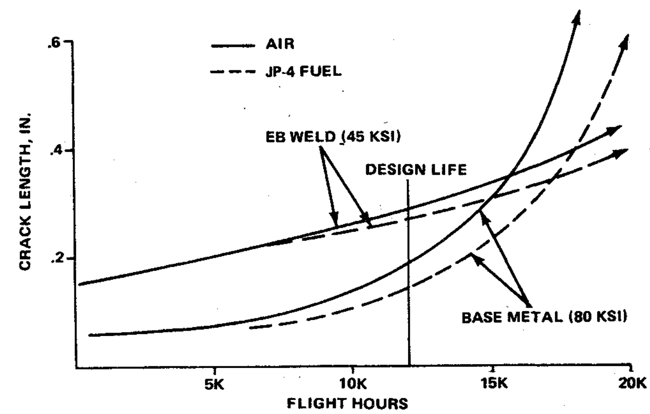

Fig. 7 Environmental flaw growth rate, Ti-6A1-4V plate.

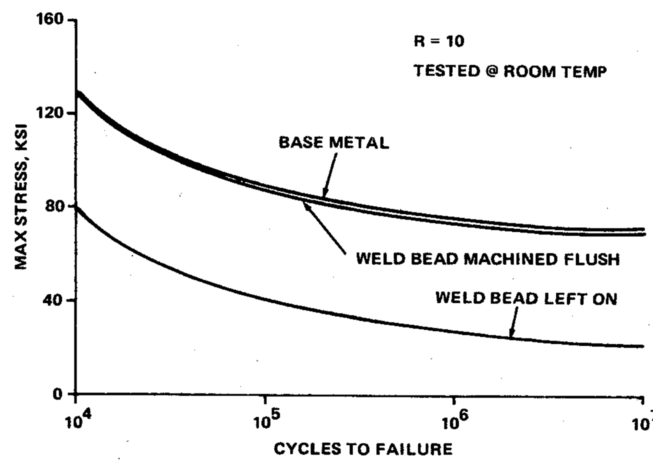

Fig. 8 Effect of EB welding on fatigue strength of annealed stress relieved Ti-6A1-4V plate.

lives fell within a limited scatter band. The data that are plotted in Figs. 6 and 7 give some of the results of this program, showing the allowable initial flaw size and the critical crack length for the design fatigue spectrum vs weld stress. For a weld stress of 45,000 psi, for example, the allowable initial flaw size is 0.270 in. This will grow to a critical length of 0.470 in. after 12,000 hr of severe spectrum fatigue. For the actual range of weld stresses shown, rather large-size flaws can be tolerated. Acceptable flaw sizes used in production for inspection control are set below these values by at least a factor of two.

In static and fatigue tests, an EB welded and stress relieved specimen with the bead machined flush exhibited 100% of parent metal strength (see Fig. 8), but fracture toughness in the immediate vicinity of the weld was found to be about 56% of the parent metal (see Fig. 9). In order to overcome this disadvantage and maintain a consistent level of resistance to a brittle-type fracture, the stress levels of the weld were reduced to 56% of the allowables used in the parent metal by use of symmetrical weld lands. Preliminary weld stresses were established by use of the relative fracture toughness of the weld and parent metal to allow the same size critical crack length in both. The weld metal initial crack size tolerance and stress levels were verified subsequently by tests simulating aircraft loading conditions.

Wing Center Section Design

The F-14 wing center section is a single-cell box beam. It consists of four basic weld assemblies fabricated from thirty-three detailed machined parts. The box transmits wing outer panel loads (applied at each pivot) into the fuselage nacelles and centerbody. Seventy EB welds in twenty-five setups are required to fabricate each wing box. Fifty-seven welds are square butt joints and thirteen are angles or scarf joints. Weld thickness ranges from ½ to 2 in., and the longest weld is 65 in. The open -box configuration shown in Fig. 10 is the result of intensive design efforts to guarantee accessibility for inspection of all welds during manufacture.

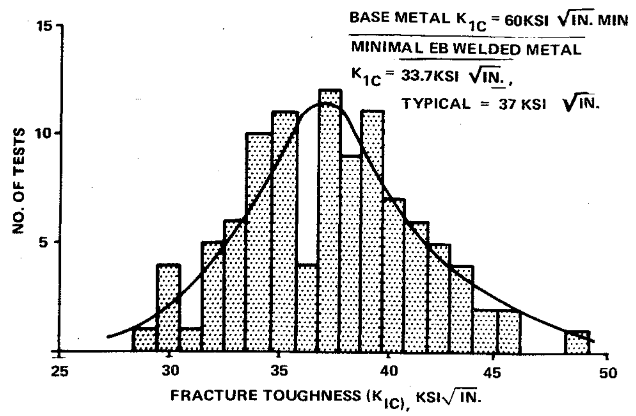

Fig. 9 Results of Grumman fracture toughness tests, using stress relieved EB welded Ti-6Al-4V plate.

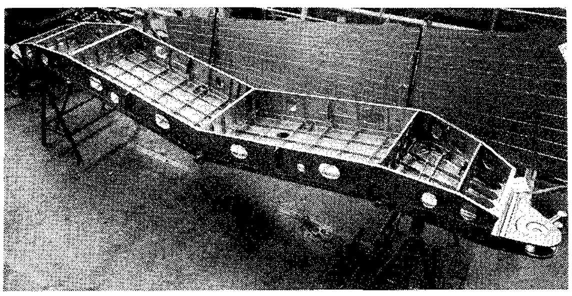

Fig. 10 Open-box configuration.

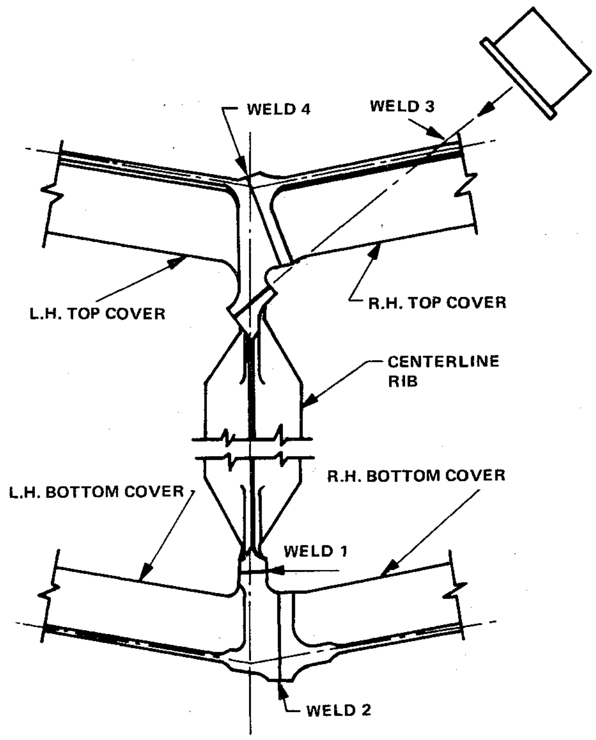

Fig. 11 Wing box centerline rib joint details.

A fundamental type of joint, affected by primary structural and manufacturing considerations, exists in the wing center box at the intersection of the full-depth ribs (see Fig. 11). The ribs must be located so that load lines intersect at a common point. Conversely, the weld plane joining cover-to-cover must be located as close as possible to the rib to eliminate large weight penalties, yet the offset must be large enough to permit beam exit, machining, and, most important, adequate inspection of the welds. The ribs carry the compression and tension loads generated by the cover stiffeners. In order to eliminate thick rib welds, the rib stiffeners (2 in. deep) are properly tapered to a 0.40-in., finish machined weld thickness.

Eliminating EB Welding Problems

A few major problems associated with EB welding have been designed out of the F-14 wing box. Solidification defects associatied with starting and stopping have been eliminated by the use of external tabs, integral or welded in place, and start-stop plugs for rib intersection holes. Tabs and plugs are machined off after weld inspection acceptance.

Spiking, or cold shut, is caused by partial penetration, solidification, and contraction, leading to minute voids and/or cracks, which are difficult to detect by conventional means. Figure 12 shows the sequence of operations used to weld two modules, each composed to two 1-in. beam webs and a 2-in.-thick cover. A 5/16-in. diam intersect hole is drilled to provide beam exit for the first (cover-to-cover) weld. The second and third welds then are made by starting at the free end of each web and ending in a stop tab as shown. This precludes defects from occurring in the lower cover at point A. Then, based on x-ray film shots of the intersect hole and the inside surface of the beam, the hole is enlarged to a specified diameter and the external tab is machined off. In order to counteract the stress concentration of this open hole, a symmetrical weld land is provided, so that the total thickness at the land is roughly twice the thickness of the lower cover cap. Then an interference fit, A-286 pin is driven into the hole. The pin alone brings the effective stress concentration from a factor of 3.0 for an open hole to approximately 1.6.

Table 1 Wing fracture toughness comparison

A/C	Material	Max stress (fatigue test) (f_F), ksi	typ K_{IC}[d]	Fract. stress[a] (f_M), ksi	Fig. of merit (f_m/f_F)	Service exper-ience
A	2020-T651	1.24×33=41	22	41.	1.0	poor[c]
B	7075-T651	1.2×34=42	27	51.5	1.22	fair
C	7075-T6	1.0×45=45	27		1.14	accept.
D		1.0×40.7=40.7			1.26	good
E		1.0×37=37			1.17	accept.[c]
F		1.25×35=43.7			1.39	good[c]
G	D6AC	1.0×123=123	45[b]	86.	0.70	unaccept[c]
F-14	6Al-4V	1.25×84=105	70	133.	1.27	good
	weld area	1.25×44.5=55.6	37	70.6	1.27	good

[a]To rupture A/C at max load. [b]Expected 90. [c]Service experience unknown; judgement based on Fig. of merit. [d]A measure of a material's ability to sustain load in the presence of a flaw. Plane stress-stress intensity factor (K_C) is a measure of lowest load at which significant crack extension occurs.

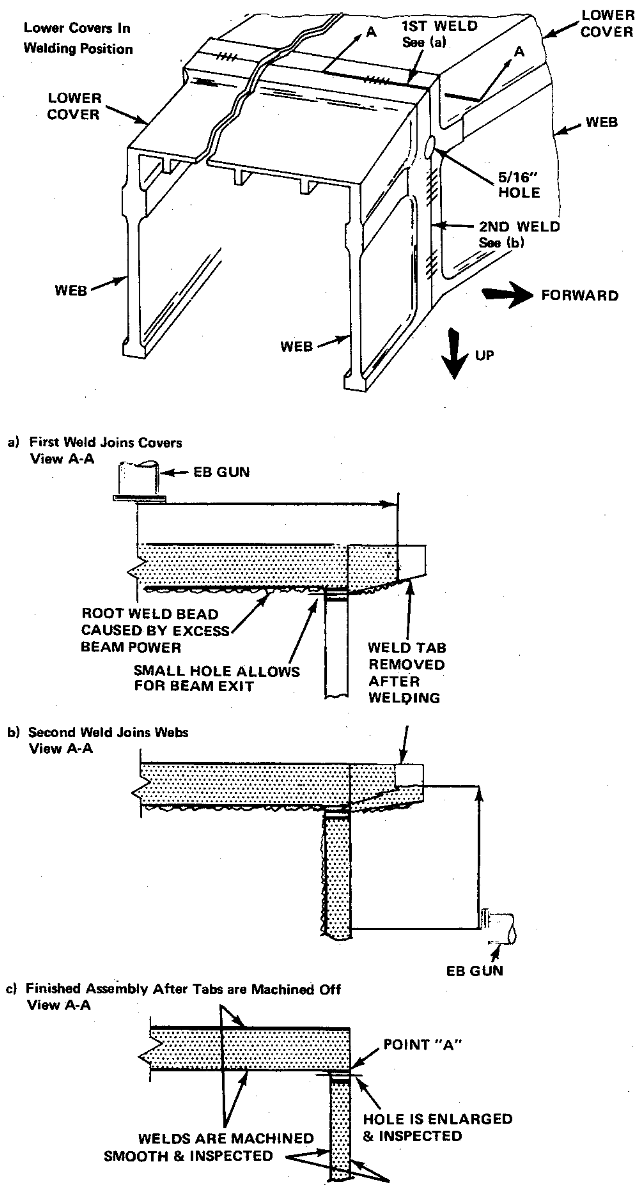

Fig. 12 Procedure for welding two lower covers.

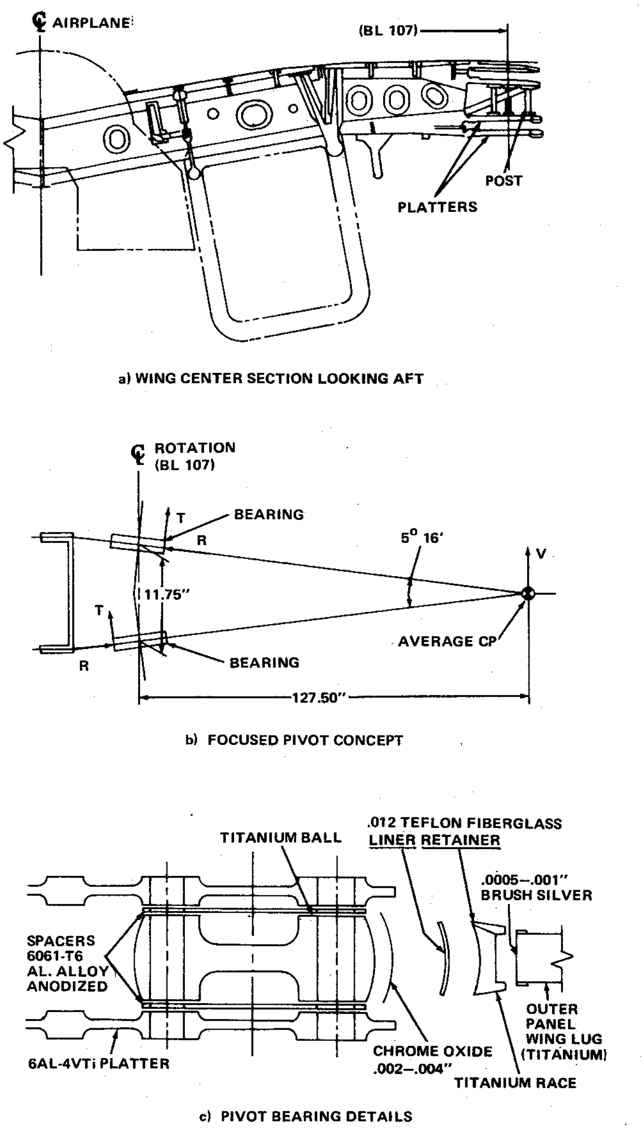

Fig. 13 wing center section and pivot details.

Hole Considerations

In addition to welding details, particular attention was given to stress concentrations due to holes in the lower cover of the box which is subject to spanwise tension. The only holes permitted are weld intersection and mandatory fuel fill and drain holes. In order to minimize stress concentrations, two- and three-dimensional photoelastic models and finite-element analyses were used to tailor local reinforcements. Furthermore, only oval access holes are used in the beam webs to permit attachment of clamp-on-type covers, eliminating the stress concentration of loaded holes at the edges of access holes.

Pivot Design

One of the most demanding design problems in the wing box was the pivot (see Fig. 13a). A focused-pivot configuration was used with a double-shear connection at the upper and lower wing covers. The lugs are not parallel, but converge so that their axial loads intersect at an average wing center of pressure (CP) (see Fig. 13b). Wing shear is carried primarily by the resulting truss action, thus minimizing shear loads on the lugs. Since the wing CP moves fore and aft as the wing sweeps, the lugs also are inclined aft at an average wing sweep. Spherical bearings (see Fig. 13c) provide for the continuing relative change of plane of the wing as it sweeps (misalignment). Since there is no through-pin, the space between each pair of lugs is used for running lines out to the outer panel. The upper compression platters are post-stabilized (Fig. 13a) against buckling. In the high-wing-sweep condition, differential bending causes the platters to move fore and aft relative to each other. The post pivot axes are lined up so that no secondary moments are developed in the posts by this motion.

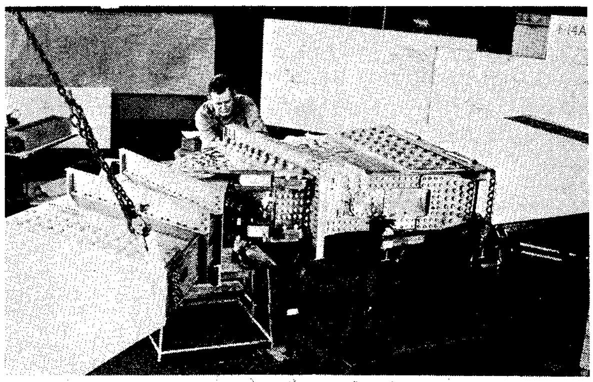

Fig. 14 Early pivot bearing test specimen [note size (weight) and space requirements of bolted construction].

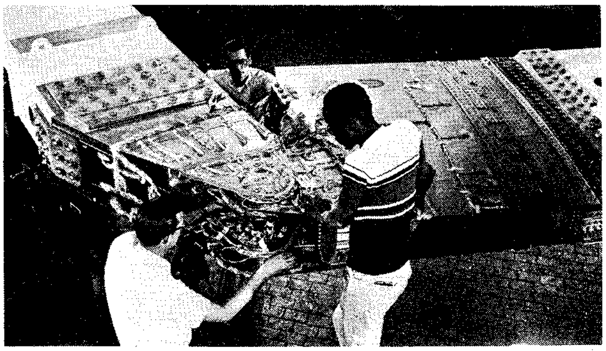

Fig. 15 Wing pivot specimen, test 9 (represents pivot region only).

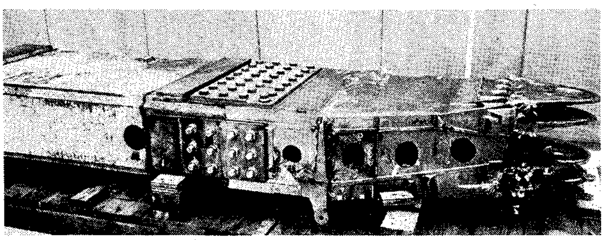

Fig. 16 Wing pivot specimen, test 9A (represents final configuration up to nacelle attachment fitting).

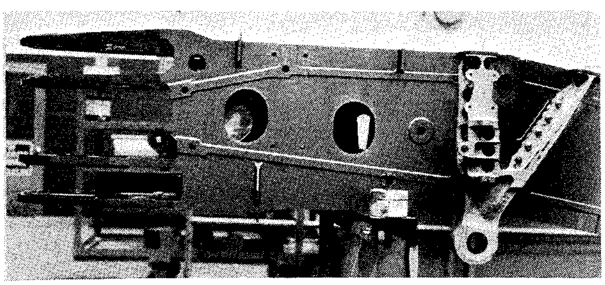

Fig. 17 Closeup of final design showing nacelle attachment fitting (note weld lands and weld intersection holes).

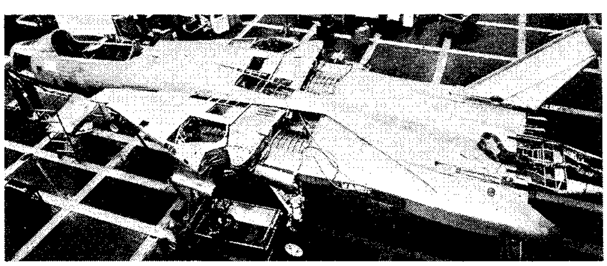

Fig. 18 Static test airplane in test lab (note how wing center section follows aircraft contours).

IV. Static and Fatigue Tests

Full-scale static and fatigue testing was a vital aspect of the comprehensive structural development test program, which supported design of the F-14 wing center section. A total of five major development tests (Figs. 14-18) contributed significantly to the success of the final production design and proved conclusively that targeted performance levels were achieved. The first fatigue test was conducted to investigate the bearing, lug, and platters, and was representative of the structure in those areas only. The test article (see Fig.15) failed at approximately 80% of design life. Primary failure initiated between bolt holes in the bottom platter. Cause of the failure was an uneven bolt load distribution due to the large tolerance specified in an attempt to obtain interchangeable bearings. A secondary failure occurred at a partially missed weld seam in platter no. 3, not detected during fabrication inspection. As a result of this test, engineering specified line reaming the bolt holes, thickening the platters in the bolt-hole regions, and improving the EB witness line technique to insure detection of missed seams. In order to assure that existing NDT technology adequately controls fracture requirements and guarantees fatigue life, the first thirty-nine wing boxes were proof-tested successfully to 125% of limited load.

The second fatigue test article (Fig.16) was representative of the final pivot design and transition into the stiffened covers. The test was stopped after reaching the equivalent of six service lives! Cracks were found in the unmachined, nonstress-relieved upper shear welds. (It had been considered unnecessary to stress relieve top cover welds, which are primarily in a compression region.) The cracks were attributed to high tensile residual stresses in the unmachined weld beads, causing local areas to cycle in tension.

Unmachined, nonstress-relieved upper cover welds also were present in the center box at this stage of the design. These welds, however, were working at a much higher stress level than those in the pivot transition area. In order to preclude failure in any cover welds, the assembly procedure was modified to include final machining of all cover welds and stress relieving the entire final assembly. In order to check for ultimate strength and lug and platter instability, a full-scale static test of the pivot area also was performed. Ultimate lug loads up to 1.1×10^6lb were applied successfully with no evidence of instability or structural failure.

The major feature of the development test program was a fatigue test of the complete wing center box. This test demonstrated the fatigue life of the wing center box by subjecting the test specimen to a service life of 6000 hr multiplied by a scatter factor of 2, or 12,000 hr of extreme-use flight loads. The specimen was virtually hidden by the maze of test equipment used during this test.

No problems were encountered with the wing center box itself. In fact, after the 12,000 hr, the condition of the box was satisfactory enough to permit its installation in one of the two static test qualification airplanes. Problems did arise, however, in the nacelle attachment fittings. The aft outboard attachment fitting failed after 1200 hr, and galling of the attachment pins (nickel-plated pins in stainless-steel bushings) also occurred. The failed fittings were redesigned using silver-plated attachment pins in aluminum bronze bushings to prevent fretting.

V. Results to Date

To date, over one hundred-eighty wing center section boxes have been electron-beam welded, one hundred thirty-five aircraft have been delivered, and over 10,000 flight hours have been accumulated. The F-14 has flown to speeds in excess of Mach 2.0, pulled over 7 *g's* in maneuvers, and landed at sink speeds in excess of 24 fps. In addition, the 2065-lb wing box is 19 lb lighter than its target weight, and has become the second least expensive structure to produce (in terms of $/lb) for the F-14A. This state-of-the-art advancement in the construction of high performance aircraft was the result of an extraordinary detail design effort, involving untried combinations of materials and manufacturing processes.