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AERONAUTICAL RESEARCH LABORATORY

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Research Report 12

DAMAGE STATES AROUND FASTENER HOLES IN THICK GRAPHITE/EPOXY COMPOSITE LAMINATES

by

S.C. GALEA
G.K. DEIRMENDJIAN
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SUMMARY

This Report presents qualitative studies of the development of damage around countersunk fastener holes in thick graphite/epoxy laminates which have been subjected to fatigue loading. The experiments have shown that damage at the hole bore surface produces sites for the nucleation of delaminations. The morphology of the delaminations and ply cracking was mapped extensively. This Report discusses the methods used for the mapping of the damage and presents detailed maps of the damage for two loading conditions. In addition several methods for the non-destructive monitoring of damage development and growth were investigated. These were load/displacement hysteresis measurements, fastener tilt measurements using shadow moiré fringe techniques and ultrasonic time-of-flight C-scanning.

This Report briefly discusses the possible mechanisms by which damage develops and grows around the fastener holes. From the damage maps it was found that the volume of material around the fastener hole, damaged by the fatigue loading, adopted a characteristic shape; the volume of damaged laminate increased towards the faying surface of the laminate and (metal) fixture. This characteristic damage volume was generated by the fastener rocking under the fatigue loads. The growth of the delaminations has been shown to be preceded by intra-ply cracking and, as fatigue loading continued, more delaminations were generated at the hole bore surface.

The experimental work has shown that damage development around fastener holes is a complex process, usually producing several delaminations in the region of the fastener hole which grow and may ultimately lead to the failure of the joint.

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1. INTRODUCTION

The use of advanced composite materials in modern military aircraft has increased significantly in the last 10 years. Aircraft such as the F/A-18 and AV-8B use significant amounts of graphite/epoxy laminates in components such as wing panels and doors. These wing skin laminates are often quite thick, (up to 50 plies) and are attached to the aluminium alloy, titanium alloy or composite airframe structure with titanium alloy or steel fasteners as illustrated in Figure 1. With this type of design, quite large loads can be imparted to the wing skin through the fasteners and consequently damage can be accumulated at the fastener hole. At the present time, however, the strains imparted to the composite wing skins are usually kept low, by design, typically of the order of 3500 to 4000 micro-strain. However, if the weight savings expected by the use of composite laminates are to be fully realised, then the strain levels in structural laminates must be increased further. This may lead to problems in the certification of aircraft structure and so the behaviour of composite structures at high strains is an important area of study. The major problem in assessing fatigue-life behaviour of mechanically fastened joints is the difficulty in visualising the morphology of damage generated at the fastener hole and then predicting the fatigue-life behaviour of the component (or coupon). Typical failure modes of mechanically fastened coupons are illustrated in Figure 2. Some failures of the types shown in this figure are induced by fatigue loading and are investigated in the following experimental program.

Various types of fatigue damage have been observed at fastener hole locations in thick laminates. These are hole wear (viz., change in dimensions of the hole bore), laminate break-up at the faying surfaces and delamination growth around the fastener hole. Additionally, overload failure modes such as bearing failures, shear failures and fastener failures often occur in multi-fastener systems where in one region the composite or fastener fails in fatigue and the load on the remaining fasteners is increased leading to overload failure of the graphite/epoxy laminate and/or remaining fasteners.

Of particular concern in the present work is the nucleation and growth of damage within the laminate surrounding the fastener hole. While the growth of delaminations around fastener holes has been the subject of a number of investigations [1-3], the mechanisms of delamination nucleation and the morphology of damage development have not been examined in detail. The potential for damage (delaminations) to develop around fastener holes, during hole preparation, has been investigated by Condon [4]. In his work the effects of anomalies in fastener hole preparation on the fatigue lives of mechanical joints were measured. It was found that the most significant defect was the delamination nucleated around the hole by the drilling process. This incipient delamination gave a measurable decrease in the fatigue life and static failure load of a mechanical joint. Crews [5] also examined the effects of incipient delaminations around fastener holes and showed that these grew under fatigue loading.

Delaminations around some fastener holes can also arise from the assembly of the component. For example, delaminations were found to have been generated around some of the fastener holes during the assembly of the F/A-18 wing, Figure 3, Bishop and Morton [6].

Rubin [7] investigated the effect of multi-level delaminations caused when unshimmed gaps between skin and substrate are pulled together during fastener installation. The failure criteria used for the bearing stress allowable was hole elongation (maximum elongation of 4% of the hole diameter) or laminate failure. Bearing strength tests were undertaken at 93°C/1% moisture condition for damaged and undamaged laminates. It was found for fastener diameter/laminate thickness ratios of less than 1.1 no effect occurred. However ratios greater then 1.1 sustain a 15% reduction in maximum static bearing strength for laminates with damage compared to undamaged laminates. Incipient multi-level delaminations at fastener holes may also arise from impact damage.

To the knowledge of the authors, the effects of incipient damage on the fatigue-lives of mechanical joints has not been reported. However the effects of impact damage on the static and fatigue properties of carbon fibre composite laminates has been the subject of extensive research [8]. At the present time it is not known if these incipient delaminations grow during service and this requires investigation, but experience with graphite/epoxy laminates suggests that, at the present operating strain levels, these delaminations do not pose a structural problem. Incipient delaminations at fastener holes were not studied in the present work.

The aim of the reported work was to study the morphology of damage generated around fastener holes in a simple specimen which simulated a mechanical joint. The main objective of the work was to map the progression of fatigue damage with increasing numbers of blocks of an aircraft loading sequence. This mapping presented a detailed representation of damage morphology through the laminates. Additionally, non-destructive methods for the monitoring of damage progression during fatigue were investigated.

2. EXPERIMENTAL PROGRAM

2.1. Material

The material studied in the present work was AS4/3501-6 as a 6.7mm thick laminate of 50 plies. The layup used was:

$$[(+45_2,-45_2,0_4)_3,90]_S$$

which was similar to sections of the wing skin of the F/A-18 aircraft.

2.2. Specimen Preparation

The specimen was a small coupon, 50x220x6.7 mm which was incorporated into a fixture simulating a mechanically fastened joint, as shown in Figure 4 [9]. While not the subject of the present investigation, it has been found that the level of secondary bending can influence the fatigue-life behaviour of mechanical joints [10]. The secondary bending level at the maximum test load was 0.32 for the present coupon/fixture arrangement. This was considered realistic and appropriate to composite wing skins [11]; only this one level of secondary bending was investigated. A listing of all coupons prepared is provided in Table 1.

2.2.1. Fastener Holes

Drilling of the fastener holes was performed in two operations. The coupon was first drilled through with a 6.2 mm tungsten carbide three fluted drill and then reamed to the final size using a 12° fluted reamer. The fastener holes were countersunk with 100° countersinks and the holes were relieved at about the mid-ply layer with a 50° countersink. The drilling and reaming processes used in the present work were undertaken in accordance with McDonnell Douglas Process Specifications [12]. The two fastener holes were placed in line along the axis of the coupon; and were designated "A" and "B", see Figure 4. Hole sizes were measured using the micrometer stage on a Wild microscope. The initial hole sizes are given in Table 1. As can be seen from the dimensions given in the table, there was some variability in the initial hole sizes and this was found to have a minor, and somewhat random, effect on coupon fatigue life. Water was used as the lubricant and coolant and also prevented the formation of carbon fibre dust. The quality of the bore surfaces was found to be high with very little fibre pull-out and damage.

2.2.2. Fasteners

The fasteners used in this program were of alpha/beta titanium alloy, designated NAS-664V-7VS. The fasteners were installed in the coupon and fixture and tightened to a torque of 10 Nm. The fasteners were constrained from rotating during testing by lock washers, manufactured for this purpose, and the use of Loctite on the fastener thread. Using this method of fastener installation it was found that there was no reduction of torque during fatigue testing.

2.3. Fatigue Testing

The specimens were subjected to sequence loading. The load sequence used for this program was the McDonnell Douglas (MCAIR) 300 h block (MACSEQ), with an exception in the case of coupon 06-02. This coupon was tested under load sequence SQ1055 which was a sequence derived from RAAF operational F/A-18 loads. This sequence was considered to be more severe than MACSEQ. The peak loads applied to the specimens were derived from MCAIR wing fold strain (load) data and were then factored by 1.12 or 1.34. Thus for the tests two load conditions were investigated; one with a peak tensile load of 30 kN and the other with a peak load of 36 kN. (The one case for which SQ1055 was used had a peak load of 30 kN.)

Some of the fatigue tests were taken to the failure of a fastener, other tests were interrupted and the internal damage mapped, see Section 2.7. The fatigue test history of each specimen studied is also provided in Table 1.

2.4. Load/Displacement Hysteresis

The progression of damage in some of the composite coupons was studied by monitoring the load/displacement hysteresis behaviour between the coupon and fixture. An extensometer was mounted on the specimen, as shown in Figure 5, and this provided measurements of displacement under each loading cycle. This provided a measure of compliance of the fastened joint. While this method was only at a preliminary level of development, a qualitative measure of changes in the specimen could be seen as variations in the shape of the hysteresis plots with increasing fatigue loading.

For the hysteresis studies, the fatigue loading program for a number of coupons was interrupted at various intervals. At these intervals load/displacement hysteresis loops were plotted for a single load cycle with limits at 30 kN in tension and 19 kN in compression.

2.5. Shadow Moiré Fringe Study

During the fatigue loading program selected specimens were studied for any changes in the out-of-plane displacement (tilt) of the fastener heads using shadow moiré fringe techniques, described in detail in [13]. A schematic diagram of the movement of a fastener under load is shown in Figure 6. For this experiment both fastener heads were filled either with a white, two-part epoxy resin (gelcoat) or a polyester body filler. When dry, the excess was polished down to produce an optically flat surface, flush with the coupon surface, which was suitable for the visualisation of interference fringes. In order to visualise the fringes the polyester body filler was painted white. For the griding used in the present work, each fringe represented 0.08 to 0.15 mm out-of-plane displacement.

Fatigue testing was interrupted at various stages in the fatigue loading program. The specimen was loaded statically to load levels of 5, 10, 15, 20 and 25 kN. At these loads photographs were taken of the fringes appearing on fastener heads 'A' and 'B'. A typical shadow moiré fringe result due to fastener tilt is illustrated in Figure 7.

2.6. C-Scanning

A time-of-flight ultrasonic C-scanning method was used for the non-destructive assessment of damage. The method is described elsewhere in detail, [14,15]. The method was sensitive to delaminations only, giving no information on intra-ply cracking. The morphology of the delamination damage within the laminate was visualised as a colour map with eight colours each representing 1/8th of the full thickness. C-scanning thus provided colour coded maps, representing thickness bands within which delaminations lay.

The experimental setup is illustrated in Figure 8.

Each test coupon was C-scanned prior to testing to ensure that a sound coupon was tested. Some of the coupons were scanned upon the termination of fatigue testing. In an effort to observe progression of damage a limited number of coupons was also scanned at various stages of fatigue testing.

2.7. Optical Microscopy

At the termination of a fatigue test the coupons were sectioned through the damaged laminate near the fastener hole and damage assessed using optical microscopy. The observed damage was then mapped onto a scaled line drawing of the ply lay-up. The measurement of the length of cracks and delaminations was facilitated by the use of an X-Y micrometer stage on a WILD microscope. These damage maps gave a clear view of the damage patterns generated by the fatigue loading.

In addition to the conventional microscopy techniques, as described above, the opposite faces of the sections studied above were impregnated with a fluorescent dye and photographed under ultra-violet light. This produced good resolution of the morphology of the damage.

3. RESULTS

The results of the present study of fatigue behaviour of mechanically fastened joints in thick graphite/epoxy laminates are presented in this section. Table 2 summarises the different studies undertaken on each coupon.

3.1. Load/Displacement Hysteresis

Load/displacement hysteresis curves obtained from coupons 04-03, 06-03 and 06-02, at various stages of fatigue testing are shown in Figures 9, 10 and 11 respectively. A measure of the changes in load/displacement hysteresis is the area enclosed within the loop which is itself a measure of the energy absorbed by the specimen. Changes in hysteresis against number of loading blocks are plotted Figures 12, 13 and 14.

The hysteresis loops, shown in Figures 9, 10 and 11, show subtle changes in their characteristics with progressive fatigue loading of the specimens. Narrow loops were obtained prior to the commencement of fatigue loading and wider loops were established early in the fatigue programs. The energy absorbed in the specimen was found to increase rapidly, approaching a steady state condition, as shown in Figures 12 and 13. In the case of coupon 06-03, for example, there was a 64% change in hysteresis after the first 5 blocks of MACSEQ. These early initial changes indicated rapid changes in damage level at the holes due to the movement of the fastener. As was expected, hole shape changes were found to occur mainly in the direction of principal loading and were due to the erosion of material and damage development which occurred predominantly in this orientation, see section 3.3. Once a stable level of hysteresis was established (which varied somewhat from specimen to specimen) this then tended to decrease slowly with additional fatigue loading until (in these cases) fastener failures occurred. This behaviour accords with earlier work by Saunders et al [9] in which damage at fastener holes was studied using scanning electron microscopy. The work showed that eroded material (ie crushed carbon fibres and matrix debris), was compacted onto the hole bore thus reducing to some extent the effects of erosion of material, and may account for the establishment of an apparently stable damage state. Indeed fasteners were often found to be jammed tightly into the hole after prolonged fatigue loading. This view of establishment of a stable damage condition is confirmed by the behaviour hysteresis variation of coupon 06-02 (tested under load spectrum SQ1055). As shown in Figure 13 two sharp changes in the level of hysteresis occurred at blocks 57 and 85 where fastener replacement occurred, see Table 1 and [9]. Failure of fastener A occurred at 57 blocks and at 85 blocks both fasteners A and B were replaced. It appears that removal of the fasteners, by force, also removed welded debris, and thus provided a larger hole for the new fastener and exposing new matrix and fibres to erosion. A new damage level then had to be established by further erosion of matrix and fibres.

The hysteresis behaviour of specimens fatigued under MACSEQ with a peak load of 36 kN showed similar initial trends but a full stabilisation of hysteresis was not realised because of fastener failures, as seen in Figure 14. Coupon 09-01, from a defective panel, contained a large initial delamination but the coupon displayed less hysteresis change than coupon 07-12 which had no discernible incipient damage. Both coupons had identical initial hole sizes and were tested under identical conditions and so the disparity in hysteresis change remains unexplained. Both coupons lasted 7 full blocks of MACSEQ, and this highlighted the magnitude of scatter in the hysteresis data. These data then could only be used to show general trends in the fatigue life behaviours of the mechanically fastened composite joints.

3.2. Shadow Moiré Fringe Study

Moire fringe photographs, showing fastener tilt, taken of coupons 07-11, 07-12, 09-01 and 06-04 at various intervals during testing are presented in Figures 15 to 18 respectively.

All coupons studied here were tested under the higher MACSEQ spectrum loads (peak load of 36 kN). As expected, fastener tilt increased with level of load applied. More importantly, however, the degree of fastener tilt, for a given applied static load, was found to increase with the number of fatigue loading blocks. In all of the cases, the expected trend of increased fastener tilt with fatigue, was indicated by the increase in the number of fringes observed for a given static load level. Since each fringe represented 0.08 to 0.15 mm of out-of-plane movement an estimate of fastener tilt could be made, see Figure 19, and the trends for coupons 09-01, 07-12 and 06-04 are plotted in Figures 20 to 22, respectively. These plots can only be treated as approximate because of the inherent inaccuracies in the measurements. However, it can be seen that permanent deformation (seen as fastener tilt) was established in the fatigue loading program. This may be associated with the establishment of the stable damage state, but of course could arise due to plastic deformation of the fastener. It was not possible to separate the two contributions to the permanent deformation.

The trend towards a stable damage state with increasing number of loading blocks was not realised here because of the use of the MACSEQ spectrum at the higher load level (36 kN peak load) which accelerated the failures (of fasteners). This can be seen in Figure 14, for coupon 07-12, where stability was not fully achieved due to fastener failure.

3.3. Damage Mapping

Coupons 04-01, 04-03 and 06-05 were sectioned at the completion of testing. Damage maps of the regions studied were drawn to scale and are presented in Figures 23 to 25, respectively, where individual section maps are also provided. These coupons were tested under the lower MACSEQ spectrum loads and results of the scanning electron and optical microscopy were reported in [9]. The damage conditions of the hole surfaces developed under fatigue loading are reproduced from reference [9] in Appendix 1. Here Figures A.1 to A.4 show the fastener hole surfaces and profiles for coupons 04-02, 04-01, 04-03 and 06-05 at loading blocks 5, 15, 45 and 90 respectively.

Coupons 06-02, 06-08, 06-09, 06-03, 06-10, 07-08, 04-06 and 06-04 were C-scanned at the termination of testing and all coupons were then sectioned and damage maps (drawn to scale) constructed of the faces studied. The opposite faces to those mapped, were impregnated with fluorescent dye and photographed under ultra-violet light, in order to highlight the damage. The result of the three methods of examination are presented for each of the coupons in Figures 26 to 33, respectively. Coupons 06-02, 06-08, 06-09 and 06-03 were tested under the lower spectrum loads, and the remaining four coupons were tested under the higher loads (see Tables 1 and 2). For the two levels of spectrum loads, the coupons are listed in order of number of loading blocks undergone prior to the termination of test or failure. In addition, coupons 07-12 and 09-01 were also scanned at various stages during testing in an attempt to monitor progressive damage development.

Coupon 09-01 was found to contain a large delaminated region prior to testing. This was believed to have been caused by incorrect manufacture of the composite panel. C-scan maps of coupons 07-12 and 09-01 are given in Figures 34 and 35 respectively, where the progressive damage behaviour can be observed for the former.

From the results reported in [9] and the visualisation of damage morphology in the present work it was possible to gain an over-all view of the damage generation process at the fastener holes. As the number of fatigue loading blocks increased, the level of damage sustained by the composite laminate at the fastener holes increased. At both load levels the form of damage was similar. The form of damage involved erosion of matrix material [9] and the development and growth of delaminations. Generally, extensive delamination growth was observed with increasing numbers of loading blocks, although the were some anomalous behaviours, for example coupons 06-03 (30 kN peak load) and 06-04 (36 kN peak load). The reasons for these anomalous behaviours were undetermined.

During prolonged fatigue testing the 0° plies exhibited increased erosion of matrix material [9] and the damage maps showed that cracking occurred deeper into the laminate. The nucleation site(s) for early delamination development appeared to be at the 45/0° interfaces, but with extensive fatigue loading and associated generation of extensive damage, delaminations were also nucleated at the +45/-45° and 0/0° interfaces. As fatigue loading continued the density of intra-ply cracking also increased. Damage mapping showed progressive (intra-ply) cracking of the 0° plies and induced cracking in the $\pm$45° ply layers.

It was observed that, for the test configuration studied, one hole (normally hole B) sustained slightly greater damage than the other. The degree of damage sustained in either hole is influenced by the relative magnitudes of the loads transferred through the holes and it was considered that this, in turn, was influenced by factors such as coupon/fixture compliance, hole geometry and fastener fit. The effects of initial fastener hole size on laminate fatigue behaviour was difficult to assess. There was no obvious relation between initial hole sizes and fatigue life of the fasteners or the extent and type of damage developed at the fastener hole. The effects of hole size may have been overwhelmed by the scatter in the fatigue behaviour arising from (localised) laminate variability. As fatigue loading proceeded, the volume of damage at the fastener hole increased as shown in the damage maps in Figures 27 and 28 (30 kN peak load) and in Figures 30 to 32 (36 kN peak load). The form of damage appeared to be a progressive crushing failure, and this produced an increase in the volume of damage at the fastener hole. The most important aspect of this damage development was its characteristic shape. It was found that as the crushing failure progressed into the material it was accompanied by the nucleation of additional delaminations, particularly under the fastener head, and the growth of existing delaminations. The progressive crushing of the laminate has been observed in similar experiments on thin graphite/epoxy laminates under high bearing loads [16,17] and explains the generation of elongated holes reported by Poon and Gould [18].

4. DISCUSSION

The present study of the fatigue behaviour of mechanically fastened joints in thick graphite/epoxy laminates has shown that, at current aircraft laminate strain levels, the joints exhibit excellent fatigue resistance properties. In the present work and also that reported earlier by Saunders et al [9] it was found that the most significant "structural failure" was the fatigue fracture of the titanium fasteners. These failures were not considered to represent a major structural problem in aircraft for the conditions investigated in the present work. Other failure modes, investigated in the present work were fastener hole wear and the nucleation and growth of delaminations around fastener holes. Both failure modes were considered to represent significant degradation of the material and, at higher strain levels, would be considered a potential structural problem in aircraft if, as a result of the damage development, load paths and/or significant stiffness changes occurred

The wearout of fastener holes was studied by scanning microscopy by Saunders at el [9]. Most importantly this work showed that the erosion of matrix material from between ply layers nucleated delaminations which then grew into the laminate. The progressive development of damage states was also studied by indirect measurements such as load/displacement hysteresis and fastener movement. Both of these indirect methods showed that damage (erosion and delamination nucleation and growth) developed rapidly and, in particular, the load/displacement hysteresis measurement showed the tendency of the specimen to establish a stable damage state from which further degradation slowly generated. The establishment of stable or characteristic damage states early in the fatigue life of a mechanically fastened composite laminate is consistent with the reported degradation behaviours of plain, notched and impact-damaged composite laminates [19-21].

While the experimental work demonstrated that it was possible to "visualise" the establishment of stable damage states it was not possible to determine the contribution of the various damage and deformation conditions to the overall measured changes. For example fastener movement can be the result of damage to the laminate (clearly visible by microscopy), but it can also arise from plastic deformation of the fastener and elastic deformation giving rise to fastener rotation along its major axis. It was not possible to ascribe the relative contributions to measured fastener movement of the possible modes of damage and deformation.

The growth of damage into the laminate was studied in some detail by damage mapping which has led to a general understanding of the damage growth process in thick laminates. Once nucleation had occurred (by matrix erosion) the delaminations progressed into the laminate under the action of the compressive loads. The damage mapping showed that, as a general rule, the growth of delaminations in the centre of the laminate was preceded by the generation of cracks in the $\pm 45°$ ply layers. Such a situation was similar to the fatigue process reported by Reifsnider and Jamison [19], Highsmith et al [20] and Clark and Saunders [22] in which delaminations were nucleated at the cross-over point between cracks in adjacent non-parallel plies. Growth of delaminations near the faying surface appeared to involve less cracking in the $\pm 45°$ ply layers. The damage maps did not reveal any pattern of dominant delamination formation; degradation was an on-going process with new delaminations (and associated cracking) continually being nucleated throughout the fatigue loading program of a specimen.

Significantly, there was usually a delamination which formed under the fastener head after a significant number of blocks of loading and it was observed that this delamination grew significantly throughout the remaining life of the specimen. The nucleation of this delamination could be the result of excessive fastener movement once the stable damage configuration had been established.

5. CONCLUSIONS

[1] Optical microscopy of sections through the laminate surrounding fastener holes has allowed the visualisation of the morphology of damage around the fastener hole.

[2] Indirect measurement of damage development such as measurements of load/displacement hysteresis and fastener tilt have shown that a stable damage state developed rapidly in a fatigue loading program applied to a thick composite laminate-metal mechanically fastened joint.

[3] Damage development and growth was found to be a continuing process with fatigue loading. Although the initial changes, leading to the establishment of a stable damage state, were rapid, additional degradation occurred slowly with continued fatigue loading. Degradation was accelerated by a spectrum with higher peak loads.

[4] The process of damage growth has been shown to occur by delamination growth preceded by intra-ply cracking in the $\pm 45°$ ply layers. Intra-ply cracking was more pronounced in the central regions of the laminate.

[5] The removal of fasteners appeared to induce an acceleration of damage development because of the removal of debris, associated with the stable damage state, which led to the re-establishment of a new damage state of a higher damage density.

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Table 1: Applied peak tensile load, number of loading blocks and initial hole diameters for the composite bolted joint coupons tested.

Coupon Number[1]	Peak Load (kN)	No. of Blocks	Average Hole Diameter[2] (mm)		SEM/OM[3]
AS4-3501/6			A	B	
06-02[4]	30	103[6]	6.54[5]	6.54[5]	
06-08	30	1	6.46	6.42	-/y
04-02	30	5	6.36	6.37	y/y
04-01	30	15	6.37	6.37	y/-
04-03	30	45	6.36	6.35	y/y
04-04	30	60	6.35	6.35	y/-
06-09	30	45	6.40	6.42	y/y
04-05	30	90	-	-	
06-05	30	90[7]	6.38	6.37	y/y
06-07	30	147[8]	6.37	6.37	
06-03	30	165	6.36	6.38	-/y
06-10	36	1	6.42	6.41	-/y
07-11a	36	4	6.34	6.34	
07-08	36	5	6.35	6.33	-/y
04-06	36	6	6.38	6.34	-/y
07-09b	36	6	6.35	6.34	
07-12	36	7	6.33	6.33	
06-04	36	15	-	-	-/y
09-01	36	8	6.33	6.33	
04-07	36	10	6.33	6.33	

1 Coupon numbers in bold indicate tests were terminated before fasteners failed.
2 Measurement using microscope.
3 Coupon sectioned; SEM-bore surfaces examined under scanning electron microscope, OM-specimen cross-sections examined under optical microscope.
4 Block loading used was SQ1055, an alternative aircraft loading sequence which may be more severe than MACSEQ [23].
5 Measured at 50 blocks.
6 Fasteners removed and hole diameter measured at block 57, 85 & 103; new fasteners installed at block 57 & 85 (fastener failed at block 57).
7 Fasteners removed and hole diameter measured at block 30, 40, 70 & 90; new fasteners installed at block 30.
8 Fasteners removed and hole diameter measured at block 107 & 147; new fasteners installed at block 107 (fastener failed at block 107).

Table 2: Table of various studies undertaken on various composite bolted joint coupons.

Coupon Number	Various Studies Undertaken				
	Load/Displ. Hysteresis	Moire Fringe Photography	Ultrasonic C-Scanning	Damage Mapping	Dye Penetrant Photography
06-02+	✔		✔	✔	✔
06-08@			✔	✔	✔
04-01@				✔	
04-03@	✔			✔	
06-09@			✔	✔	✔
06-05@				✔	
06-07@	✔				
06-03@	✔		✔	✔	✔
06-10#			✔	✔	✔
07-11#	✔	✔			
07-08#			✔	✔	✔
04-06#			✔	✔	✔
07-12#	✔	✔	✔		
09-01#	✔	✔	✔		
06-04#		✔	✔	✔	✔

+ Tested under load spectrum SQ1055,: peak load 30kN.
@ Tested under load spectrum MACSEQ: peak load 30kN.
# Tested under load spectrum MACSEQ: peak load 36kN.

APPENDIX 1.

This Appendix reproduces four figures from reference [9] which are of particular relevance to this Report. The following figures depict the damage developed around fastener holes in a 50 ply AS4/3501-6 composite laminate. The lay-up was:

$$[(+45_2,-45_2,0_4)_3,90]_S$$

which was the same as that used in this Report.

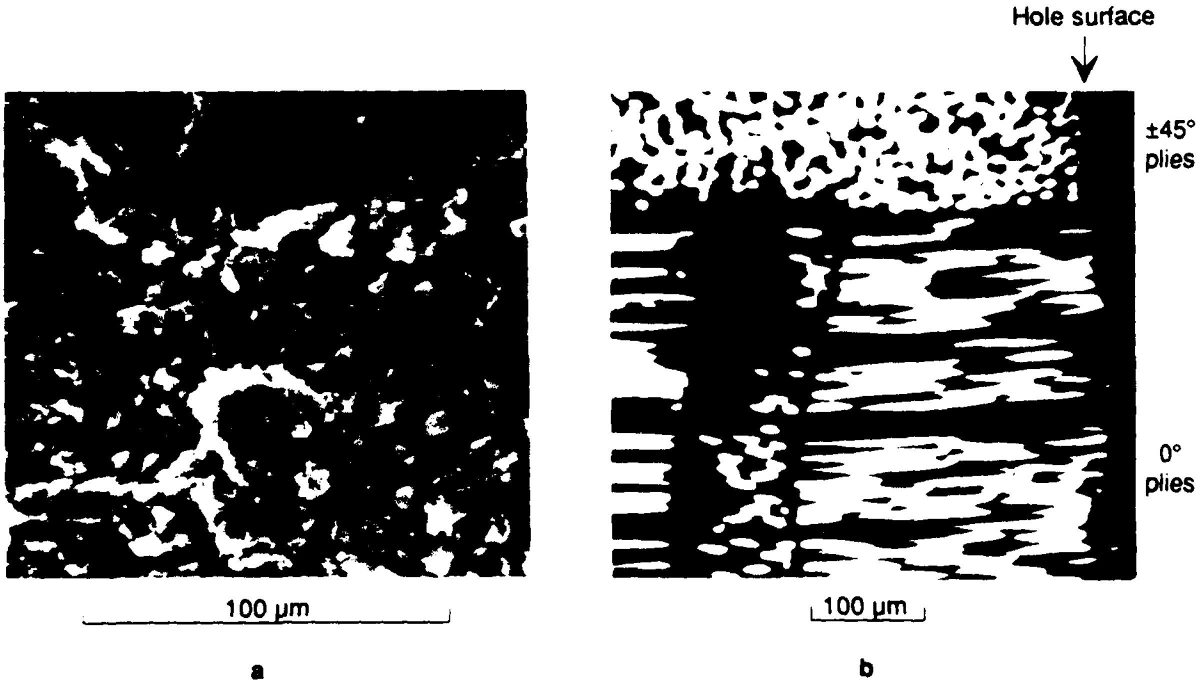

Figure A.1 Fastener hole B from coupon 04-02, fatigued to 5 blocks MACSEQ (30 kN peak load).
a) Scanning electron micrograph showing accumulated debris and minor erosion of the 0° ply layers.
b) Optical micrograph showing cracking in the 0° ply layers close to the fastener hole surface and minor erosion of matrix material.

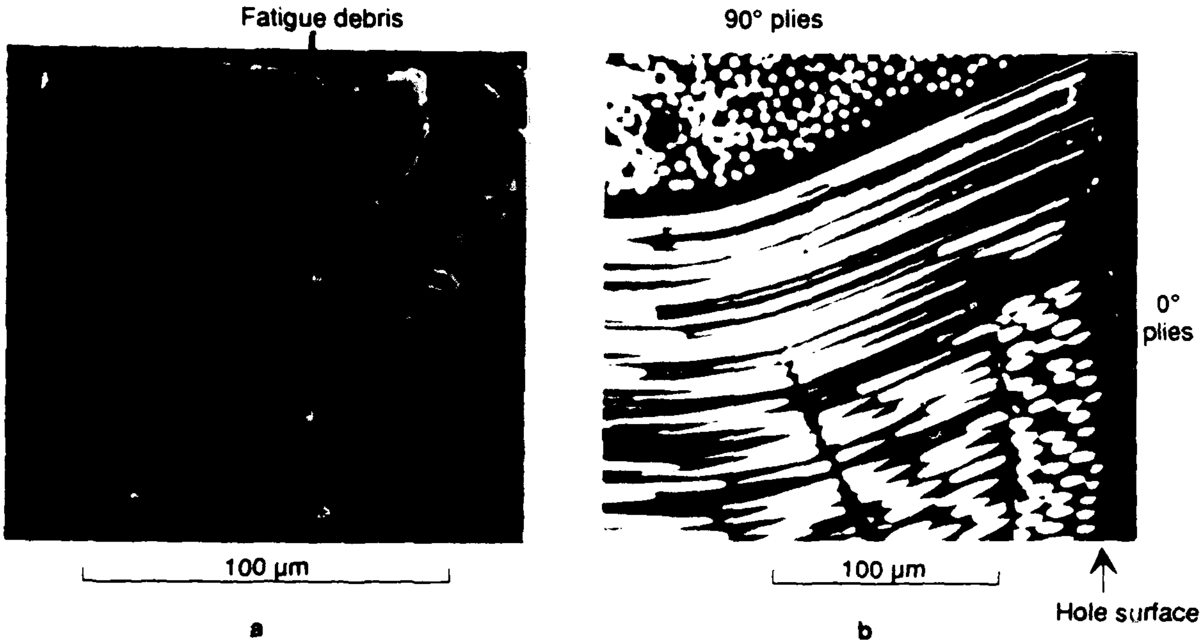

Figure A.2 Fastener hole B from coupon 04-01, fatigued to 15 blocks MACSEQ (30 kN peak load).
a) Scanning electron micrograph showing accumulated debris over a 0° ply layer.
b) Optical micrograph showing cracking in the 0° ply layers close to the fastener hole surface and minor erosion of matrix material.

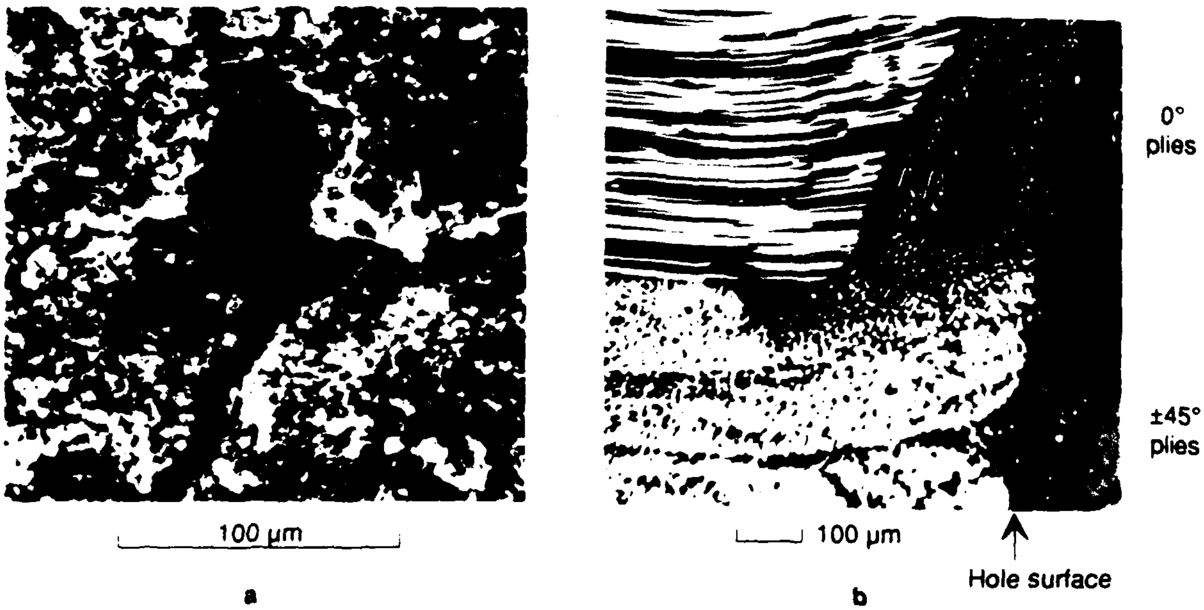

Figure A.3 Fastener hole B from coupon 04-03, fatigued to 45 blocks MACSEQ (30 kN peak load).
a) Scanning electron micrograph showing accumulated debris covering ply layers in the principal loading direction.
b) Optical micrograph showing the thick deposited layer of fatigue debris, damage to the 0° ply layer and erosion of matrix between the 0° and the ±45° ply layers.

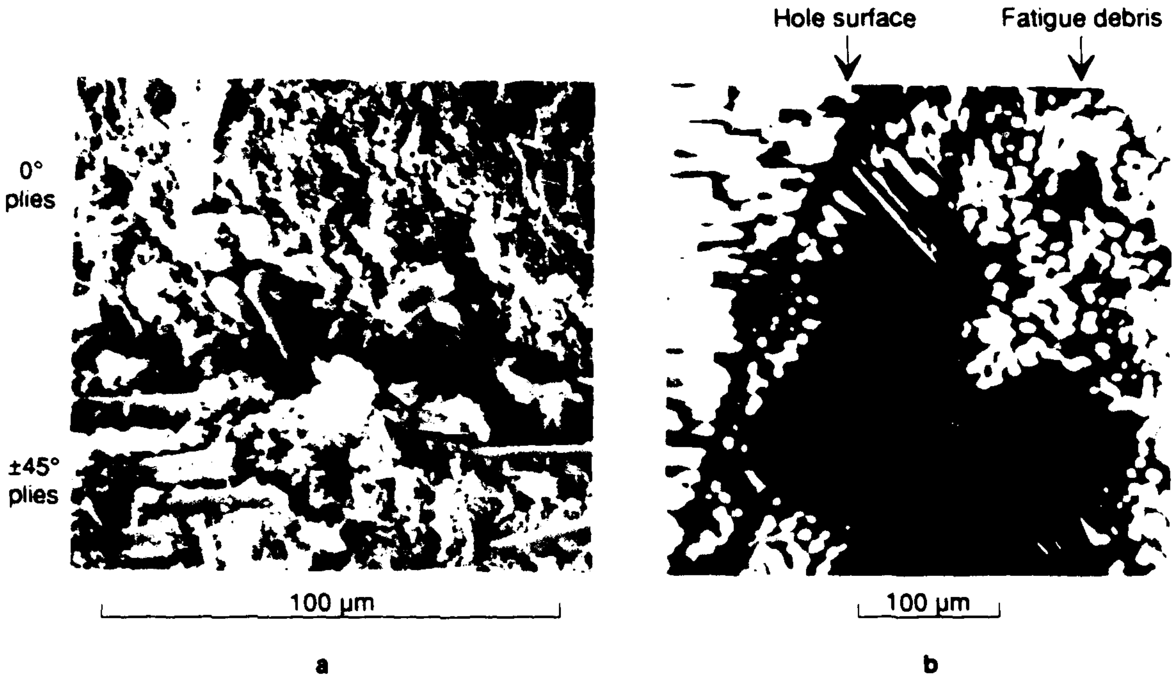

Figure A.4 Fastener hole B from coupon 06-05, fatigued to 90 blocks MACSEQ (30 kN peak load).
a) Scanning electron micrograph showing severe damage to the 0° ply layers and removal of material between a 0° and a 45° ply layer.
b) Optical micrograph showing accumulated debris of both matrix material and carbon fibres and severe 0° ply damage.

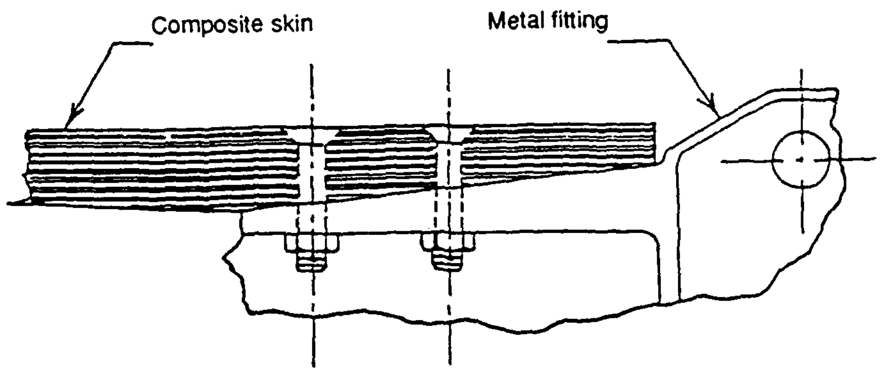

Figure 1. A typical mechanical joint in a thick wing panel.

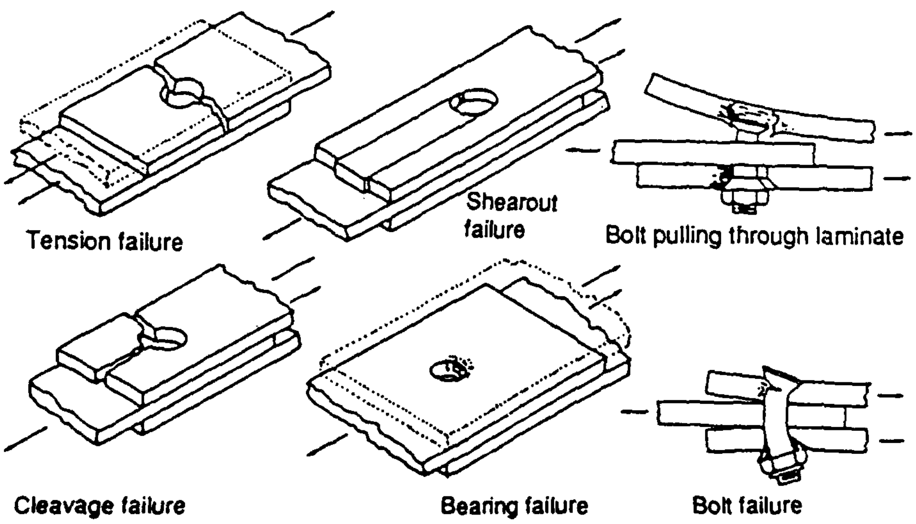

Figure 2. Typical failures of mechanically fastened joints in composite laminates [24].

Figure 3. Delaminations generated around a fastener hole during assembly.

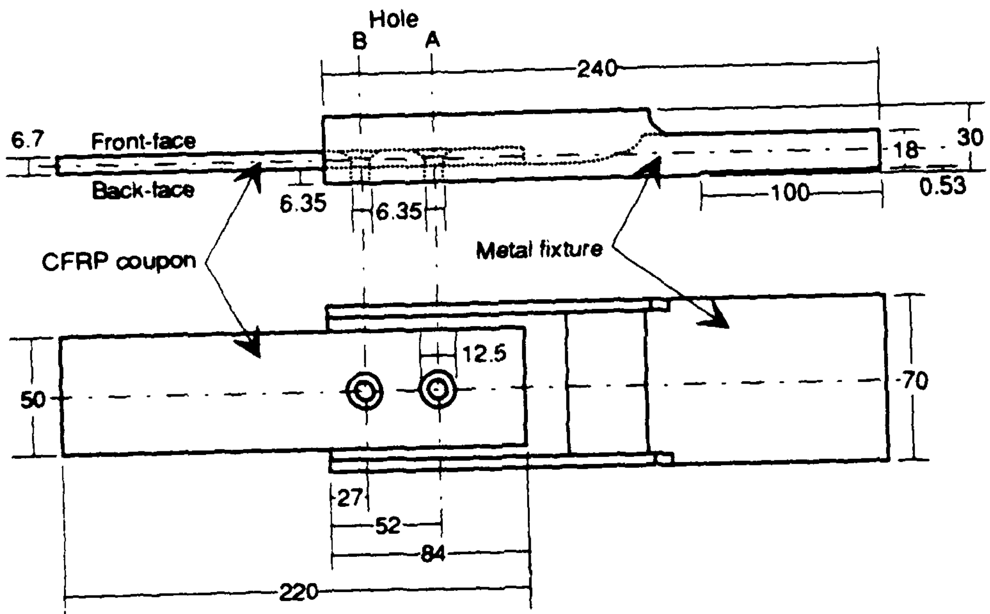

Figure 4. The composite-to-metal specimen for the fatigue testing of mechanically fastened joints.

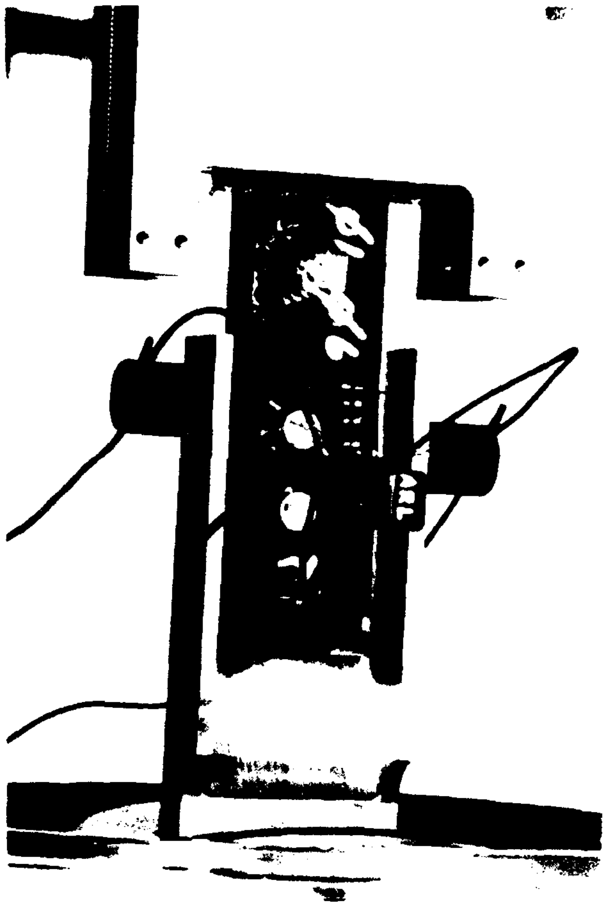

Figure 5. Extensometer set-up for load/displacement hysteresis measurement.

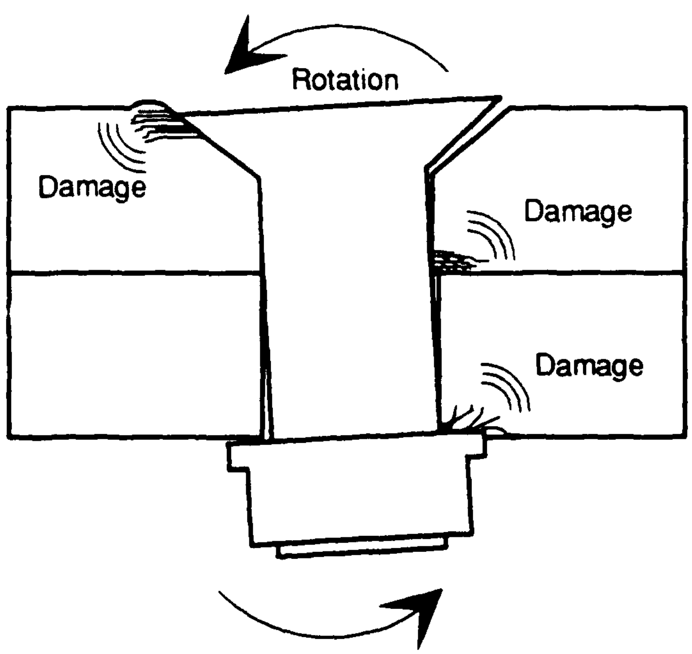

Figure 6. Fastener movement under fatigue loading.

Figure 7. Shadow moiré fringe of fastener movement.

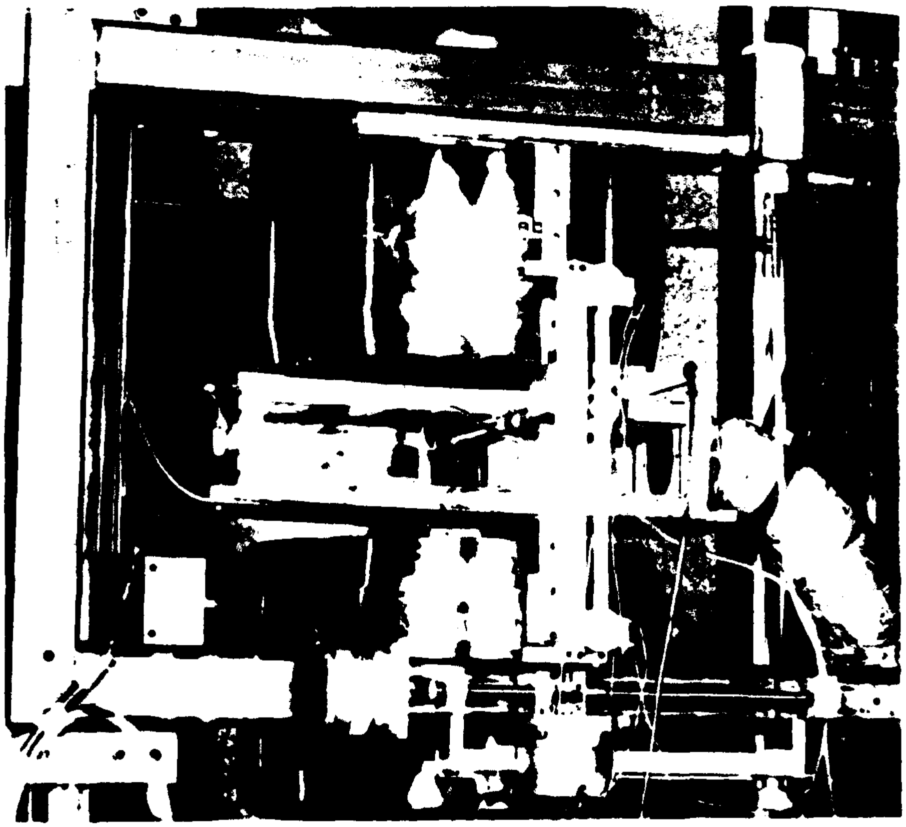

Figure 8. Experimental set-up for in-situ C-scanning of composite bolted joint coupons.

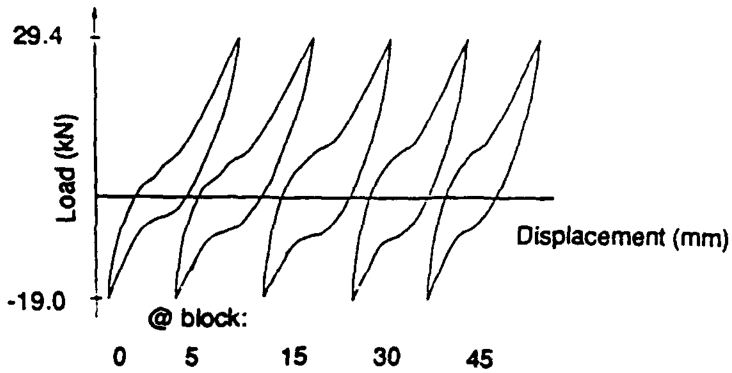

Figure 9. Load/displacement hysteresis behaviour for coupon 04-03 (30kN peak load).

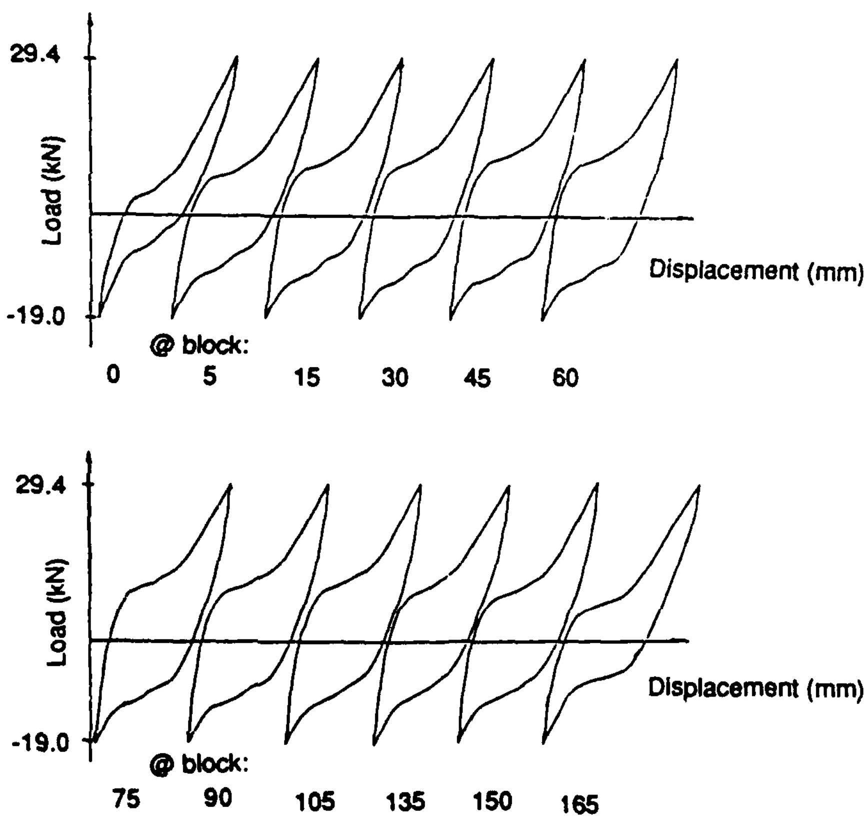

Figure 10. Load/displacement hysteresis behaviour for coupon 06-03 (30kN peak load).

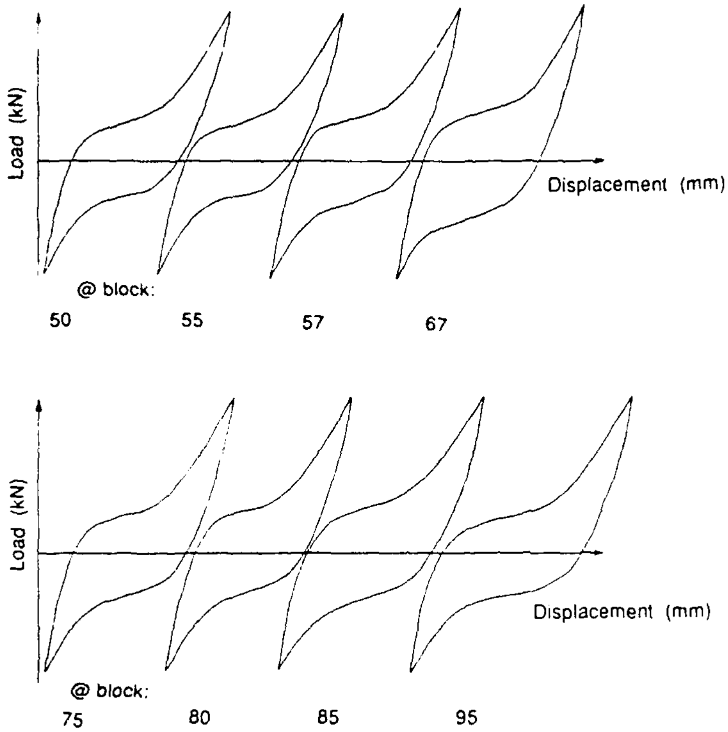

Figure 11. Load/displacement hysteresis behaviour for coupon 06-02 (30kN peak load).

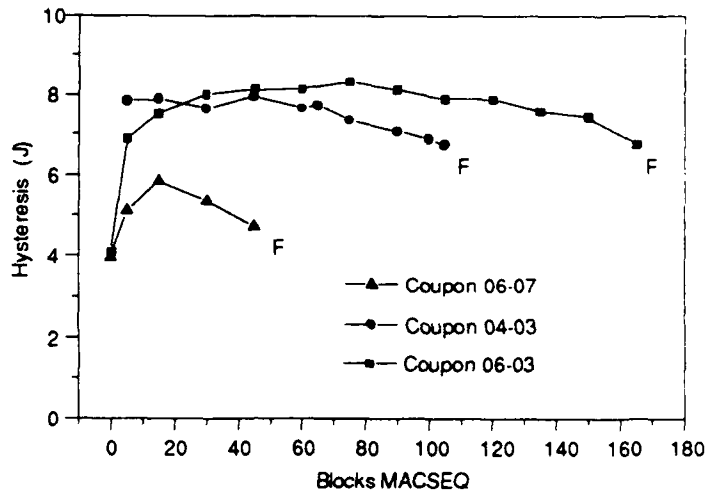

Figure 12. Variation of area under load/displacement hysteresis curves with increasing number of blocks MACSEQ (30kN) until fastener failure (F).

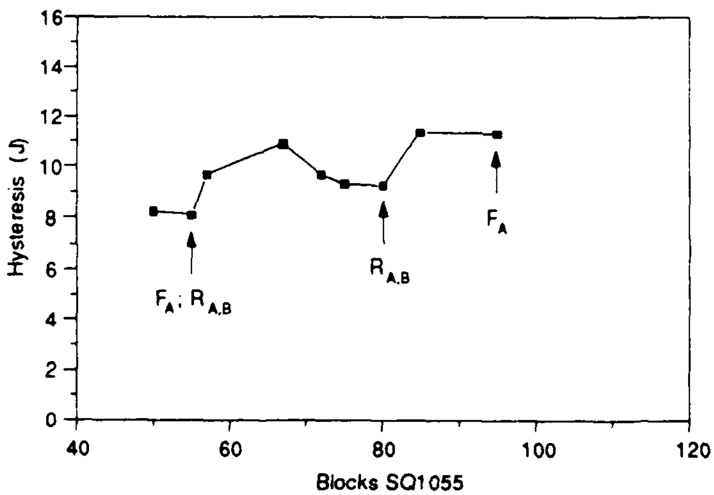

Figure 13. Variation of area under load/displacement hysteresis curves with increasing number of blocks SQ1055 (30kN) for coupon 06-02 including fastener replacement (R) and fastener failure (F) for fasteners A and B.

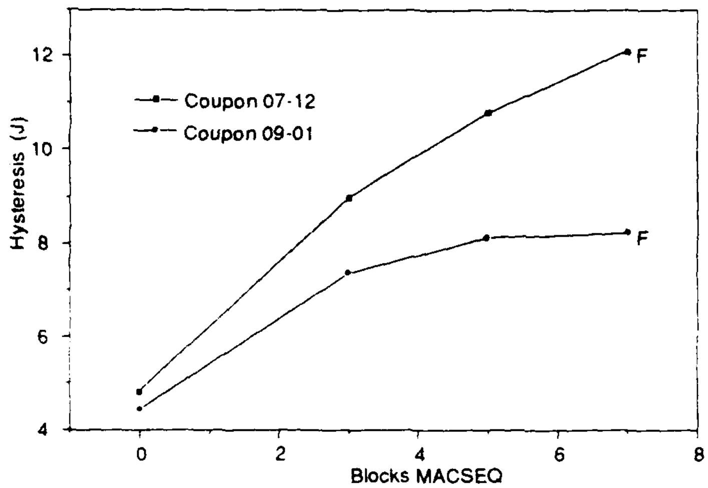

Figure 14. Variation of area under load/displacement hysteresis curves with increasing number of blocks MACSEQ (36kN) until fastener failure (F).

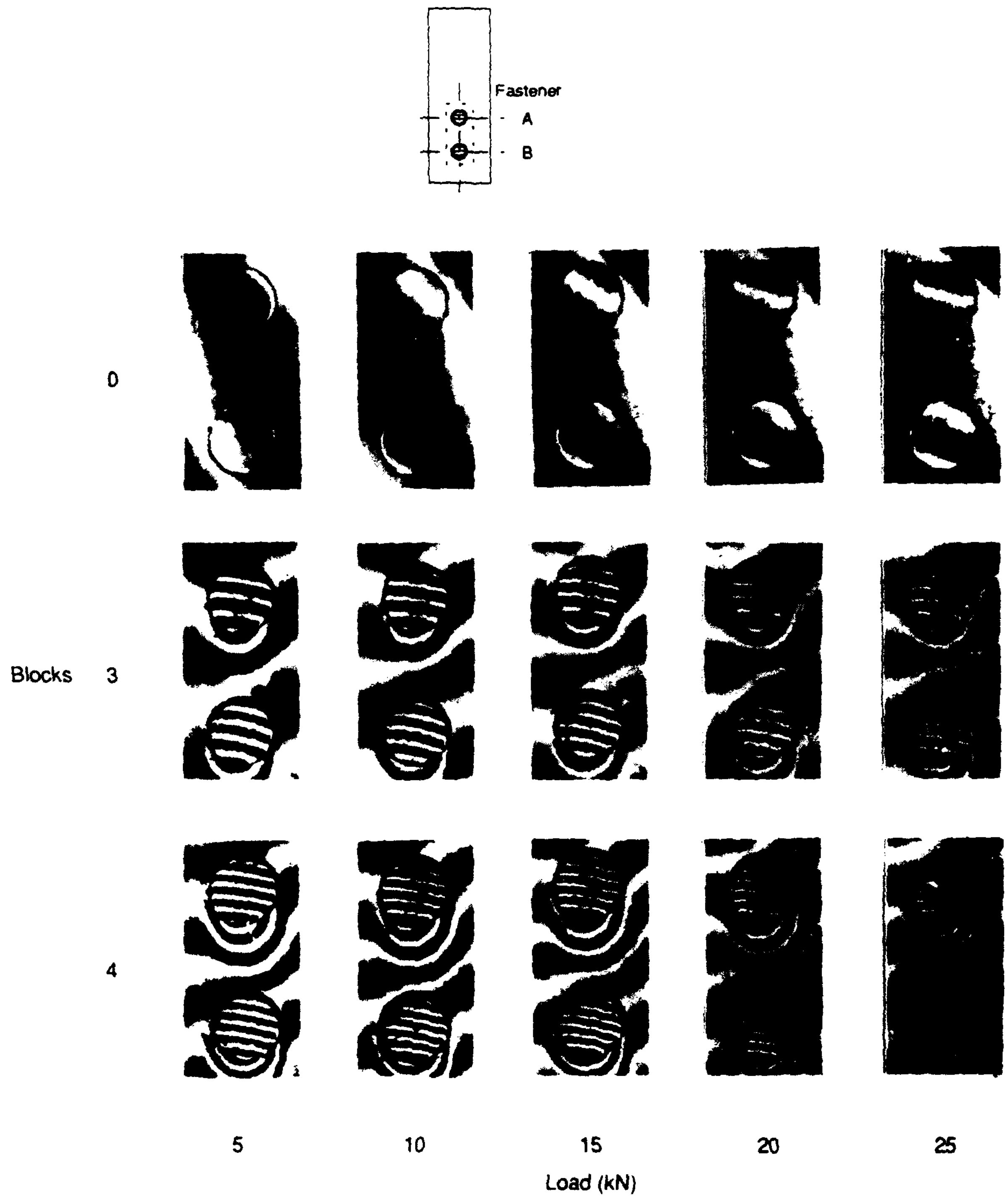

Figure 15. Shadow moiré fringes developed by loading coupon 07-11 (36kN peak load).

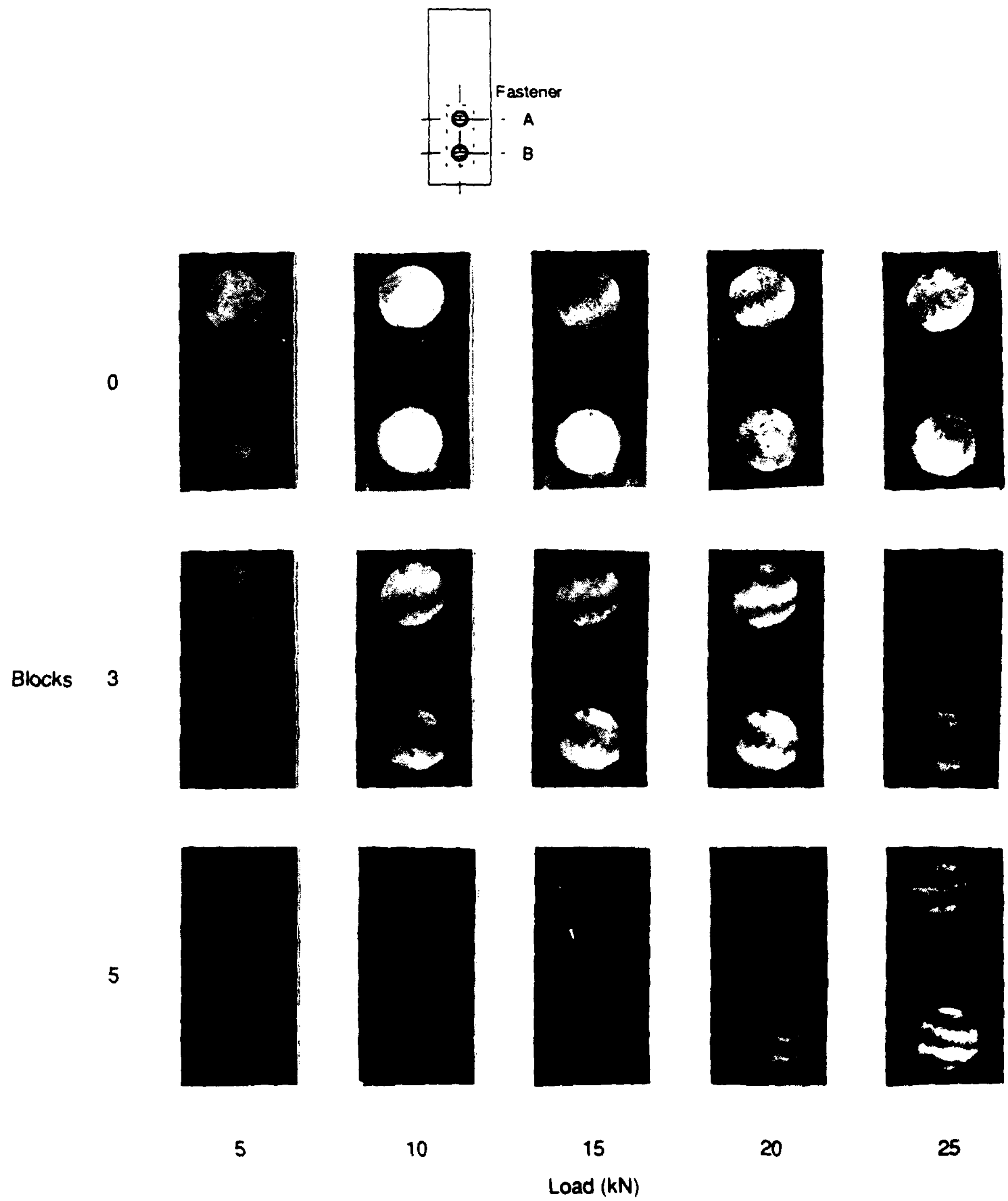

Figure 16. Shadow moiré fringes developed by loading coupon 07-12 (36kN peak load).

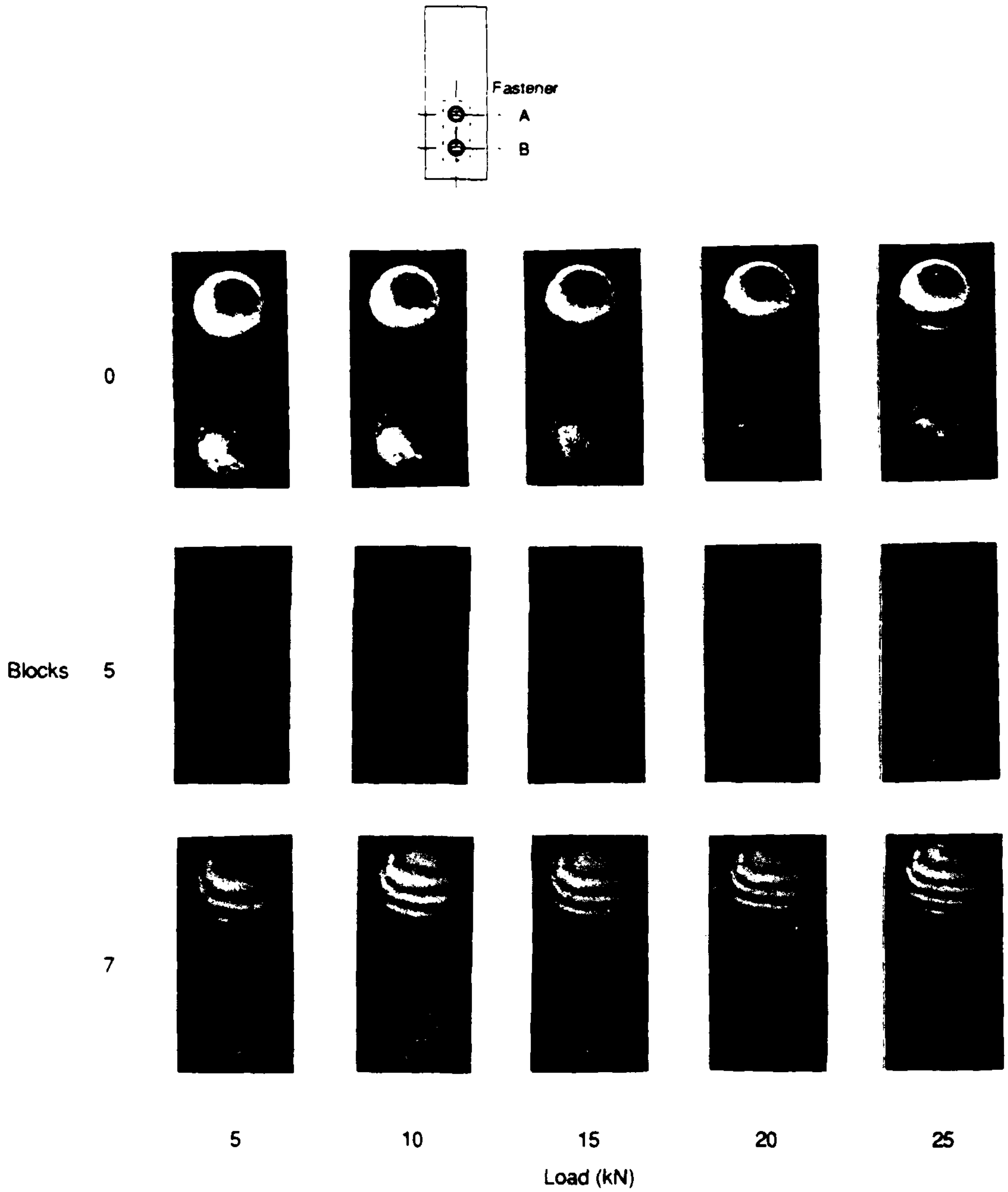

Figure 17. Shadow moiré fringes developed by loading coupon 09-01 (36kN peak load).

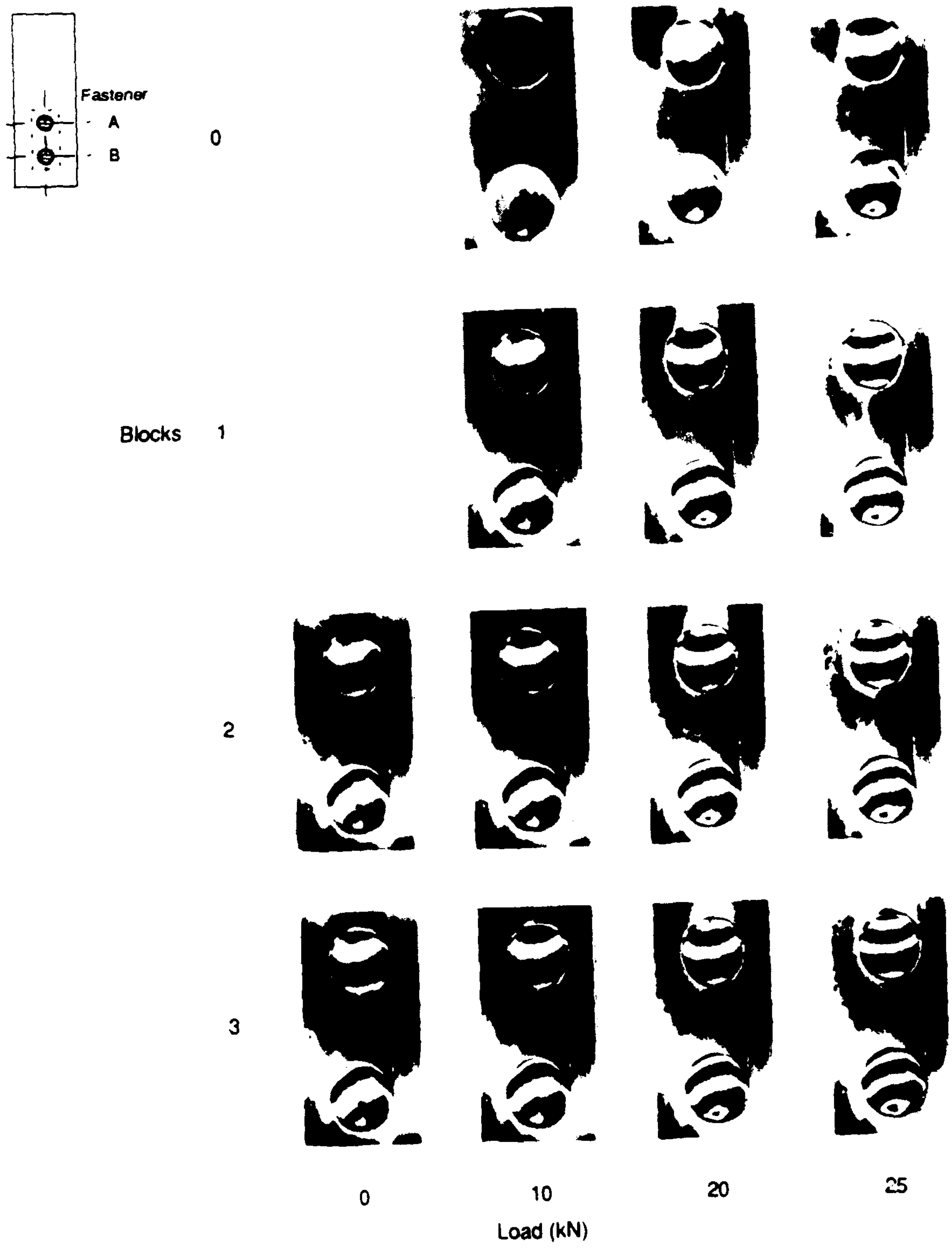

Figure 18. Shadow moiré fringes developed by loading coupon 06-04 (36kN peak load).

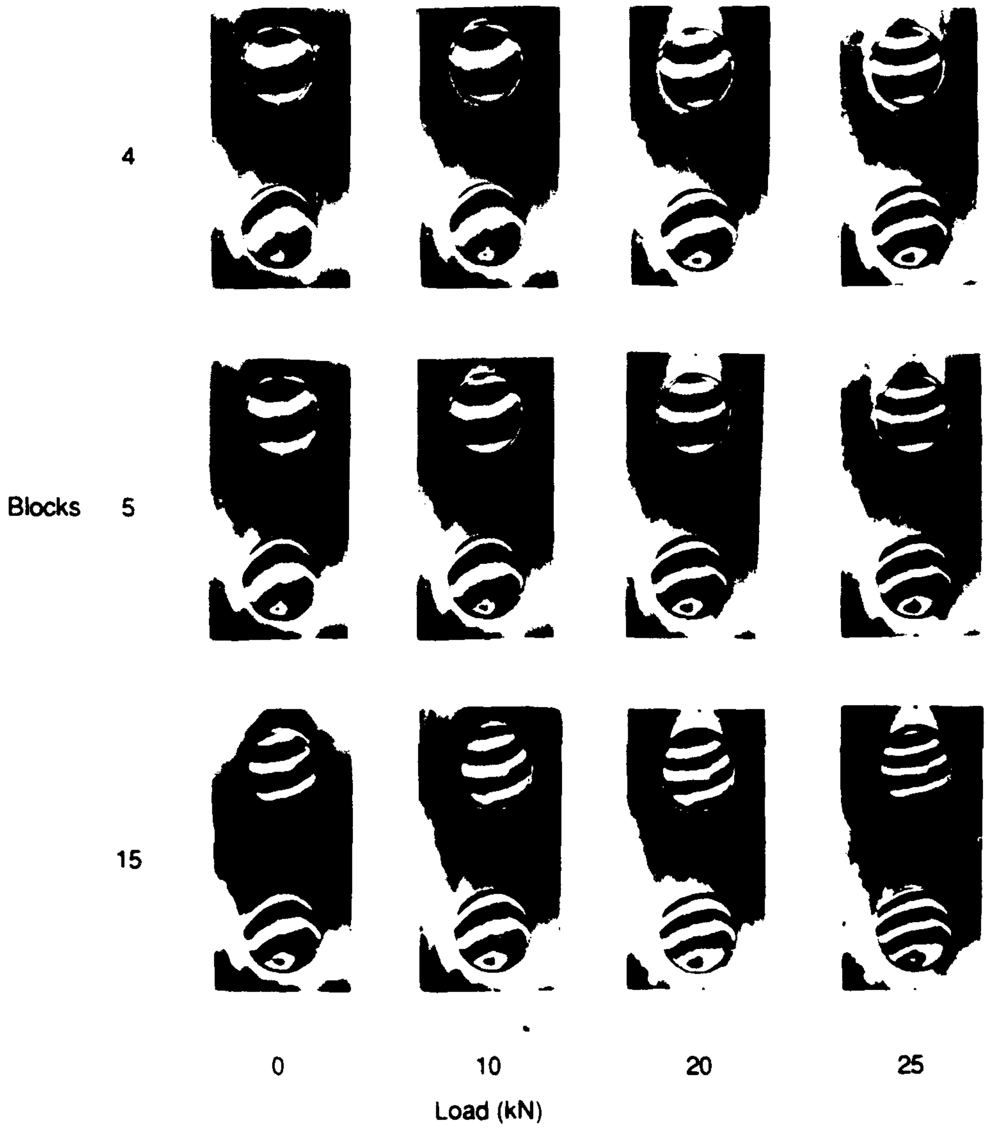

Figure 18. Cont.

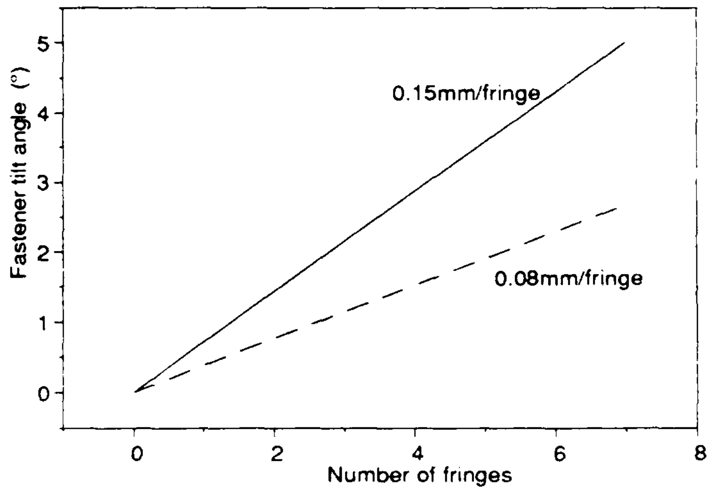

Figure 19. Fastener tilt angle versus number of shadow moiré fringes.

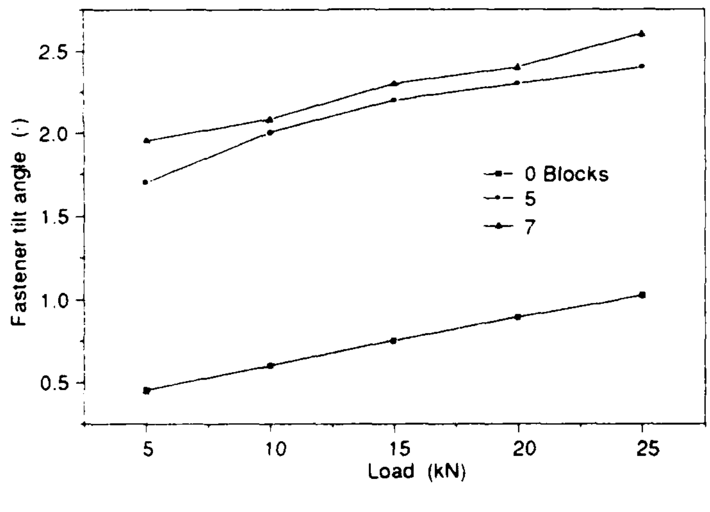

Figure 20. Variation of fastener tilt angle with increasing static load for coupon 09-01 (36kN peak load).

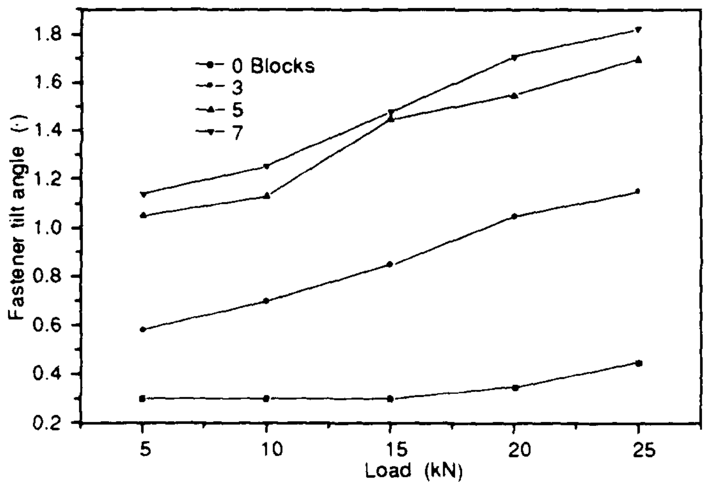

Figure 21. Variation of fastener tilt angle with increasing static load for coupon 07-12 (36kN peak load).

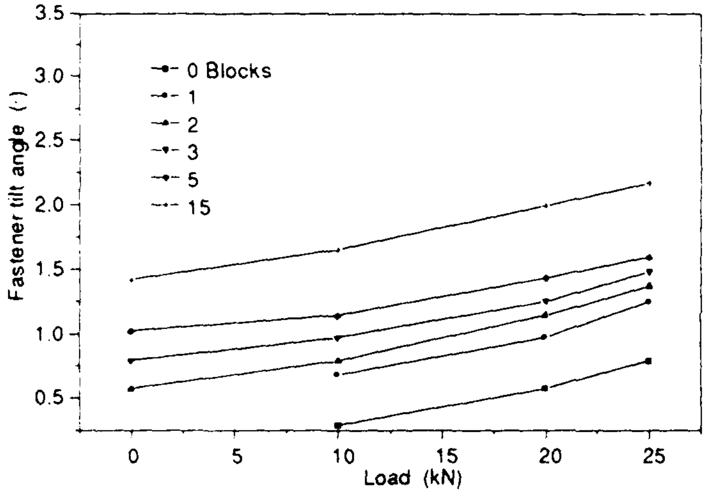

Figure 22. Variation of fastener tilt angle with increasing static load for coupon 06-04 (36kN peak load).

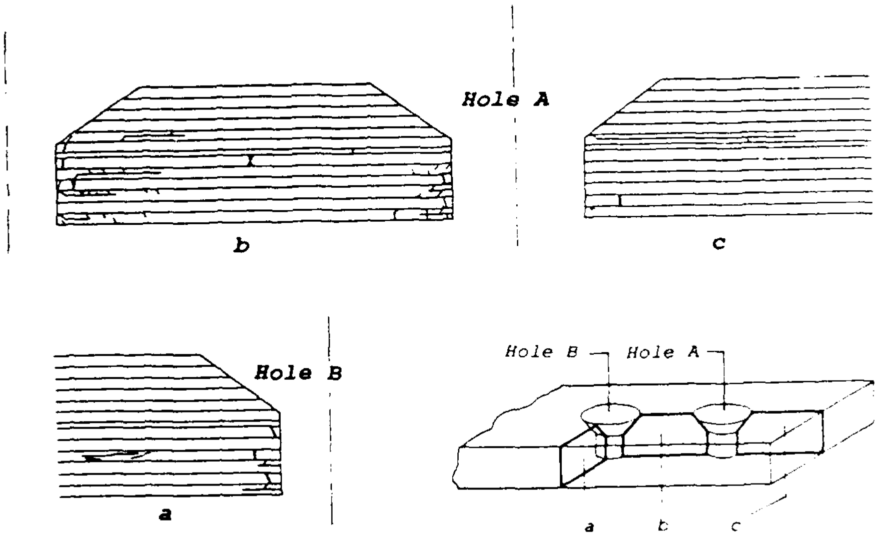

Figure 23. Damage maps from coupon 06-05, fatigued to 90 blocks MACSEQ (30 kN peak load).

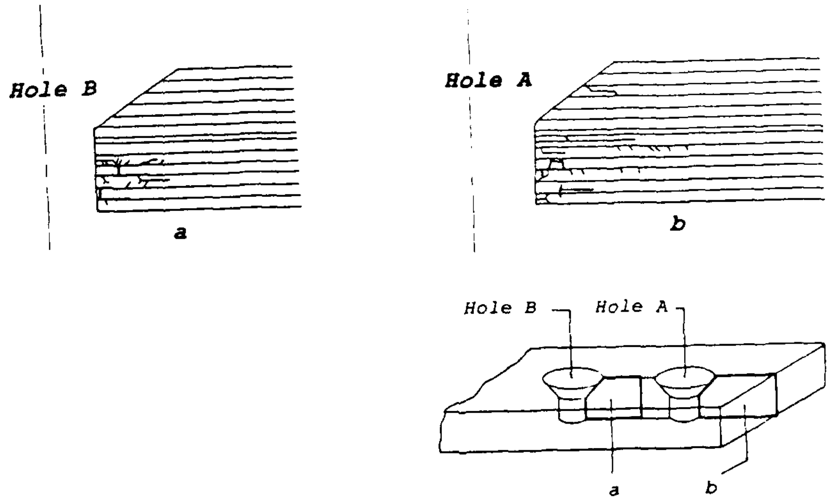

Figure 24. Damage maps from coupon 04-03, fatigued to 45 blocks MACSEQ (30 kN peak load).

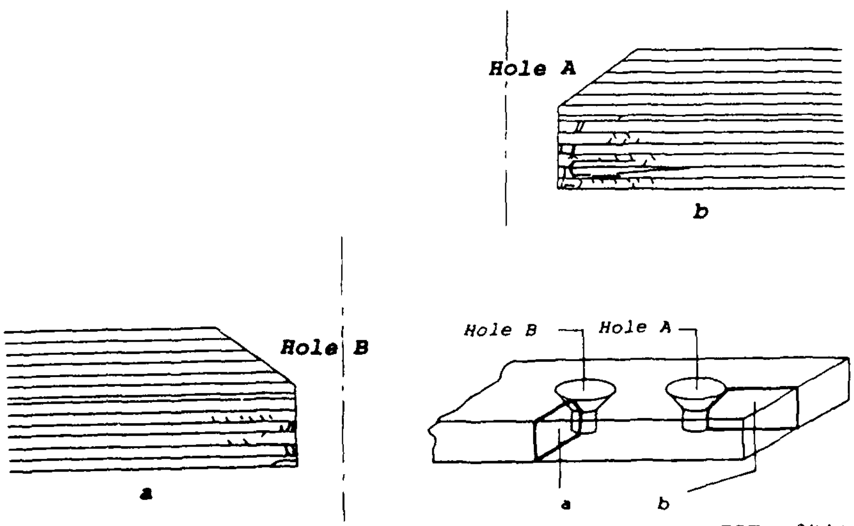

Figure 25. Damage maps from coupon 04-01, fatigued to 15 blocks MACSEQ (30 kN peak load).

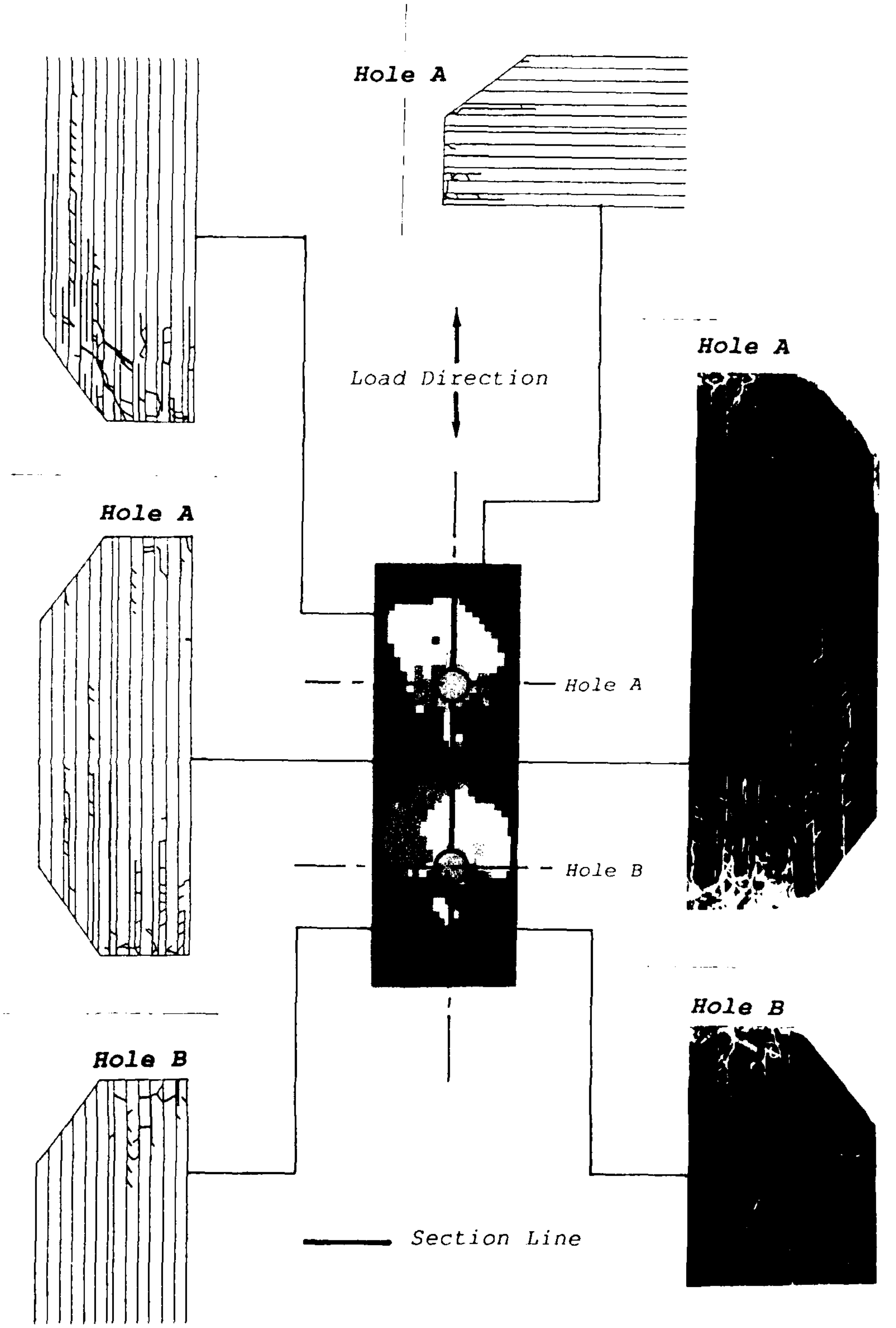

Figure 26. Damage maps from coupon 06-02, fatigued to 103 blocks SQ1055 (30 kN peak load).

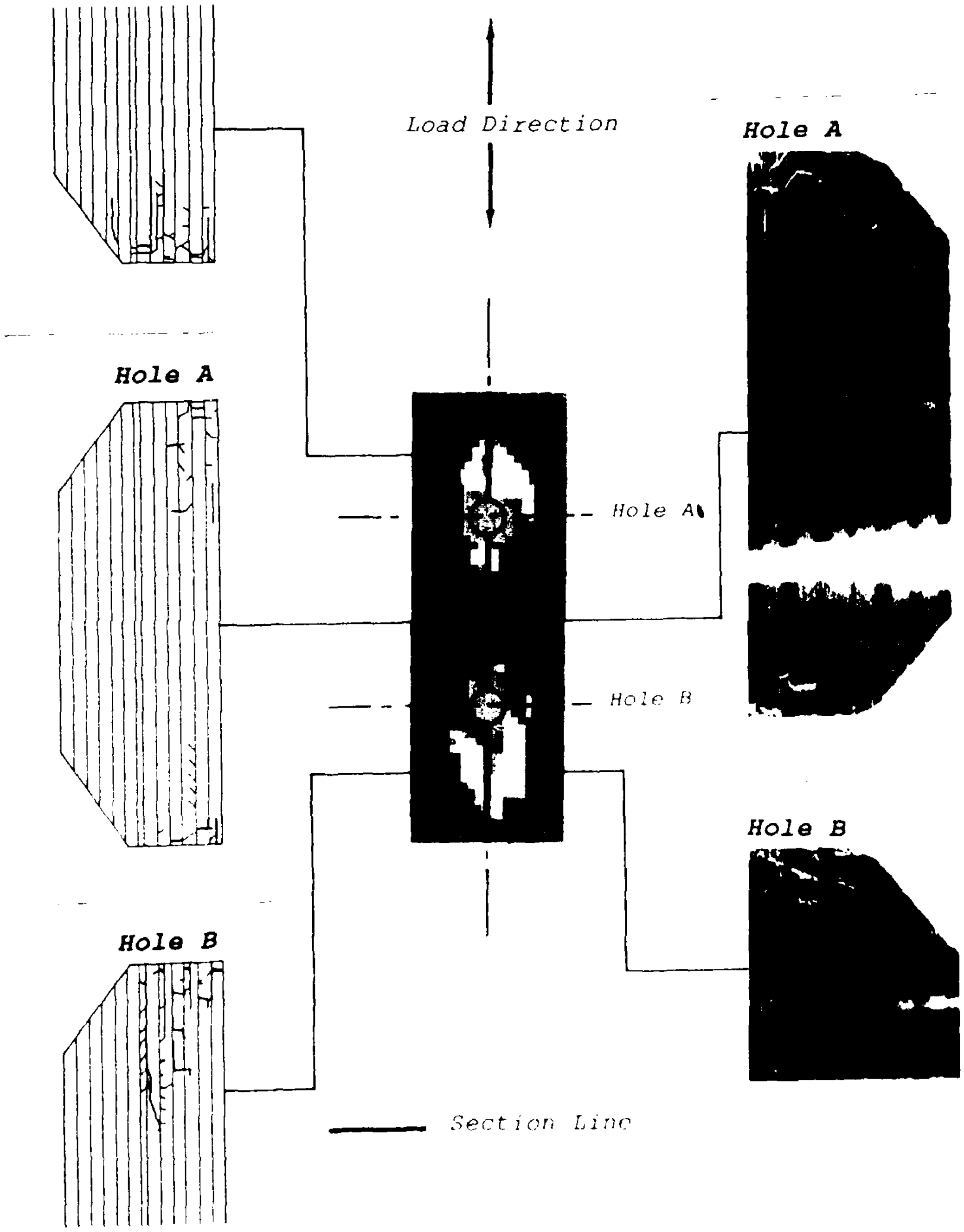

Figure 27. Damage maps from coupon 06 08, fatigued to 1 block MACSFQ (30 kN peak load)

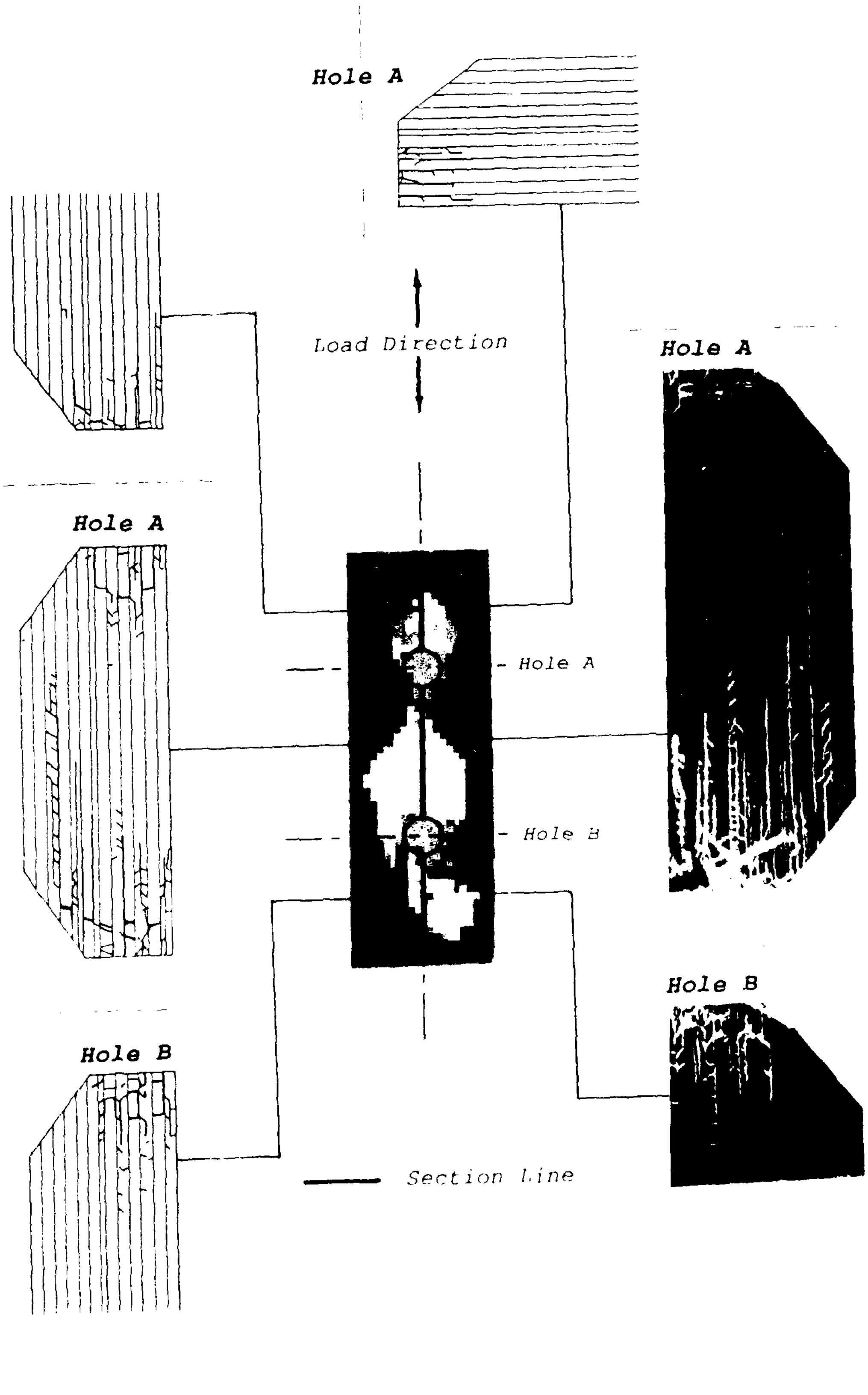

Figure 28. Damage maps from coupon 06-09, fatigued to 45 blocks MACSEQ (30 kN peak load)

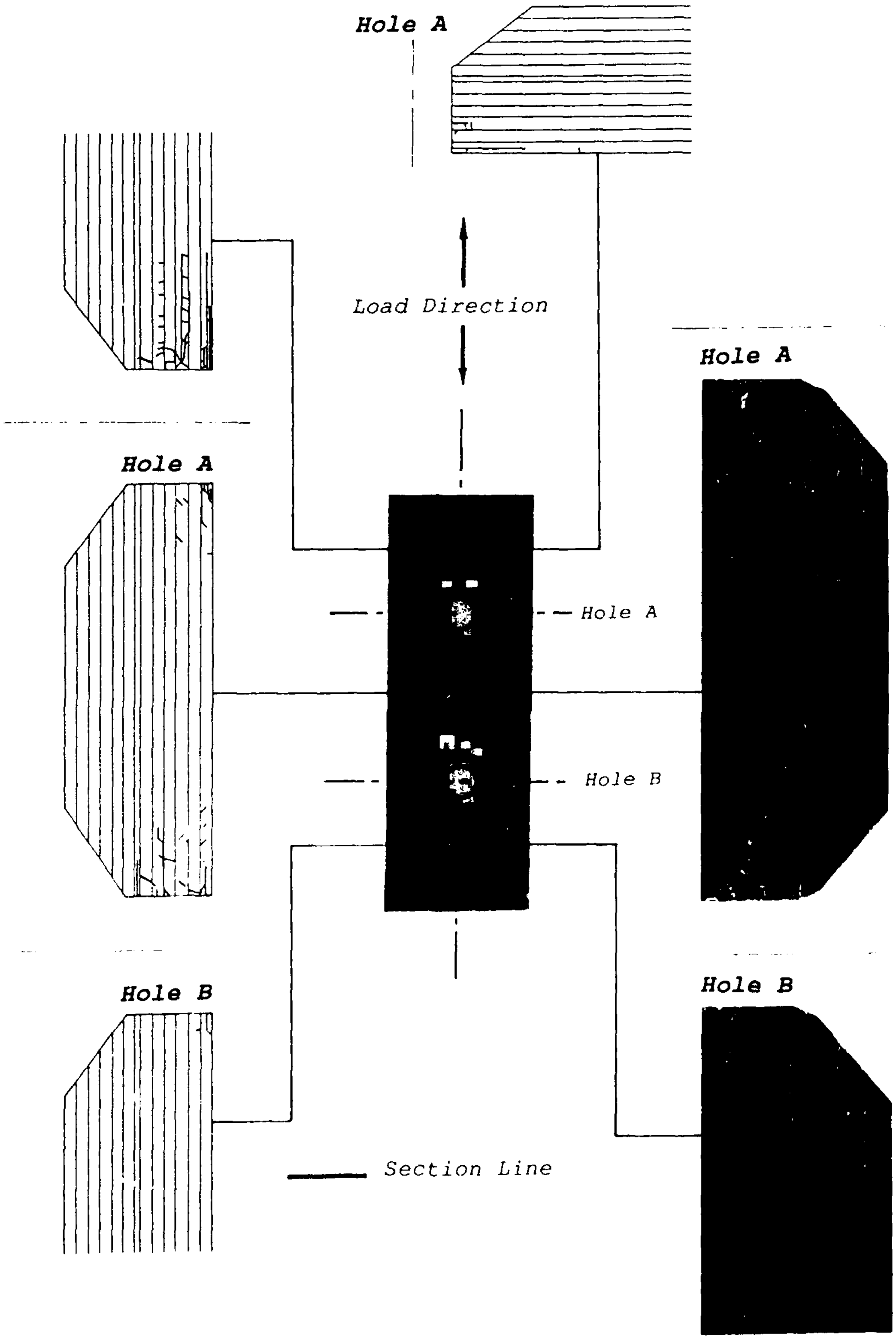

Figure 29. Damage maps from coupon 06-03, fatigued to 165 blocks MACSEQ (30 kN peak load).

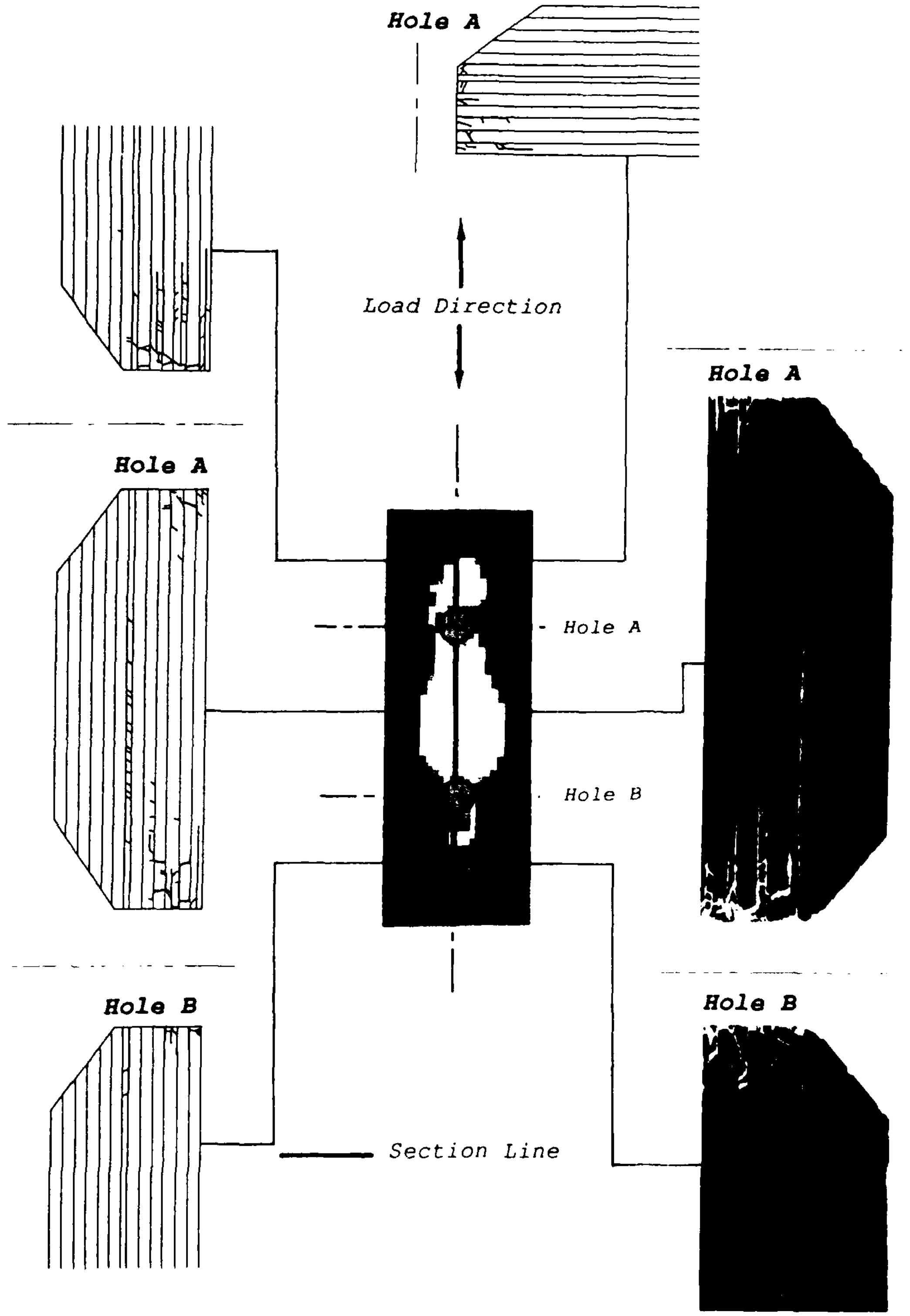

Figure 30. Damage maps from coupon 06-10, fatigued to 1 block MACSEQ (36 kN peak load).

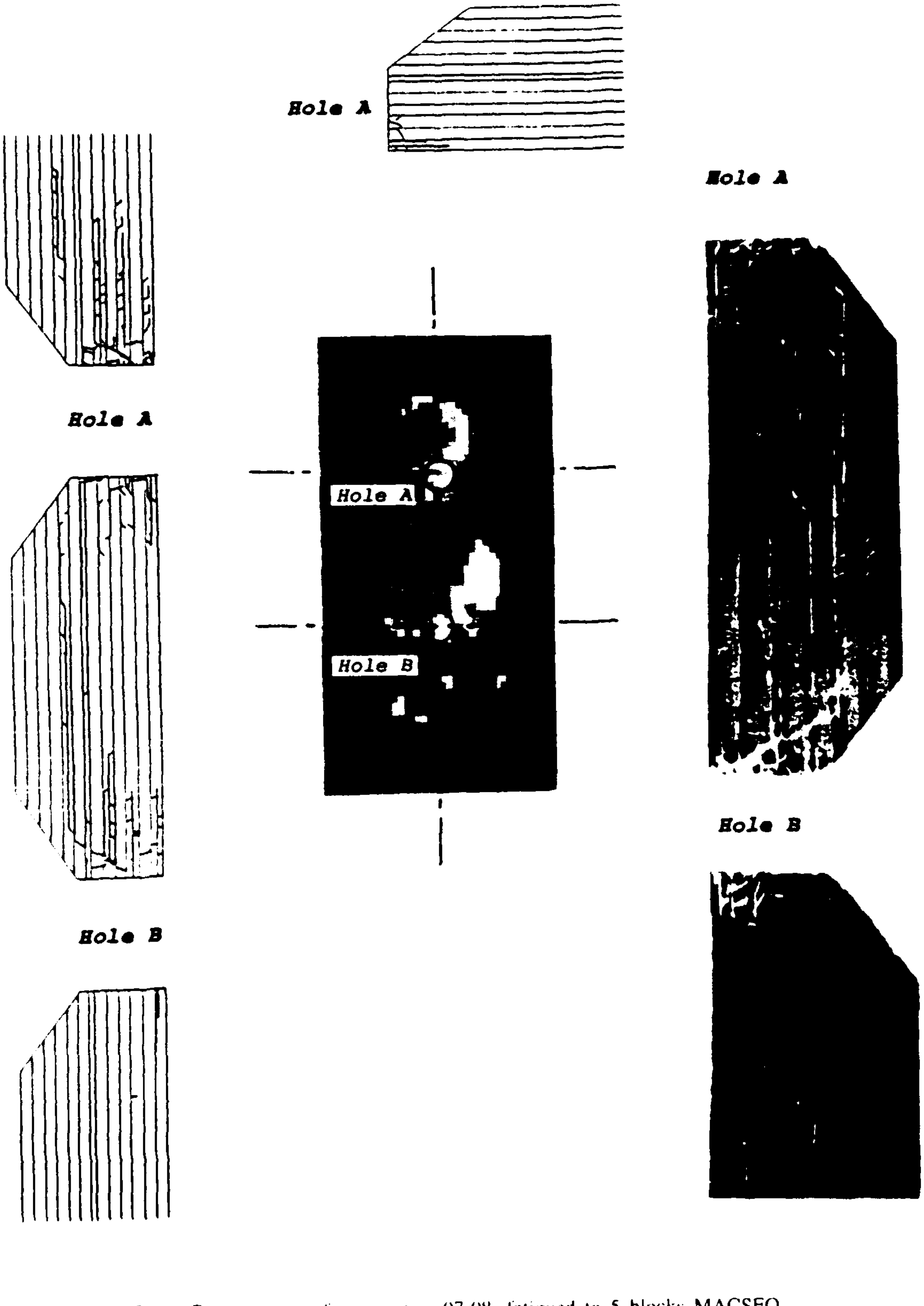

Figure 31. Damage maps from coupon 07-08, fatigued to 5 blocks MACSEQ (36 kN peak load).

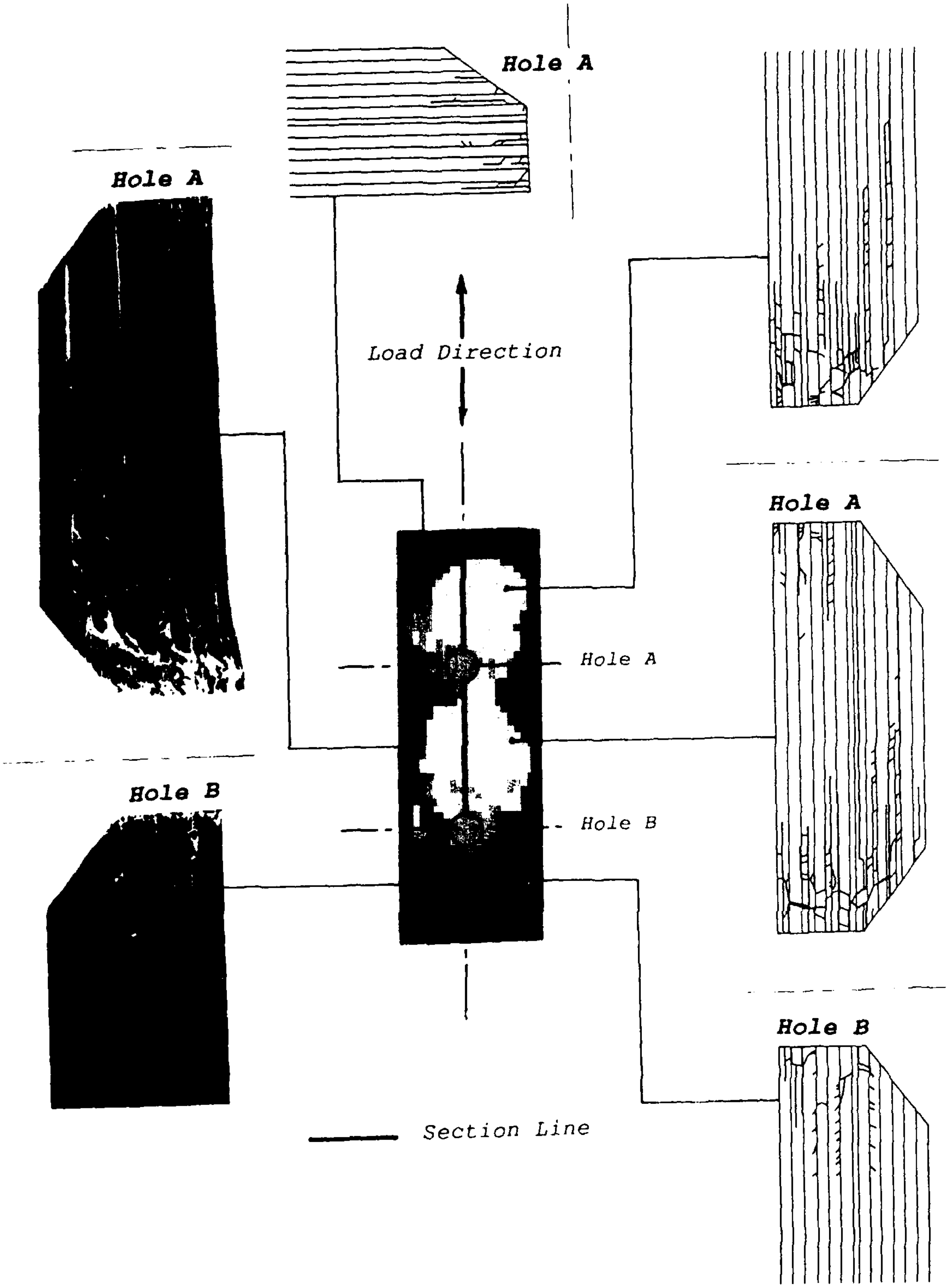

Figure 32. Damage maps from coupon 04-06, fatigued to 6 blocks MACSEQ (36 kN peak load).

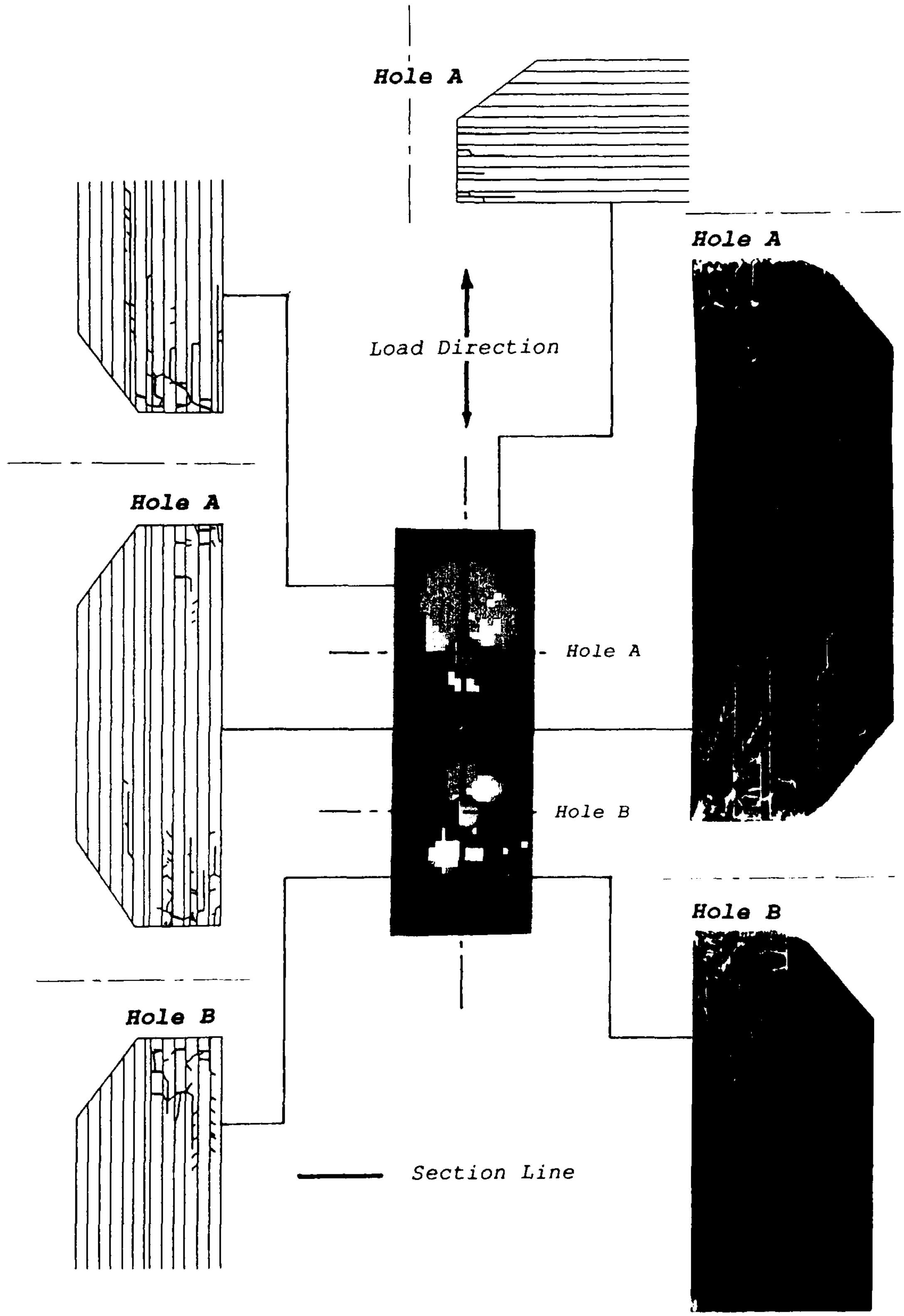

Figure 33. Damage maps from coupon 06-04, fatigued to 15 blocks MACSEQ (36 kN peak load).

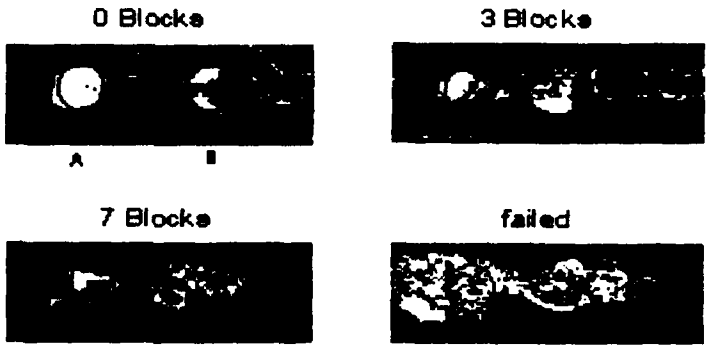

Figure 34. C-scans illustrating damage growth around the fastener holes of coupon 07-12 with increasing blocks MACSEQ (36kN).

Figure 35. C-scan of coupon 09-01, before fatigue testing, illustrating large delamination area due to incorrect panel fabrication.

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14. DESCRIPTORS

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16. ABSTRACT

This Report presents qualitative studies of the development of damage around countersunk fastener holes in thick graphite/epoxy laminates which have been subjected to fatigue loading. The experiments have shown that damage at the hole bore surface produces sites for the nucleation of delaminations. The morphology of the delaminations and ply cracking was mapped extensively. This Report discusses the methods used for the mapping of the damage and presents detailed maps of the damage for two loading conditions. In addition several methods for the non-destructive monitoring of damage development and growth were investigated. These were load/displacement hysteresis measurements, fastener tilt measurements using shadow moiré fringe techniques and ultrasonic time-of-flight C-scanning.

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16. ABSTRACT (CONT).

This Report briefly discusses the possible mechanisms by which damage develops and grows around the fastener holes. From the damage maps it was found that the volume of material around the fastener hole, damaged by the fatigue loading, adopted a characteristic shape; the volume of damaged laminate increased towards the faying surface of the laminate and (metal) fixture. This characteristic damage volume was generated by the fastener rocking under the fatigue loads. The growth of the delaminations has been shown to be preceded by intra-ply cracking and, as fatigue loading continued, more delaminations were generated at the hole bore surface.

The experimental work has shown that damage development around fastener holes is a complex process, usually producing several delaminations in the region of the fastener hole which grow and may ultimately lead to the failure of the joint.

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