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EVALUATION OF AIRCRAFT STRUCTURAL REPAIR/ANALYSIS CODES

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13. ABSTRACT *(Maximum 200 words)*
This study evaluates existing structural integrity analysis methods for the repair of aircraft structures, primarily focusing on composite (patch) to metal surface structures. This research was necessitated by the growing need to keep current aircraft in service well beyond their normal design lives. When defects are discovered during inspections the components must be either repaired or replaced. In most instances, it is not economically feasible to replace entire components. Therefore, repairing the damaged area(s) is usually preferred and critical. Additionally, repairs must be made quickly so that the aircraft may be returned to service as soon as possible. The results generated in this study evaluate the status of various repair analysis codes, determine which tools are potentially the most useful to ALC engineers, and provide information to assist Wright Laboratory engineers in deciding whether these codes address current and future US Air Force requirements. However, this evaluation does not intend to "recommend" or "disapprove" the use of any one software or methodology to Air Force, government or contractor personnel. Also, this evaluation of the composite repair/analysis codes relates solely to the versions that were available during the evaluation period of July 95 to 28 Feb 96. This report program covers the determination of ALC requirements, a review of current repair/analysis codes, the determination of equivalent capability, and an evaluation of repair/analysis codes.

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FOREWARD

This report was prepared by Analytical Services & Materials, Inc., Hampton Virginia for WL/FIBEC, Wright-Patterson Air Force Base, Ohio under contract F33615-94-D-3212, "Structural Integrity Analysis and Verification for Aircraft Structures." The U.S. Air Force project engineer was Mr. James A. Harter, WL/FIBEC. The government contract manager was Lt. Davis S. Conley. The period of performance for this report was July 1995 through February 1996.

The work performed under report (Delivery Order 0004) was performed by Analytical Services & Materials, Inc. personnel located at the WL/FIBE Fatigue & Fracture Test Facility, Bldg. 65, Area B, Wright-Patterson AFB, OH. The Principal Investigator of this research was Mr. Kevin L. Boyd. The authors of this report were Mr. Kevin L. Boyd and Mr. Srinivas Krishnan. Technical inputs were submitted by Mr. James A. Harter and Dr. Mohan Ratwani. Special thanks to Dr. Ratwani of R-Tec for his technical guidance and review of this report.

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1. INTRODUCTION

The purpose of this study was to evaluate existing structural integrity analysis methods for the repair of aircraft structures, primarily focusing on composite (patch) to metal surface structures. This research was necessitated by the growing need to keep current aircraft in service well beyond their normal design lives. When defects are discovered during inspections the components must be either repaired or replaced. In most instances, it is not economically feasible to replace entire components. Therefore, repairing the damaged area(s) is usually preferred and critical. Additionally, repairs must be made quickly so that the aircraft may be returned to service as soon as possible.

Presently, engineers at the ALCs (US Air Force Logistics Centers) do not have integrated analysis tools to assist in the design of repairs and subsequently perform Damage Tolerance Analyses (DTA) to assign inspection intervals to repaired structures. Therefore, it is important that analysis methods be developed and implemented to assess the structural integrity of repairs being performed at various US ALCs. It is not only important that analysis methods be developed, but they must be user friendly (easily used and understood by a variety of ALC personnel), accurate (methods verifiable by actual test or service data), fast (computationally), economical, upgrades readily available, easily integrated into the ALCs in a timely manner, well documented, and be supported by knowledgeable USAF personnel.

It is anticipated that the results generated in this study will fairly evaluate the status of various repair analysis codes, determine which tools are potentially the most useful to ALC engineers, and provide information to assist Wright Laboratory engineers in deciding whether these codes address current and future US Air Force requirements. However, this evaluation does not intend to "recommended" or "disapprove" the use of any one software or methodology to Air Force, government or contractor personnel.

It is also important to note that the evaluation of the composite repair/analysis codes relates solely to the versions that were available to AS&M during the evaluation period of July 95 to 28 Feb 96. It is entirely possible that the developers of the codes have implemented changes and/or enhancements that obsolete some of the findings of this report. AS&M believes that it is the responsibility of an engineer using this evaluation for guidance to assess any reported enhancements and determine whether or not they are available in the currently available version of their code.

This program has been broken down into four sections; determination of ALC requirements, review of current repair/analysis codes, determination of equivalent capability, and evaluation of repair/analysis codes. The determination of ALC requirements is described in Section 2. A review of current repair/analysis codes is described in Section 3. A determination of equivalent capability is discussed in Section 4. The evaluation of the repair analysis codes is described in Section 5.

2. DETERMINATION OF ALC REQUIREMENTS

Personnel from AS&M and R-Tec visited the five major ALCs in order to evaluate, from a research applications perspective, their repair and analysis requirements. The purpose of the visit was not to anticipate or evaluate the overall needs of the ALCs, but to gain a better understanding of the repair process, how repairs are performed at the ALCs, and determine (within the context of the evaluated codes) whether these institutions employed the personnel or had authority to take advantage of these software technologies. It was also intended to examine and discuss current and past composite to metal repairs performed at the ALCs and establish "benchmark" analysis problems by which the repair/analysis software programs could be evaluated.

A short questionnaire was also supplied to the engineers at the ALCs and can be found in Appendix A. From the interviews and questionnaires the capabilities of the ALCs could be better understood. In addition, a "wish list" of criteria (requirements) could be constructed to establish whether the repair/analysis codes address repair issues currently of interest to ALC engineers.

2.1. Interviews with ALC Personnel

The five major ALCs were visited in order to gain a "bigger picture" of their repair/analysis requirements. These ALCs were Oklahoma City (Tinker AFB), Ogden (Hill AFB), Sacramento (McClellan AFB), San Antonio (Kelly AFB) and Warner Robins (Robins AFB). While a complete analysis of the detailed responses from engineers would be next to impossible, the following paragraphs attempt to properly represent findings the authors find significant and applicable to this evaluation.

The list of ALC contacts was supplied to AS&M by Wright Laboratory personnel. Many of the interviewees were ASIP managers and/or members of the Structural Integrity Assessment and Life Extension Methodology Working Group, who work as representatives to the Wright Laboratory Aging Aircraft Program. Initially, AS&M and R-Tec personnel gave an overview of the evaluation to the engineers, followed by the presentation of a questionnaire to be returned whenever convenient. In most cases, the ALC engineers provided a brief tour of their facilities in which they discussed current and future problems, needs, reservations, and past experiences.

A modest amount of surveys (26) were handed out. Overall there was approximately a 42% return rate from the field. In most cases, it was determined that many of the respondents were from the same organization, so many of the engineers elected one of the engineers to complete the survey on their behalf. Otherwise there may have been a greater amount of respondents. In any event, AS&M appreciated the cooperation of all individuals who took part in the process whether by attendance, survey response, or on-site guidance.

It was interesting to note that most ALC personnel were familiar with composite repair, but had a lot of reservations (and doubts) about seeing the composite repair process being routinely implemented at their facilities. Two major reasons included untrained (or unskilled) on-site government personnel performing repairs and adequate verification ability for "certification" of repair patches. In general, it was felt that until these (and other) issues were resolved, the need to perform routine design/analyses of composite repairs was premature. In most cases, the ALC engineers expressed a preference for a good *bolted-repair* analysis program.

The exception to the above mentioned situation was encountered at Warner Robins ALC. Most routine composite repairs at Warner Robins are performed by specially trained government personnel. These personnel have applied over 1000 composite repair doublers to the C-141 fleet and these repairs were not included in a Technical Order. In addition, the prime contractors were not required to "approve" of the installations because the US Air Force is the approving authority for the aircraft.

Also, most routine composite repairs performed by an ALC were certified by the prime contractor and included in a Technical Order, describing its installation and application procedure. Any significant or unique repairs were designed, analyzed and installed by the prime contractor, or a qualified subcontractor approved by the prime contractor. It is also very important to note that while a loads analysis in the region of the anticipated repair is critical in the design of the repair, this information is not always available to the ALC engineers.

Therefore, the following section attempts to highlight several important facts and issues derived solely from the questionnaires.

2.2. Survey Responses

This section attempts to summarize the survey responses to the general questions and compare/contrast the ALCs visited. AS&M has attempted to the best of its ability to interpret and accurately represent the survey responses from the ALC personnel in this report. When survey questions weren't answered or returned, responses were derived from notes taken from interviews with ALC personnel. Any misinterpretations of survey questions by AS&M is purely unintentional.

First, it became very apparent that all ALCs were very different. This appears to be a combination of locality, labor mix, management, etc., but specifically the aircraft supported has the largest influence on mode of operation. Each aircraft has specific problems to address, due to their large differences in design, construction, materials, mission requirements and time in service. It appears that the ALCs repair/analysis needs are widely different, as well as their opinions surrounding composite repair to aircraft structure.

All of the facilities visited were ALCs, with "Depot" status for other aircraft. Also all of the facilities were very familiar for standard repair processes; routing or stop-drilling fatigue cracks, cold-working holes, replacing fasteners, replacing skins, etc. There were many bolted metal-to-metal repairs being performed, with widely varying levels of skill. As mentioned earlier, bolted metal repairs were mentioned as the most frequently performed, while often misunderstood (analytically) by ALC engineers. All of the ALCs mentioned repairing both fuselage and wing structures.

Repairing fatigue cracks was the primary source of damaged mentioned by personnel at all of the ALCs. Repair of corrosion was the largest concern with the KC-135 (Oklahoma ALC), while repairing "routine" corrosion was mentioned at every ALC, excluding Ogden ALC (F-16). Maintenance-induced damage was mentioned as a concern at Ogden ALC. Stress corrosion cracking was also listed as a concern at Oklahoma City (KC-135) and San Antonio (C-5) ALCs. All of the ALC personnel mentioned having some experience with aluminum and boron/epoxy as repair materials, with the addition of graphite/epoxy at Ogden and Warner Robins ALCs. San Antonio ALC mentioned having limited experience with GLARE as a repair material (C-5 fuselage upper crown).

The repair design approach varied according to aircraft, instead of ALC. Several aircraft employed a static strength design approach (KC-135, most of C-5), while most aircraft employed both static strength and damage tolerance analysis (DTA) approaches (B-52, E-3, F-16, C-5, F-5, T-37, T-38, C-130, C-141, F-15). All of the aircraft based their inspection intervals on a damage tolerance analysis. The KC-135, E-3, C-5, F-5, T-37 and T-38 aircraft also included in-service experience as a criteria for scheduling inspections.

The repair design procedure also varied according to ALC and aircraft. At Oklahoma (KC-135, B-52, E-3) and Sacramento ALCs (A-10, F-111) the prime contractor designs and approves most repairs. At Ogden ALC, contractors design repairs, while the prime approves the repairs. At San Antonio and Warner Robins ALCs both government and contractor personnel design repairs, while the appropriate U.S. Air Force organization (System Program Office) approves the repairs.

While there were no questions on the survey concerning the application of the repair, it was clear that in most cases bolted metal repairs were carried out largely by government personnel, while composite repair was largely performed by contractors (prime or other subcontractors).

Since one of the goals of this evaluation was to determine research directions for repair analysis software, it was important to understand the computing capabilities of the individual ALCs. These facilities also varied by ALC and aircraft. For example, at Oklahoma City ALC, engineers supporting the E-3 have access to a Mainframe computer and PCs, while engineers supporting the KC-135 only have access to PCs. However, an engineering support organization at Oklahoma City also has access to a computer workstation. Ogden, San

Antonio, and Warner Robins ALC engineers have access to mainframe, computer workstations, or PCs. Sacramento ALC engineers only had access to PCs.

The same was true with access to computer software. It appears that the more sophisticated the computer hardware was available, the more sophisticated analytical tools were available to the engineers. At Oklahoma ALC, most finite element analyses were performed by the prime (KC-135, B-52) or other contractors (E-3). Only the engineering support group (ES) had an on-site finite element capability available to government employees. Ogden ALC had its analytical support in terms of on-site contractors. San Antonio and Warner Robins ALCs had on-site civilian personnel capable of performing most analyses (FEA and DTA). Sacramento ALC mostly depended on outside analytical support from contractors.

One major reason for the limited number of on-site civilian personnel to support analytical studies was the absence of geometry, material and loads data from the prime contractor. Oklahoma ALC personnel had expressed a lack of data for the KC-135, B-52 and E-3 aircraft. Oklahoma, Sacramento, and Warner Robins ALC personnel claimed to have limited load and structural detail data, but never as much as they would like. Ogden and San Antonio ALC personnel seemed to be the exception, with no expressed desire to acquire more (or better) data of this type. Oklahoma (KC-135, B-52), Ogden, San Antonio and Warner Robins ALC personnel were confident that analyses were performed to account for the redistribution of stresses due to the presence of a repair. Sacramento ALC personnel weren't sure if their contractors performed this type of analysis. Oklahoma ALC (E-3) personnel said they did not perform this type of analysis. Oklahoma, Ogden, San Antonio and Warner Robins foresee the need for a non-FEM-based DTA tool to assess the structural integrity of repaired structures. Sacramento ALC personnel weren't sure of their analytical requirements since their contractors performed most of this type of analysis.

Responses concerning the major need in repair/design procedure varied between the ALCs. Some ALCs (Oklahoma, San Antonio) focused on standardizing composite repairs, making them faster and less expensive, and incorporating skilled personnel for patch installation. Other ALCs (Ogden, Warner Robins) were more concerned with developing better, faster, more user-friendly analytical tools to evaluate repair patch effectiveness. There were also additional comments regarding repair, patch feasibility, improved NDI techniques, and developing mechanically-fastened design/analysis tools.

2.3. Development of Evaluation Criteria

AS&M and R-Tec personnel developed a set of criteria, from experience and ALC interviews by which to rate the candidate repair design analysis tools. These criteria were designed to be a "wish list", or "perfect repair code" capability statement. By creating such lofty criteria, it can therefore be surmised how much work there is to be done in this area.

The criteria were broken down into primary and secondary criteria. The primary criteria would be essential to performing an adequate design and analysis, allowing for routine engineering assumptions. The secondary criteria would be viewed as "icing on the cake." The criteria established were:

Primary Criteria:

1. User Friendliness
2. Reliable Crack-Growth Life Predictions
3. Damage Growth Rate Prediction in Repaired Structure
4. Compute Critical Crack Length in Repaired Structure
5. Compute Residual Strength in Repaired Structure
6. Identify Inspection Requirements for Repair
7. Account for Patch Moisture Absorption
8. Account for Thermal Mismatch Between Repair Patch & Structure
9. Ability to Handle Complex Geometry's
   a. Finite Geometry's (Thick & Thin Patches)
   b. Substructure Effects
   c. Cracks at Holes
   d. Cracks at Loaded Holes
   e. Curvature Effects

Secondary Criteria

10. Address Multiple Failure Modes
    a. Failure of Parent Metal Structure
    b. Adhesive Disbonding and Failure
    c. Failure/Delamination of Repair Patch
11. Address Multi-Site Damage
12. Predict Crack Initiation Life in Repaired Structure
13. Ability to Account for Load Redistribution in Structure
14. Ability to Account for Corrosion Damaged Structure

The candidate codes were to be evaluated according to both these criteria and benchmark repair situations described later in Section 4.0.

3. REVIEW OF CURRENT REPAIR/ANALYSIS CODES

In the following section a brief description of the candidate repair/analysis codes is given. The purpose of this section is to give background information and describe the intended usage and capabilities of the repair/analysis codes. Further information regarding the codes and their performance against the above criteria and benchmarks will be described in Section 4.0

The specific goal of this section is to introduce readers to the candidate analytical methods as advertised by the developers. Many of the comments come from a combination of reports and contractor brochures. It is not the intent for AS&M to state any capabilities or intentions of the candidate programs, other than those advertised by the developers.

3.1. Advanced Composites Repair Analysis Tool (ACRAT)

The Advanced Composites Repair Analysis Tool (ACRAT) program was managed out of the Sacramento Air Logistics Center (SM-ALC) Advanced Composites Program Office (ACPO) by Mr. James Song. The ACRAT program is a two-phase program, the first phase from 1 Sept 92 to 1 Sept 93, and the second phase covering the next three years. The ACRAT program is being developed by BDM as the prime contractor (contract management, analysis code integration) and MSC/PDA (ACRAT software development, database environment, repair testing) as their subcontractor. The goals of the ACRAT program are to improve reliability of aircraft structural repairs, provide a medium to train Air Force structural engineers in designing and performing composite repairs, facilitate repair technology in the US Air Force and increase the maintainability of composite aircraft structures.

The ACRAT program was developed as a knowledge-based, engineering decision support system. The program will provide the information necessary to assist in the definition of an appropriate composite repair. Otherwise, users can add their own information to the programs database. The ACRAT program serves as an extensive database query device for the retrieval of information necessary to assist in the design and analysis of repairs for specific (or general) US Air Force weapons systems. The ACRAT program allows the use of several diverse functional environments and procedures in order to span the broad range of technical disciplines and manufacturing environments needed to support generic composite aircraft structural component repairs. The ACRAT program was designed to provide users with a full range of database functionality, a comprehensive suite of analytical and modeling tools, and knowledge-based procedures to guide a user to the appropriate repairs for specific damage situations.

The ACRAT system is computer-workstation based, and provides one simple graphical user interface (GUI) environment in which to assist a user through all of the required steps necessary to design a composite repair for a repair situation on a specific aircraft. The

ACRAT program attempts to make use of a combination of commercial software, such as MSC/PDA's M/VISION and P3/PATRAN whenever necessary.

3.2. CALCUREP

The CalcuRep program was developed by Maj. Rob Fredell at the Delft University of Technology (DOS Version) and further at the United States Air Force Academy (PC-Windows Version). The CalcuRep code is based on a mathematical model developed by L.R.F. Rose [1], which is based on four theories: elasticity theory, fracture mechanics, the theory of bonded joints and heat transfer.

The Rose model of crack patching is based on an infinite, center-cracked, isotropic plate with a bonded orthotropic elliptical patch on one side of the plate. The plate is loaded by a remote biaxial stress. The Rose model makes its calculations in two steps. In the first step, the calculations ignore the crack in the fuselage. This analytical procedure calculates the stress redistribution in the skin in the presence of a bonded repair. In the second step, a crack is introduced into the plate; the stresses at the crack tip are allowed to relax to zero, causing the stress distribution to change sharply. The primary outputs of this second phase are the repaired stress intensity factor and maximum tensile stress in the patch. This closed-form (continuum) analysis, elasticity-based approach was developed to allow maintenance engineers with limited knowledge of fuselage design stress levels to design and analyze bonded patch repairs to metallic structures.

This user-friendly, crack-patching code calculates the stress intensity factor at the crack tip after the repair, maximum tensile stress in the repair patch, maximum tensile stress in the repaired skin, maximum shear strain in the adhesive and load transfer length. These results are compared with design guidelines in the code, that are determined by the developers to be valid for typical operating loads in aircraft fuselages (pressure loading, cruising altitude).

For example, the guideline for the repaired stress intensity factor is based on research on the influence of residual stress on fatigue crack growth rate (based on typical operating loads on aircraft fuselage structure). The guideline for maximum stress in the patch is one-half (or below) the yield stress of the patch material in the longitudinal direction. The guideline for maximum stress in the skin is a combination of load transfer, load attraction, bending and thermal effects. The maximum skin stress guideline is also one-half (or below) of the skin's yield stress. CalcuRep suggests good design practices by designing patch tapers to 1:10. The guideline for maximum adhesive shear strain depends on the yield strain of the adhesive used and recommends designing a repair such that the maximum shear strains are equal to (or below) one-half of the maximum shear strain of the adhesive. The guideline for load transfer length is a conservative rule of thumb based on the research of Hart-Smith [2] recommending that the patch length be 40 times the load transfer length or greater than (or equal to) 160 times the maximum repair patch thickness.

The aircraft input variables are skin material and thickness, frame and stringer spacing, crack length, maximum operating altitude, cabin pressure and fuselage radius. The repair patch inputs are patch material and dimensions, adhesive type, cure temperature and heat blanket dimensions. Most of the material-related properties and common patch materials are found in the CalcuRep program's database, for quick, simple retrieval. If one or more of the design guidelines (repaired stress intensity factor, maximum patch stress, maximum skin stress, maximum adhesive shear strain, load transfer length) are not met, CalcuRep informs the user of ways to improve the patch design. CalcuRep does this by way of an on-line help function offering suggestions to the user.

3.3. COMPAT_3D

COMPAT_3D is a software package for analysis of fracture in three-dimensional bodies with or without adhesively bonded repair patches. The program is based on the Finite Element Alternating Method (FEAM) and is applicable where the assumptions of Linear Elastic Fracture Mechanics (LEFM) are valid. The COMPAT_3D FEAM code is currently being developed by Knowledge Systems Inc. by Dr. Daniel S. Pipkins and Prof. S.N. Atluri (Georgia Institute of Technology, Georgia Tech. Univ.). Versions of the code are available for both the personal computer and UNIX-based workstations.

The software is distributed in a package consisting of two programs; COMPAT_3D, and PATCHGEN_3D, the preprocessor used to generate input files for the main program. PATCHGEN_3D, in turn, accepts a mesh created by an external finite element code, as input. This finite element code is typically a commercial package, such as P3/PATRAN or MSC/NASTRAN. PATCHGEN_3D translates the mesh and loading and boundary conditions and accepts other data relevant to the analysis, such as crack configurations and locations. It then creates an input file for COMPAT_3D, which must be executed separately.

The FEAM is a relatively quick and accurate method for calculation of stress intensity factors in flawed three-dimensional solids [4]. It is valuable to the analyst as only one finite element mesh (of the unflawed structure) is required and only one finite element analysis is performed, reducing both preprocessing and computation time and effort. The method requires one finite element solution of the uncracked structure, which is combined with the analytical solution for an infinite body with the same crack configuration in an iterative procedure to obtain the solution for the cracked structure.

The types of flaws COMPAT_3D is capable of analyzing are limited to fully-embedded elliptical cracks, semi-elliptical surface cracks and quarter-elliptical corner cracks. COMPAT_3D also has constant-amplitude fatigue life prediction capability. This feature, however, is restricted by the code's inability to analyze through-the-thickness cracks, or any crack configuration that is not one of the three types mentioned above. Restrictions are also placed on allowable aspect ratios of the elliptical or part-elliptical flaws. The program requires that the major axis to minor axis ratio not be below 1.15.

The program does not offer a graphical user interface. The user must rely on an external package to create the mesh and then translate the mesh using PATCHGEN_3D, which is keyboard-driven. PATCHGEN_3D is adequate and easy to use, and offers options to edit material properties, crack sizes and locations, etc. COMPAT_3D is a "silent" analysis engine that accepts a file as input and writes out results to other files. Included among the output files are files that contain deformation and stress data that can be read into a commercial package such as PATRAN to view certain results graphically.

3.4. FRacture ANalysis Code for Plane Layered Structures (FRANC2D/L)

FRANC2D/L is a workstation-based software tool designed to perform simulation of discrete crack growth in two-dimensional layered structures [5,6]. Structural behavior is modeled using the finite element method, and the fracture calculations used by the software are based on linear elastic fracture mechanics (LEFM) concepts. The FRANC2D/L computer code was developed by Prof. Anthony Ingraffea (Cornell University) and several generations of graduate students. Currently, FRANC2D/L is being developed by Prof. Daniel Swenson (Kansas State University) and is being supported by NASA Langley Mechanics of Materials Branch. The program utilizes a menu-driven graphical user interface (GUI) based on the X Window system. Versions of the code are available for a variety of UNIX platforms. The code is distributed as a package which includes a mesh generator (CASCA) developed especially for use with FRANC2D/L, and a translator to read meshes generated by CASCA into FRANC2D/L. Translators are also available for meshes created using the commercial codes, ANSYS and PATRAN.

Mesh generation is performed externally, using CASCA, ANSYS or PATRAN, and each layer is modeled individually. The meshes are then combined, using one of the translator codes, to create a data file that is then read into FRANC2D/L. The data in this file represents a model of the uncracked structure. All other preprocessing, including modeling of cracks is performed using FRANC2D/L.

FRANC2D/L is capable of analyzing fracture in layered structures such as lap joints or bonded repairs. While each layer is an individual two-dimensional structure, it is possible to capture out-of-plane displacements and bending effects. Connectivity between adjacent layers can be modeled using either adhesives or rivets. The finite elements available in the code include eight-node quadrilateral elements, six-node triangular elements, sixteen-node and twelve-node adhesive elements, two-node rivet elements and six-node nonlinear interface elements.

A maximum of five layers is allowed in the current version of the code. There is no limit to the number of cracks in the model, and the code may also be used to perform a two-dimensional stress analysis on the uncracked model.

3.5. Repair Assessment Procedure and Integrated Design (RAPID)

The RAPID software program is a tool for the design and analysis of mechanically fastened aircraft fuselage structural repairs. The development of the RAPID Software program is funded by the Federal Aviation Administration (FAA) through Galaxy Scientific. The major Subcontractors are McDonnell Douglas Aerospace (analysis methods) and Northrop Corporation (GUI Development). The FAA technical Manager is Dr. Paul Tan and the program manager is Dr. John Bakuckas, Jr.

The current version of RAPID is PC-Windows based, and can assist a user in designing mechanically fastened repairs with up to two doublers. The RAPID program performs both static and damage tolerance analyses of the repair. The static analysis determines if the doublers are statically adequate, while the damage tolerance analysis yields inspection intervals and residual strength.

RAPID version 1.1 limits repairs to fuselage skin repairs between the forward pressure bulkhead and front wing spar and between the aft fuselage forward bulkhead and aft pressure bulkhead. RAPID can also be used in areas away from splice joints, door cutouts, and the window belt.

The static strength analysis calculated the fastener joint allowable and repair doubler allowable. The margins of safety (MS) based on the repair doubler allowable and the fastener joint allowables are calculated to determine the adequacy of the repair. Other requirements such as stiffness, compressive, and shear strengths are not considered in the current versions of RAPID. The load carrying capacity lost due to the skin cutout is calculated based on the design ultimate tensile strength of the skin.

In the damage tolerance analysis part of the RAPID program, it is assumed that each fastener row carries the same amount of load. Fastener loads are calculated using a one-dimensional strip model using Swift's fastener stiffness equation [7]. The critical fastener is assumed to be in the center of the critical fastener row. The fastener loads are calculated based on the uncracked repaired skin configuration and the load transfer in each fastener remains constant throughout the damage tolerance analysis. The total number of fastener rows is limited to 10.

The damage tolerance analysis assumes a longitudinal through crack emanating from the critical fastener hole. Currently there are two initial flaw patterns available in RAPID. The first, being a single through crack emanating from the critical fastener hole that has a length of 0.05 inch. The second being two asymmetric cracks emanating from the critical fastener hole. The primary crack has a length of 0.005 inch and the secondary crack has a length of

0.005 inch. Subsequent crack growth sequences are assumed, based on the case selected to grow toward the adjacent hole(s). For the single crack tip case, when the crack tip reaches the adjacent hole, two 0.0054 inch cracks are assumed to exist instantaneously, one at each of the outer holes. For the double crack case, a through crack equal to 0.005 inches is assumed to exist instantaneously at the opposite side of the hole when a crack tip grows into the hole. Methods of superposition, compounding and similarity are used to determine the SIF needed during crack growth. The effects of interference-fit fasteners and clamp-up forces are neglected.

Crack growth is predicted using five baseline stress intensity factor solutions in RAPID and a simplified method based on Walker's crack growth equation. Walker coefficients are included for 12 common materials. Loading cases are provided in the form of a constant amplitude spectrum.

The limit stress in the circumferential direction is equal to $1.1*pR/t$, where p is the operating pressure differential (plus 0.5 psi), R is the fuselage radius, and t is the skin thickness at the repair location.

The RAPID program also calculates the first inspection and inspection intervals for an assumed initial crack size of 0.05" to grow to critical size under the limit load condition. The inspection interval can be determined according to the NDI method of crack detection.

4. DETERMINATION OF EQUIVALENT CAPABILITY

The determination of equivalent capability was not straightforward. It became obvious that each program had a different set of objectives and procedures. For example, the repair/analysis information from ACRAT (P3/FEA), COMPAT_3D, and FRANC2D/L would be finite-element-based (displacements, stresses, strains, etc.). The analysis information for CalcuRep is based on a closed form solution (Rose Model) to give stresses (metal, adhesive, composite patch) and stress intensity factors. The RAPID program was developed to design and analyze metal-to-metal bolted repairs with elasticity-based numerical solutions and gives analysis information based on stresses, strains, residual strength and fatigue life.

Both the ACRAT and RAPID programs were excluded from the analysis for reasons stated in Section 5.0. Therefore, it became obvious that one of the common denominators of the remaining codes was the calculation of K, the stress intensity of the crack tip under a repair patch.

The three benchmark programs were chosen to evaluate the repair/analysis codes. One "academic" problem, and two in-service repairs problems, namely the C-141 "weep-hole" problem, and the T-38 lower wing skin cracking problems were chosen. These three problems were chosen because of their order of increasing complexity, existence of prior information, realism, and focus on US Air Force wing structures. The lack of inclusion of a an aircraft fuselage benchmark problem was by design, because it was described to AS&M by government personnel that other governmental agencies (FAA, NASA) were already addressing problems related to aircraft fuselage structures.

4.1. Development of Benchmark Repair Problems

Three benchmark programs were chosen based on increasing complexity. The first problem was a purely academic problem, which was believed to be simple enough for all of the codes to solve easily. The second problem that was chosen was a C-141 "weep-hole" type problem, which is well documented by the Air Force and known to be a composite-to-metal repair/analysis application [8]. The third problem was a T-38 lower wing skin problem, one familiar to R-Tec personnel and documented in a report by the Northrop Corporation [9]. This third problem (T-38 lower wing skin) is not as well known as the C-141 weep-hole problem, but offers an extremely challenging composite repair/analysis application. As was mentioned earlier in this report the "realistic" benchmark problems chosen in this evaluation were wing-related by design. The combination of complex geometry, aerodynamic implications and extremely high loading conditions make these repairs the most challenging.

4.1.1. Academic Repair Problem

Since most cracks emanate from some form of inclusion of fabrication/manufacturing anomolies, especially around fastener holes, a simple plate with a hole was chosen for this problem. The following configuration for this problem is shown in Figure 1.

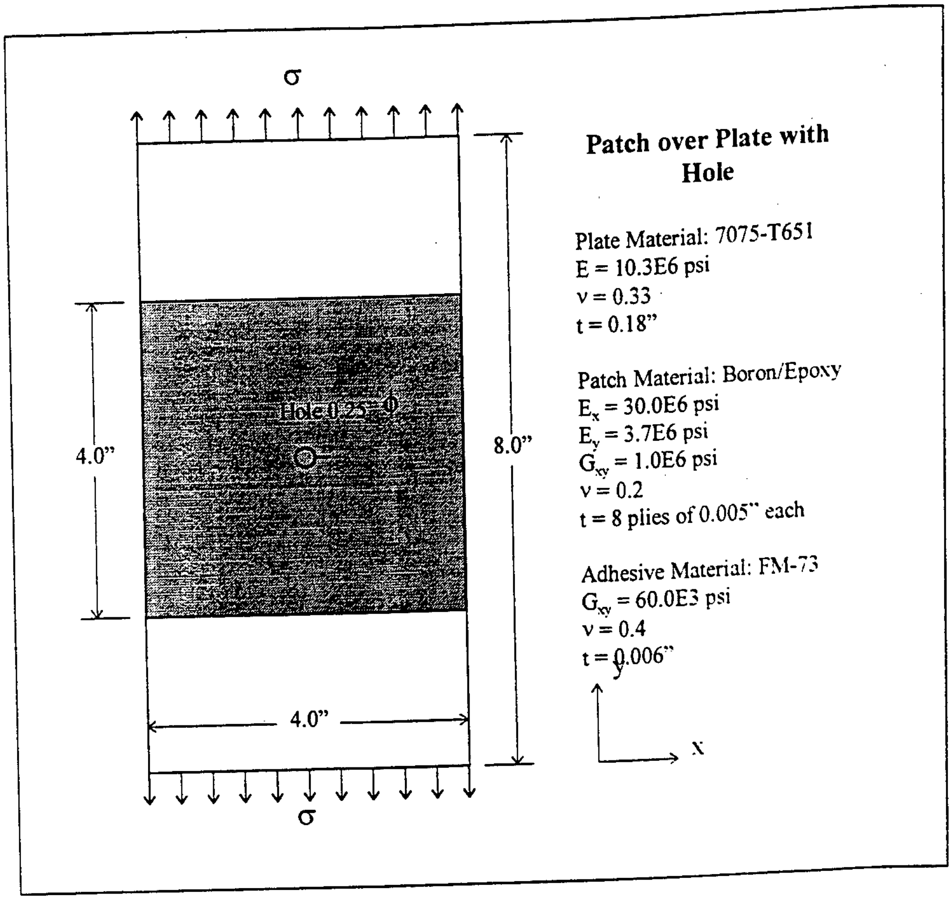

Figure 1. Academic Repair Problem

The loading to be imposed on this model was to be a remote stress of 18.0 ksi in the longitudinal direction. If desired, a unit load could be applied to the finite element models in order to normalize the stress intensity factors. Two different crack lengths were chosen for the analysis 0.1 inches and 0.25 inches. For codes evaluating part-through elliptical flaws , a 0.1 X 0.08 inch (c x a) crack case was determined to be sufficient for the problem.

It should be obvious to anyone familiar with composite repairs to metal structure that this repair is purely academic. (The patch properties were based on all zero degree plies, covering the entire width of the specimen, not tapered and of questionable load transfer length.) The specific reasons for analyzing this type of repair was first, simplicity of modeling and second, to determine if the codes would recognize a poor design.

4.1.2. C-141 Weep-Hole Problem

The second benchmark problem was based on a severe weep-hole cracking problem that was analyzed and repaired at Warner Robins ALC. The C-141 lower wing skin weep-hole cracking was found in service on a number of aircraft and have been repaired with boron/epoxy patches. An illustration of the problem is described in Figure 2.

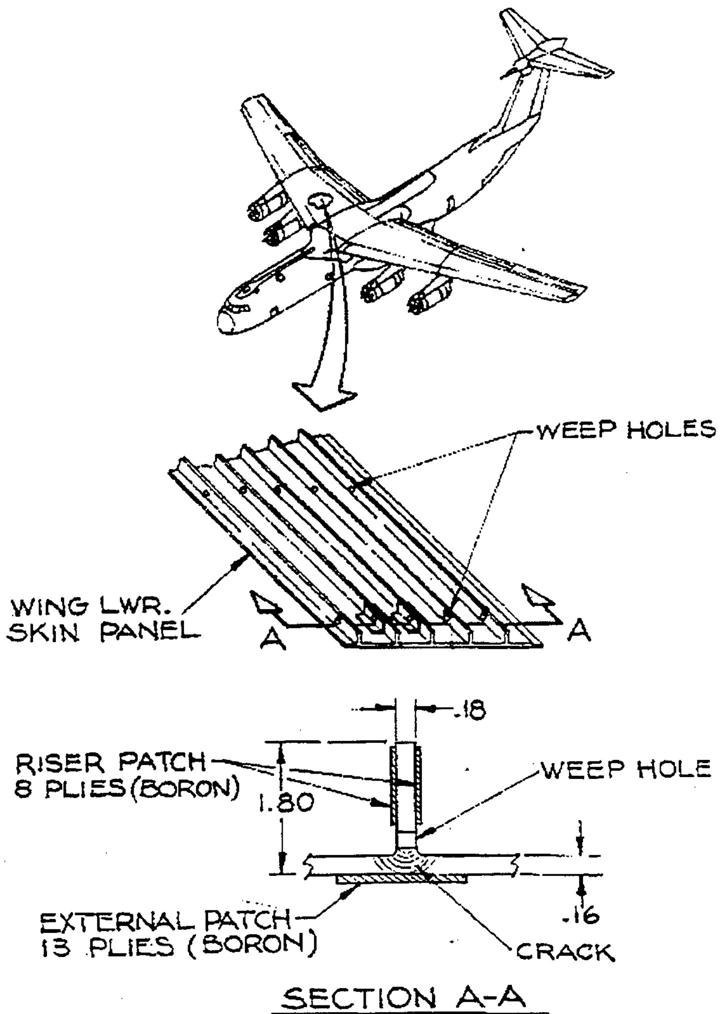

Figure 2. C-141 Weep-Hole Benchmark

For this benchmark problem the loading condition was to be 18.0 ksi, far field stress in the spanwise direction. If desired, a unit load could be applied to the finite element models in order to normalize the stress intensity factors. The through-crack length chosen for the analysis was 0.1 inches. For codes evaluating part-through elliptical flaws, a 0.1 X 0.08 inch (c x a) crack case was determined to be sufficient for the problem.

4.1.3. T-38 Lower Wing Skin

The third benchmark problem examined was derived from the T-38 lower wing skin repair at the 44% spar and "D" panel. The cracking problem was found in service at this location and has been repaired with boron/epoxy composite patches. At the time of this evaluation four wings had been repaired on three aircraft. The details and the location of the repair patches are shown in Figure 3.

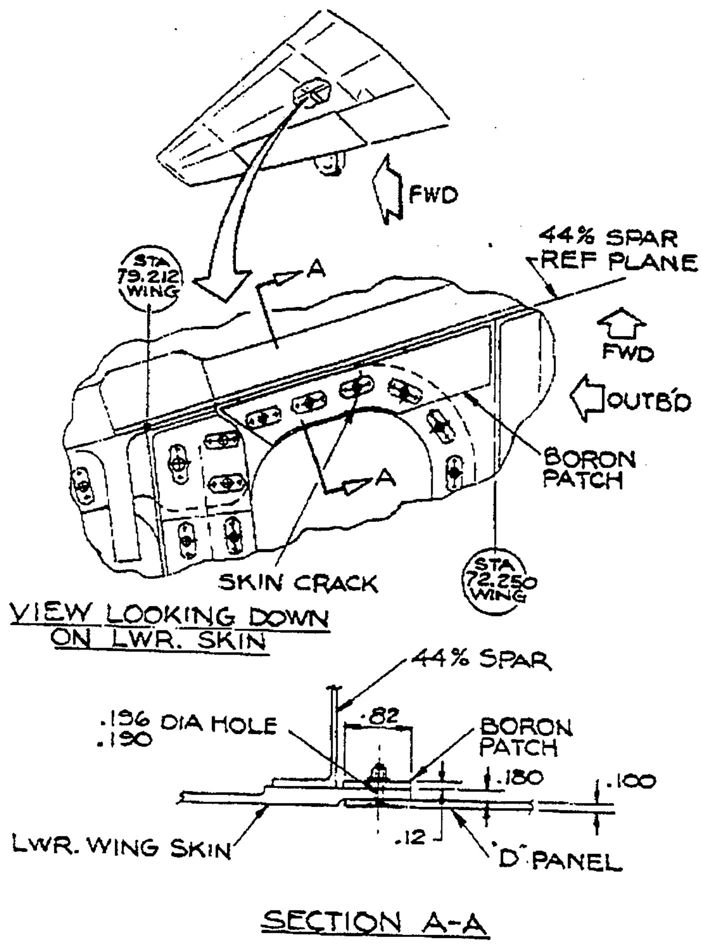

Figure 3. T-38 Lower Wing Skin Benchmark

For this benchmark problem the loading condition was to be 31.0e3 psi, far field stress in the span-wise direction. If desired, a unit load could be applied to the finite element models in order to normalize the stress intensity factors. The through-crack length chosen for the analysis was 0.1 inch. For codes evaluating part-through elliptical flaws, a 0.1 X 0.08 inch (c x a) crack case was determined to be sufficient for the problem.

5. EVALUATION OF REPAIR/ANALYSIS CODES

5.1. ACRAT

The ACRAT code was not evaluated against the benchmark problems, because AS&M could not receive an evaluation copy of the program. Government personnel attempted to obtain a copy of the ACRAT program, but could not obtain a copy that could run on a Silicon Graphics workstation. During the evaluation period, the Beta version of the code was only available for Hewlett-Packard workstations. AS&M had obtained copies of the M/VISION and PATRAN codes and has experience with these two MSC/PDA products. AS&M feels that it has the experience to comment on these codes. Government personnel were told that porting the code to an SGI workstation was possible, providing they were funded to do so. Comments addressing the overall applications and general capability of the ACRAT code are discussed in the following paragraphs.

5.1.1. Comparison Against Evaluation Criteria

Even though the ACRAT code was not run against the benchmark problems, based on knowledge of the program capabilities, it can be rated against the evaluation criteria. The ACRAT program was compared to the evaluation criteria and the results are shown in Table 1.

Table 1. Comparison of ACRAT to Evaluation Criteria

Primary Criteria	*Comment*
1. User Friendliness	Somewhat user friendly. More suitable to UNIX software users.
2. Reliable Crack-Growth Life Predictions	No capability in current version.
3. Damage Growth Rate Prediction in Repaired Structure	No capability in current version.
4. Compute Critical Crack Length in Repaired Structure	No capability in current version.
5. Compute Residual Strength in Repaired Structure	No capability in current version.
6. Identify Inspection Requirements for Repair	No capability in current version.
7. Account for Patch Moisture Absorption	Can incorporate through material property database (M/VISION)
8. Account for Thermal Mismatch Between Repair & Structure	Can incorporate through FEA code.
9. Ability to Handle Complex Geometry's	Can incorporate through FEA code.
Secondary Criteria	*Comment*
10. Address Multiple Failure Modes	Yes. Through multiple FE analyses. Very Laborious.
11. Address Multi-Site Damage	Yes. Through FE analyses. Very Laborious.
12. Predict Crack Initiation Life in Repaired Structure	No capability in current version.
13. Ability to Account for Load Redistribution in Structure	Yes. Through FE analysis.
14. Ability to Account for Corrosion Damaged Structure	Yes. Through FE Analysis.

5.1.2. Discussion

The ACRAT program is not a repair/analysis code by itself. It is rather a repair/analysis expert system (user environment) that assists engineers in the design of a repair by providing them with the information necessary to design and analyze a repair. The ACRAT program would perform as a "seamless" interface between the products bundled within the ACRAT environment. The M/VISION product would provide such information as geometry, material properties, loads, design information, etc., and P3/PATRAN (in conjunction w/FEA) would perform the analyses. The apparent usefulness of the ACRAT code lies in its expert system approach, integrated environment, and existing population of database information. If the information is available in the database for the specific weapons system of interest, this program could save someone a great deal of valuable time. If the information is not available to the government (loads, structural details, etc.) or digitized within the ACRAT database, the program is of limited help. In these cases the information will have to be obtained the old-fashioned way: through the literature, telephone calls and a lot of leg work. An engineer would then directly proceed to the analysis part of the program, as he was doing a finite element analysis (NASTRAN, ABAQUS). Of course it helps to have experience or training on the individual codes themselves (M/VISION, P3/PATRAN, etc.,). These codes are relatively expensive to purchase and maintain.

The ACRAT program is a very ambitious effort. Its reported capabilities to integrate manufacturing, geometry, materials, loads, geometry, CAD, etc.; information into a knowledge-based environment is a state-of-the-art concept. However, the level of effort required to complete, successfully transition and maintain this program may be its largest obstacle.

In all fairness to the ACRAT program, these evaluation criteria were only designed to meet composite-to-metal structural repair issue requirements. The ACRAT program was designed to meet design and analysis criteria including database and spreadsheet functions that include aircraft design, analysis (2D&3D), M&P, repair (composite-to-composite) and material data information. It would be very biased to examine one aspect of these many functions and blame the developers for overlooking these specific requirements. ACRAT was intended as a knowledge-based environment that integrates many functions that must be considered in the repair/analysis process. In other words, ACRAT may be overkill for many users that only want to consider one aspect of the repair process. However, it could be a very useful source of information (structural details, material properties, etc.,) to assist in the design of an aircraft repair. A multi-platform version of ACRAT would be a very valuable source of information and suite of analysis tools for more sophisticated users. In addition, the pure cost of development and stringent computer hardware requirements of the ACRAT program, may prevent it from becoming widely available to users.

5.1.3. Future Directions

Presently, the ACRAT Program's development has been stopped due to a lack of funding. The future of this program is uncertain, as was conveyed by the ACRAT Program Manager Mr. James Song. It is assumed that the ACRAT Program was halted somewhere during Phase II of their anticipated effort, because their Beta 2 version of the ACRAT program has been demonstrated. For further information regarding the ACRAT program, please contact Mr. James Song at Sacramento ALC, Advanced Composites Program Office.

5.2. CALCUREP

The CalcuRep program was compared both to the evaluation criteria and the benchmark problems. The benchmark problems were simplified in order to evaluate the CalcuRep program capabilities. The results can be found in the following sections.

5.2.1. Comparison Against Evaluation Criteria

A comparison of the CalcuRep program to the evaluation criteria is shown in Table 2.

Table 2. Comparison of CalcuRep to Evaluation Criteria

Primary Criteria	*Comment*
1. User Friendliness	Extremely user friendly.
2. Reliable Crack-Growth Life Predictions	No capability in current version. Program offers SIF design criteria.
3. Damage Growth Rate Prediction in Repaired Structure	No capability in current version.
4. Compute Critical Crack Length in Repaired Structure	No capability in current version.
5. Compute Residual Strength in Repaired Structure	No capability in current version.
6. Identify Inspection Requirements for Repair	No capability in current version.
7. Account for Patch Moisture Absorption	Can incorporate through material properties.
8. Account for Thermal Mismatch Between Repair & Structure	Yes.
9. Ability to Handle Complex Geometry's	Can only model center-cracked panels. Includes stiffener effects.
Secondary Criteria	*Comment*
10. Address Multiple Failure Modes	Somewhat.Through design criteria for patch, structure & adhesive.
11. Address Multi-Site Damage	No capability in current version.
12. Predict Crack Initiation Life in Repaired Structure	No capability in current version.
13. Ability to Account for Load Redistribution in Structure	No capability in current version.
14. Ability to Account for Corrosion Damaged Structure	No capability in current version.

5.2.2. Analysis of Benchmark Problems

5.2.2.1. Academic Problem

The original benchmark problem was modified slightly to account for the fact that the CalcuRep code only examines a center through-crack in a skin panel. For this benchmark problem, two center-crack cases were examined; 0.2 inch and 0.5 inch, double those of the benchmark problems. This way the unconservative results, with respect to the geometric correction factors, would be offset by a more severe cracking situation, therefore leading to a more conservative result.

The results of the analyses are shown in Figures 4 and 5.

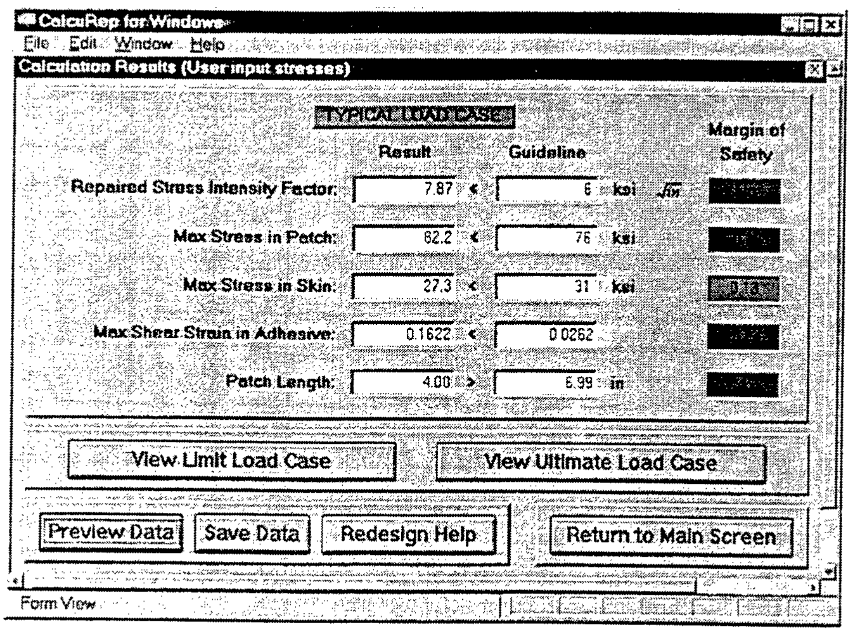

Figure 4. CalcuRep Results for the 0.2 Inch Crack Case (Academic Problem).

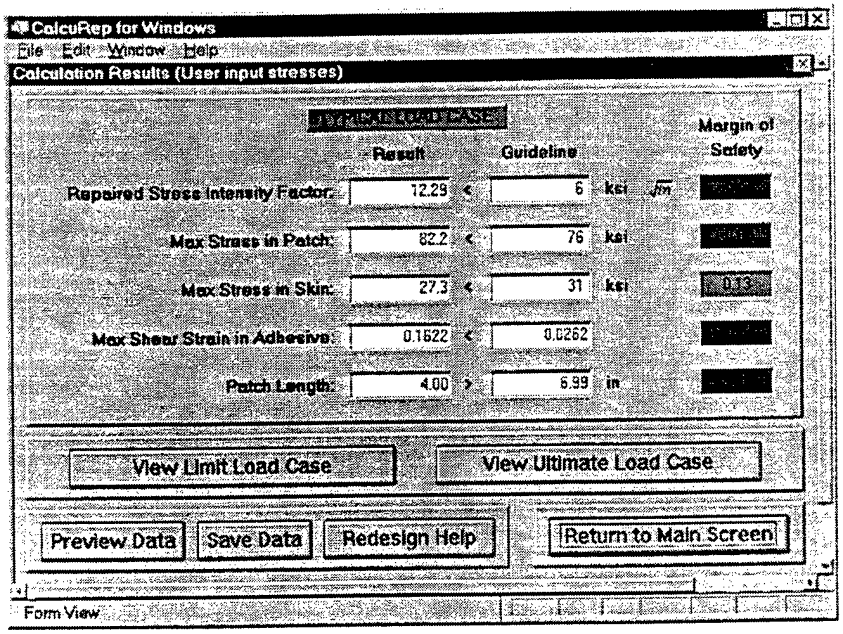

Figure 5. CalcuRep Results for the 0.5 Inch Crack Case (Academic Problem).

CalcuRep allows the user to review the limit and ultimate load cases, but for these benchmark problems only the "Typical" load case was examined. Also, the CalcuRep code offers "Redesign Help" that could have led to a better patch design for this case and is discussed later in the report. In addition, there were some assumptions that had to be made. First, the altitude was assumed to be 1 ft, to represent sea level conditions and reduce any altitude-related thermal effects. Second, the stiffener spacing (parallel and perpendicular to crack) was placed at a maximum of 39.4 inches, to reduce the effects of stiffeners (load attraction) and allow secondary bending effects. This was done to simulate a situation similar to that of an unstiffened panel.

It is immediately obvious that the patch designs are very poor. Since one of the goals of a good compsite repair code is to recognize poor patch designs, CalcuRep does this in a clear, concise manner. This is one of the primary strengths of this code.

5.2.2.2. C-141 Weep-Hole Problem

This problem is beyond the applicability of many, if not all two-dimensional, elasticity-based analysis programs. The weep-hole, seen in Figure 3 has cracking, initially occurring in the riser and propagating into the skin. This cracking scenario was reported in earlier reports [8]. One might try to model this problem using symmetry through the riser, but the contribution of stiffness in the skin cannot be accounted for within measurable limits.

Therefore for this problem, it was decided to model the through crack, once the crack had propagated into the skin. This situation can be seen in Figure 6.

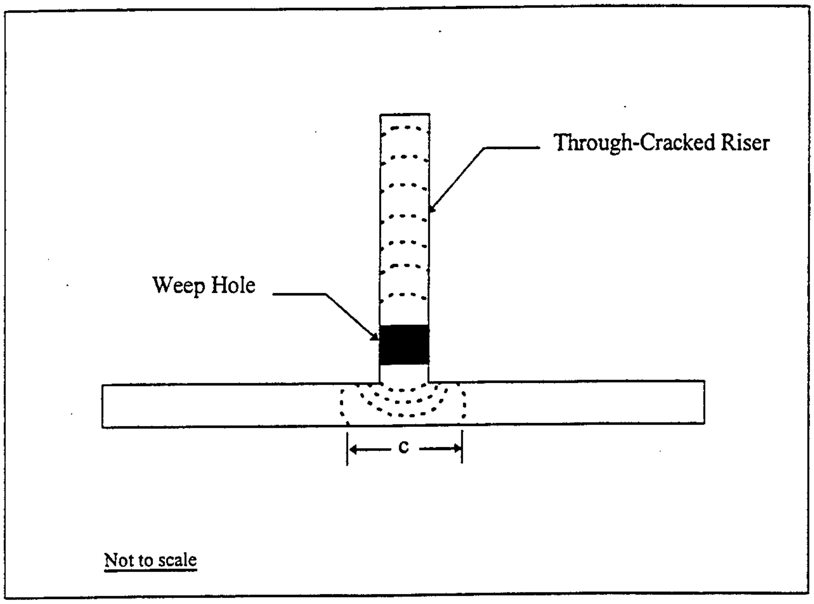

Figure 6. C-141 Weep-Hole Problem Evaluated by CalcuRep.

Since there is no straight-forward way to account for the two patches on either side of the riser, they were neglected. However, given the above situation, the through-crack in the skin will be modeled with an external patch similar to the one used in the reported weep-hole repair. This obviously disregards the increased stiffness and closure effects on the crack, due to the presence of the riser doublers on the risers. It is totally expected that this repair will be determined to be inadequate. This repair design is shown in Figure 7.

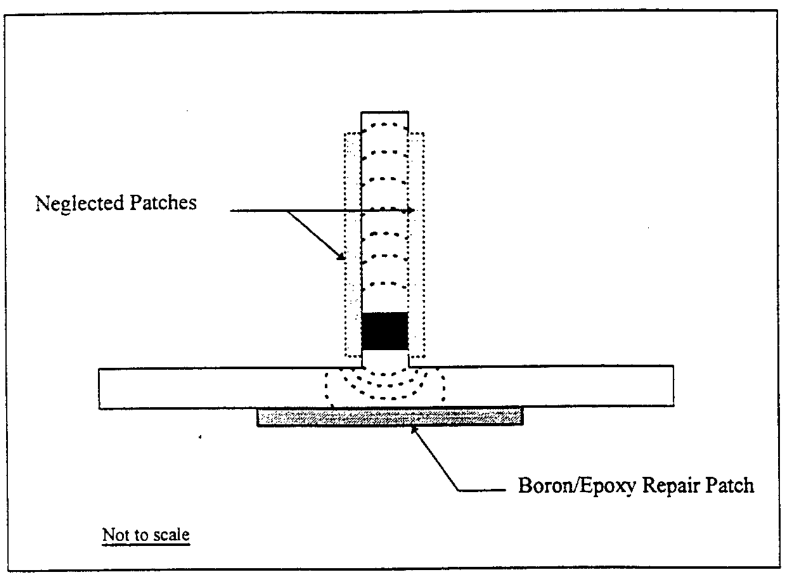

Figure 7. C-141 Weep-Hole Problem Evaluated Using CalcuRep.

The only crack case examined will be for 0.25 inch. The 0.1 inch crack case was excluded because the riser thickness is approximately 0.192 inch. Obviously, any crack propagating through the riser into the skin would be greater than the riser thickness.

The results of the CalcuRep analysis are shown in Figure 8.

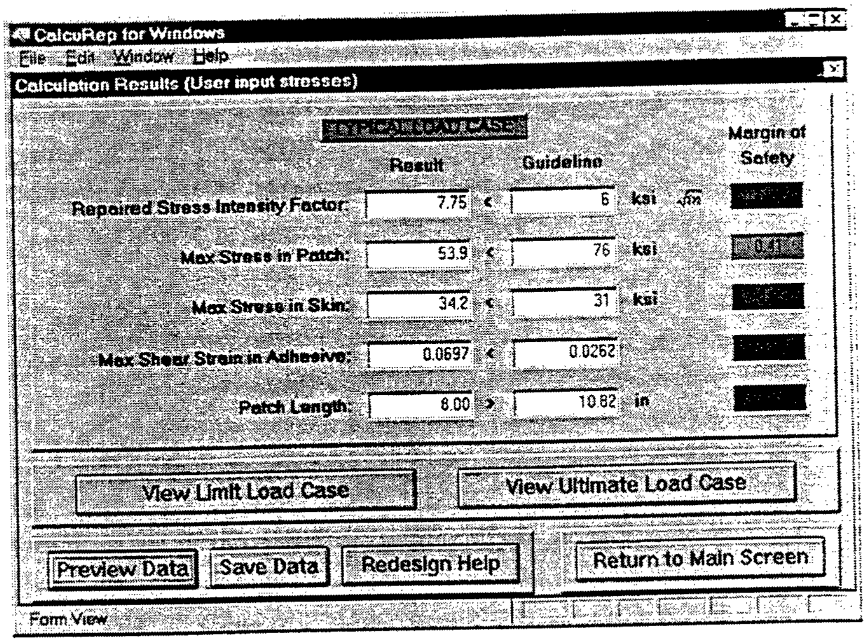

Figure 8. Results of the C-141 Weep-Hole CalcuRep Analysis.

There were several assumptions made in the process of completing the analysis. Some of the assumptions made in this problem were: all 13 plies were 0 degrees, stiffener spacing was ~4.2 inches perpendicular to crack and 39.4 parallel to crack. (These are the maximum and minimum allowable values available in CalcuRep.) The reasoning behind these choices were to artificially simulate the stiffness provided by the stiffeners (and neglected patches) over the crack. This would reduce the secondary bending effects as much as possible in the direction perpendicular to crack propagation. Also, since any spars or stiffeners in the C-141 wing would be quite large, the 39.4 inch value was used parallel to the crack.

There was no attempt to use the "Redesign Help" for this problem, since the situation was an approximate model of a benchmark problem. It is believed that, while a feasible solution could be reached with the CalcuRep code, it would not be a plausible solution for the service aircraft. It is not AS&M's intention to "second guess" the design criteria used in performing a repair as complicated as the C-141 weep-hole problem.

5.2.2.3. T-38 Lower Wing Skin Problem

There was no attempt made to analyze this complex repair situation using the CalcuRep code. This type of problem is better suited to a combination of finite element analysis, fatigue crack initiation, and fatigue crack growth analysis as reported in [9]. The combination of the problem's structural complexity (numerous details), high stresses, load transfer, short critical crack lengths and accessibility for repair (non-standard patching procedures) make it a candidate for more sophisticated analyses techniques.

These comments should not be construed as a criticism of the CalcuRep code. The CalcuRep code states in its Users Guide that it is a "quick and efficient" static strength design tool for the design of composite patch to metal aircraft fuselage structures. The developers of this code do not suggest that this design code be applicable to any and all situations on an aircraft structure. Like all of the other codes evaluated in this study, CalcuRep definitely has its place among a number of tools available for the design of composite patches to metal repair structures.

5.2.3. Discussion

In retrospect, it seems unfair that most of the evaluation criteria are based on dynamic aspects of structural integrity, primarily fracture mechanics. The CalcuRep code incorporates many excellent features of a static strength-based, design/analysis code. In a perfect world, all computer software design/analysis codes should be this easy to install, run and operate. The GUI is simple, direct and for all intents and purposes, "idiot-proof." The CalcuRep code includes a common materials (aircraft, composite, and patch) database that prevents a novice from scrambling to find manuals, reports, in -house data, etc., to obtain this information. The CalcuRep code incorporates a clever results (window) that offers "go-no-go" criteria comparisons (with red and green lights) and "redesign" help. The authors are also very quick to remind users that they should have their designs approved by the proper authorities (manufacturer) if the repairs are not found in the aircraft Structural Repair Manual.

As stated in the CalcuRep User's Manual, this code is most suitable for the design and analysis of repair patches to thin, aircraft fuselage structures. It does not take into consideration through-the-thickness variation of stress intensity factors in the repaired structure. Experimental results have shown that the crack does not grow uniformly in relatively thick ($t > 0.125$ inch) structures repaired with composite patches on one side only. In such structures. the crack-growth is non-uniform through the thickness.

There is however. a question of the ability of the Rose Model to calculate accurately the stress intensity factor under a repair patch in the presence of secondary bending. This has been challenged in previous Air Force research by other investigators [12]. While it is not the intent of this study to question the validity of the theory behind the codes, it would be the neglectful not to mention previous studies available to the authors of this report. Also, the author of this report is not certain if the developers of CalcuRep have incorporated the same bending model that was developed by Rose [13] or a modified version of that model.

In general, there are several limitations to employing static-strength-based design criteria. First, designing to a static strength (plane stress or plane strain) intensity factor, K_{design} is more often than not leading one into a false sense of security. For example, while a guideline of 6 ksi $\sqrt{in}$ is well below plain strain (K_c= ~60-70 ksi $\sqrt{in}$ [10]) or plane stress (K_c= ~56 ksi $\sqrt{in}$ [11]) fracture toughness values for this thickness of 7075-T6 aluminum, a ΔK of 6 ksi $\sqrt{in}$ (for typical cases) can lead to crack growth rates (da/dN) of approximately 1.0-6.0 E-6 in/cycle (@R=0) [10,14]. This crack growth rate is very high for some aircraft structures. Now, it may be true that the developers of this program have based this value K_{design}=6 ksi $\sqrt{in}$ on a fuselage pressure spectrum that experiences this stress level a relatively few amount of times over the structure's design life. In this case, this design guideline might be acceptable. But for many other spectra, such as various transport and fighter wing spectra, this value may not be adequate.

This also leads to the importance of load history, or spectrum effects, on crack-tip stress intensity factors. Depending on the load history the value of stress intensity is always more or less than calculated with a "stress free" crack face. The pure probabilistic aspects of being "in the ballpark" of that value in real aircraft structures is not very realistic. However, as a starting point, looking at a K_{design} value , used in conjunction with a structure's residual strength may be an acceptable starting point.

In addition, a one-dimensional, bonded joint theory does not adequately reflect the two-dimensional nature of shear stresses in a bonded repair patch along the length of a crack. This type of analysis may not be realistic, as some experimental data have shown that the adhesive shear stresses vary along the crack plane [15,16], especially in the presence of a disbond that develops along a growing crack. This behavior is more often present in highly-loaded, patched structures.

The other design criteria (Maximum Patch Stress, Maximum Stress in Skin, Maximum Shear Strain in the Adhesive, Patch Length) in CalcuRep are very conservative and have been based on "rules of thumb" developed over years of crack patching experience. These criteria appear to be both conservative and reasonable values to be used within a static strength-based analysis tool.

It is the opinion of the author of this report that the stress intensity factor components (K_{remote}, $K_{bending}$, $K_{thermal}$) should be reported separately to the user. That way, correction factors (geometry) could be applied to the individual components to account for a larger amount of structural configurations. Also, a user could then compare the individual results with other analysis methodologies on hand (finite element models, closed-form solutions, etc.,) to gain greater confidence with the programs subroutines and eliminate components deemed necessary to the current situation.

5.2.4. Future Directions

There are several enhancements/upgrades planned for the CalcuRep code over the next year. First, the developers plan to model the effects of variable amplitude loading (spectra) on the repaired structure. Second, the design and analytical capabilities will be improved to include repair patches over a greater number of structural configurations. Third, the report function will be enhanced to include graphical representations of user inputs and calculated outputs. Finally, an effort will be made to combine the features of CalcuRep with the AFGROW [11] fatigue crack growth life prediction program. In addition, plans are also being discussed to include the effects of cyclic and preexisting delamination on the structural integrity of a composite repair patch.

For further information, please contact Major Rob Fredell, Department of Engineering Mechanics, USAF Academy, CO.

5.3. COMPAT_3D

The COMPAT_3D program was compared to the evaluation criteria and also to "versions" of the benchmark repair problems. Since the operating version of COMPAT_3D only analyzed part-through cracks, cracks of this configuration were implemented instead of the original through cracks. Also, it was not possible to complete the repaired stress intensity problems, due to the current capabilities and "bugs" in this version of COMPAT_3D. AS&M personnel spent a considerable amount of time trying to operate and debug both PATCHGEN_3D and COMPAT_3D, often with the assistance of the developers. While it was not anticipated by AS&M that the COMPAT_3D program was only a Beta version with respect to repaired stress intensity calculations, it was only possible to obtain unpatched crack stress intensity factors.

5.3.1. Comparison Against Evaluation Criteria

COMPAT_3D was compared against the evaluation criteria and the results are listed in Table 3.

Table 3. Comparison of COMPAT_3D to Evaluation Criteria

Primary Criteria	*Comment*
1. User Friendliness	Not user friendly. Menu driven. Manipulate files with text editors.
2. Reliable Crack-Growth Life Predictions	Yes (limited). Outputs file for use with external FCGR codes.
3. Damage Growth Rate Prediction in Repaired Structure	Externally. Outputs file for use with FCGR codes.
4. Compute Critical Crack Length in Repaired Structure	Yes (limited). Outputs file for use with external FCGR codes.
5. Compute Residual Strength in Repaired Structure	Externally. Outputs file for use with FCGR codes.
6. Identify Inspection Requirements for Repair	Externally. Outputs file for use with FCGR codes.
7. Account for Patch Moisture Absorption	Can incorporate through material properties.
8. Account for Thermal Mismatch Between Repair & Structure	Yes. Through FEAM.
9. Ability to Handle Complex Geometry's	Yes. Through FEAM.
Secondary Criteria	*Comment*
10. Address Multiple Failure Modes	Yes. Can be done externally with FEAM results.
11. Address Multi-Site Damage	No capability in current version.
12. Predict Crack Initiation Life in Repaired Structure	No capability in current version.
13. Ability to Account for Load Redistribution in Structure	Yes. Through FEAM.
14. Ability to Account for Corrosion Damaged Structure	Yes. Through FEAM.

The COMPAT_3D code's potential capabilities match very well with other finite element codes, with the exception of reduced modeling and computation time.

5.3.2. Analysis of Benchmark Problems

5.3.2.1. Academic Problem

This section contains a description of an analysis of the academic problem, presented in Section 4.1.1, using COMPAT_3D. In this case, a three-dimensional quarter-model (to take advantage of symmetry) was developed using PATRAN, and a quarter-elliptical corner crack was introduced at the hole as shown in Figure 9. No analysis was performed with the patch applied. All problem dimensions and material properties are as in Figure 1. The three-dimensional mesh used is shown in Figure 10.

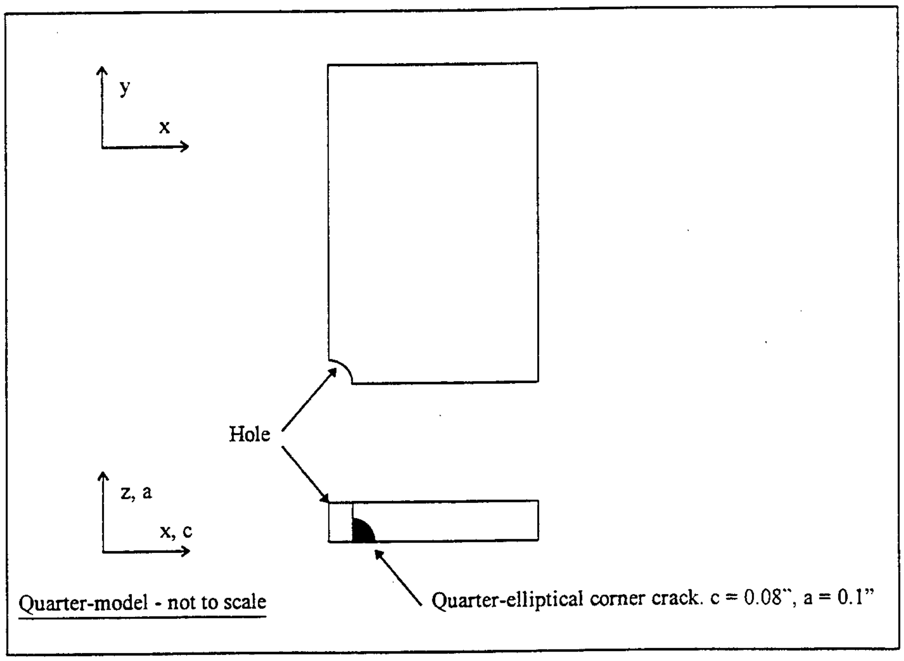

Figure 9. Crack Location & Geometry (Academic Problem, COMPAT_3D).

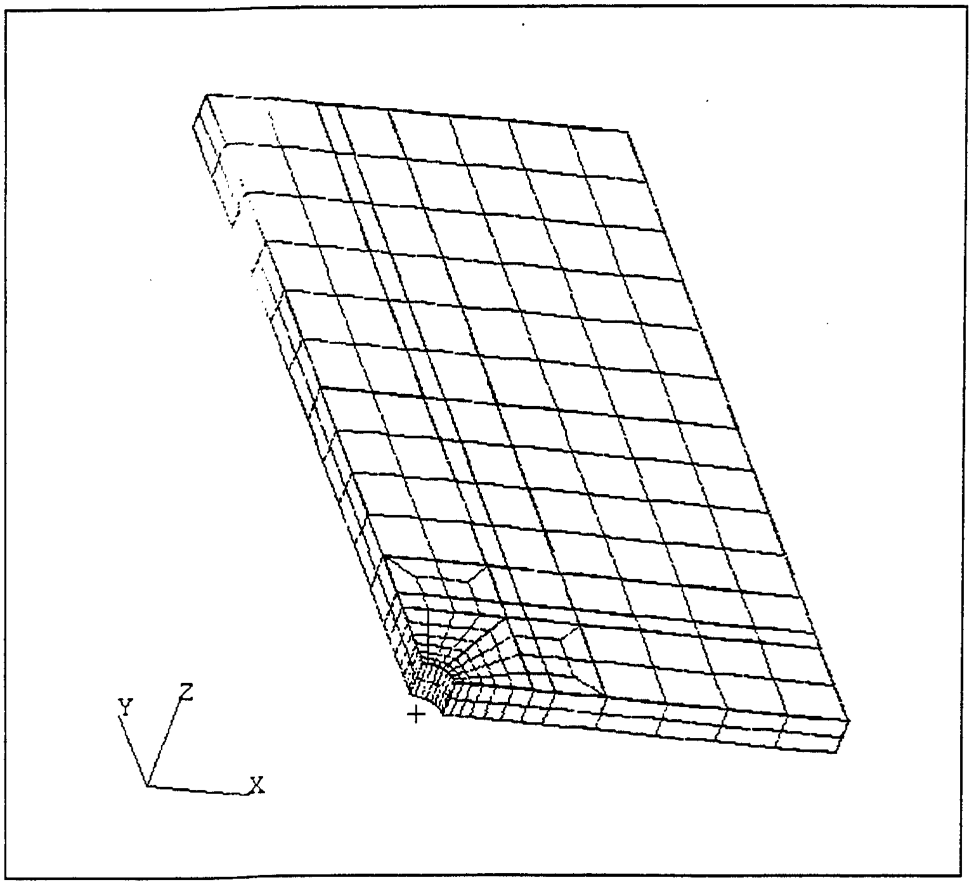

Figure 10. Mesh Used for Academic Problem (COMPAT_3D).

Only one crack configuration (Figure 9) was analyzed. The stress intensity factors obtained were plotted against crack front angle from the major axis and these are shown in Figure 11. Again, no analysis was performed with a patch applied.

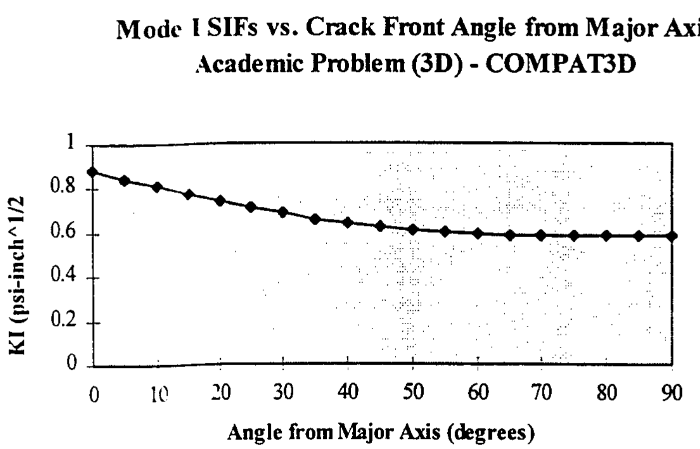

Figure 11. Normalized Stress Intensity Factors (Academic Problem, COMPAT_3D).

No comparisons were made between these results and results found in the literature. It is assumed that the developers of the code have already performed these benchmarks. In any event, stress intensity factor solutions of this type have been most successfully performed in the past using the finite element alternating method or full-blown, 3D finite element analysis.

5.3.2.2. C-141 Weep-Hole Problem

The weep-hole model of Figure 1 was analyzed using COMPAT_3D. Analyses were attempted for four different cases as described below. However, only two of these cases, with no patches applied, were analyzed successfully. Errors were encountered while attempting to solve the cases with patches applied. The program crashed in these cases.

The mesh generator used for these analyses was P3/PATRAN. A solid mesh of the un-patched structure was built first and the first two cases were analyzed with this mesh. Upon creation of the mesh, a NASTRAN Bulk Data File (BDF) was written using PATRAN. This file was then translated, using PATCHGEN_3D, into a format understandable by COMPAT_3D. Data pertaining to crack locations and sizes were also added using PATCHGEN_3D.

The process of creating a mesh and translating it to a COMPAT_3D input file presented many problems to the authors. Many of these problems were found to occur as a result of bugs in the Silicon Graphics (IRIX) version of PATRAN itself, and required the support of MSC personnel to resolve. These bugs will not be discussed in this report, as they are not part of the COMPAT_3D code. However, once these problems were overcome, successful translation of the load and boundary conditions required modifications to the PATCHGEN_3D code. It was found that the code was not equipped to handle certain card sets in the NASTRAN bulk data file. Changes were made to the code and the specific problems faced by the author during translation were eventually addressed.

After the modifications were made, it was possible to analyze unpatched models successfully. However, the code crashed when a patch was applied to the model and analysis was attempted. The model dimensions and material properties are as shown in Figure 2. Figure 12 shows the three-dimensional mesh used in these analyses. Figure 13 shows a detail of the same mesh around the weep-hole.

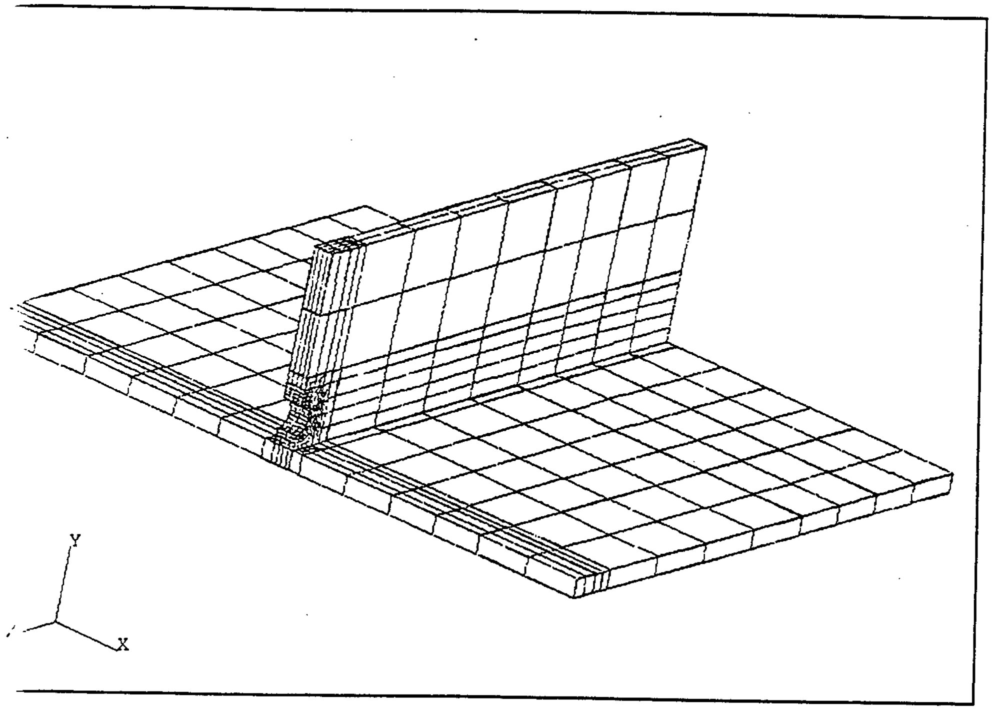

Figure 12. Mesh Used for C-141 Weep-Hole Problem (COMPAT_3D).

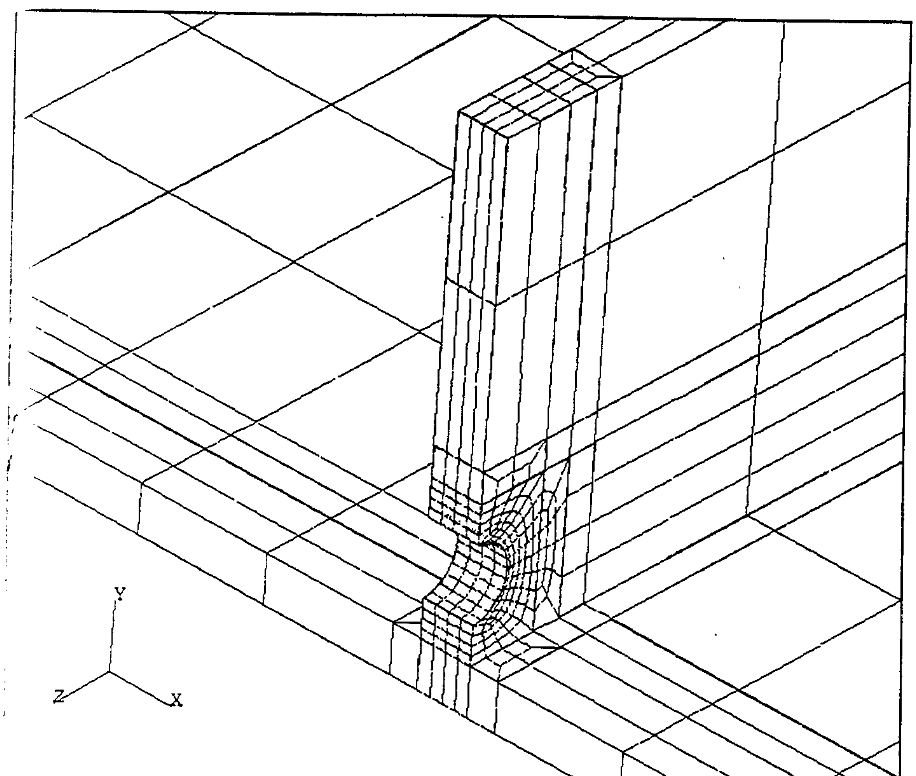

Figure 13. Detail of C-141 Weep-Hole PATRAN Mesh (COMPAT_3D)

A half-model was used and symmetry boundary conditions were applied. A unit far-field stress was applied in the 'z' direction. Two crack configurations were analyzed and stress intensity factors obtained. These are as shown in Figure 14. While COMPAT_3D has the ability to perform constant-amplitude fatigue crack growth analyses, it is not capable of transitioning these cracks into through-cracks, and hence such analyses were not performed.

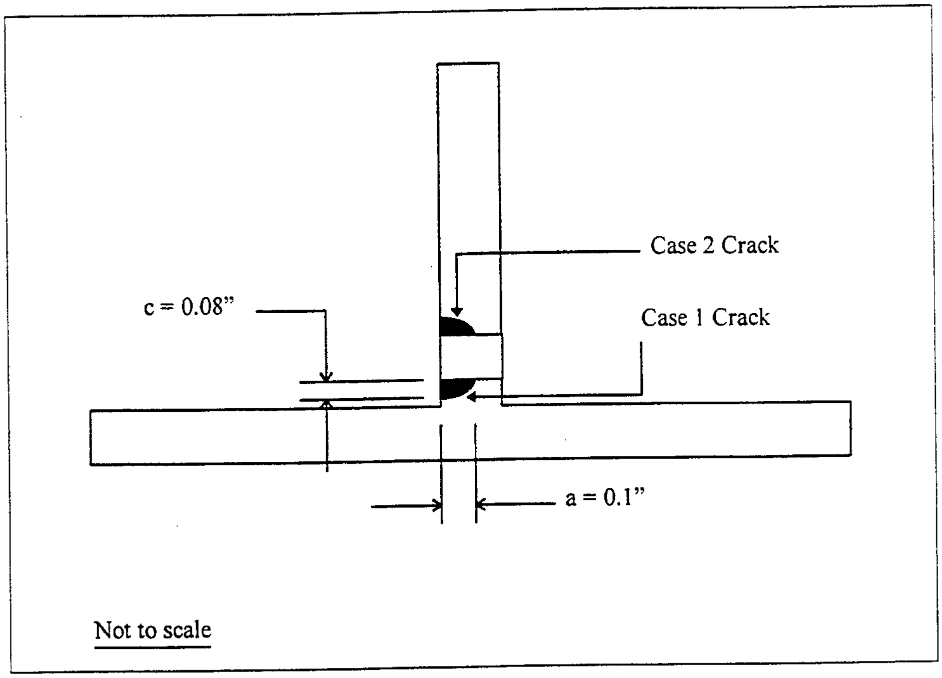

Figure 14. Case 1 & Case 2 Cracks (C-141 Weep-Hole Problem, COMPAT_3D)

Stress intensity factors obtained for the Case 1 and Case 2 cracks (analyzed separately) were plotted as a function of crack front angle from the major axis and are shown in Figure 15. As expected, the values obtained for the Case 1 crack are slightly lower than those obtained for the Case 2 crack due to the stiffening effect of the wing skin.

Attempts to analyze the above cases with the riser patches applied resulted in the code crashing. This error was not resolved during the course of this study.

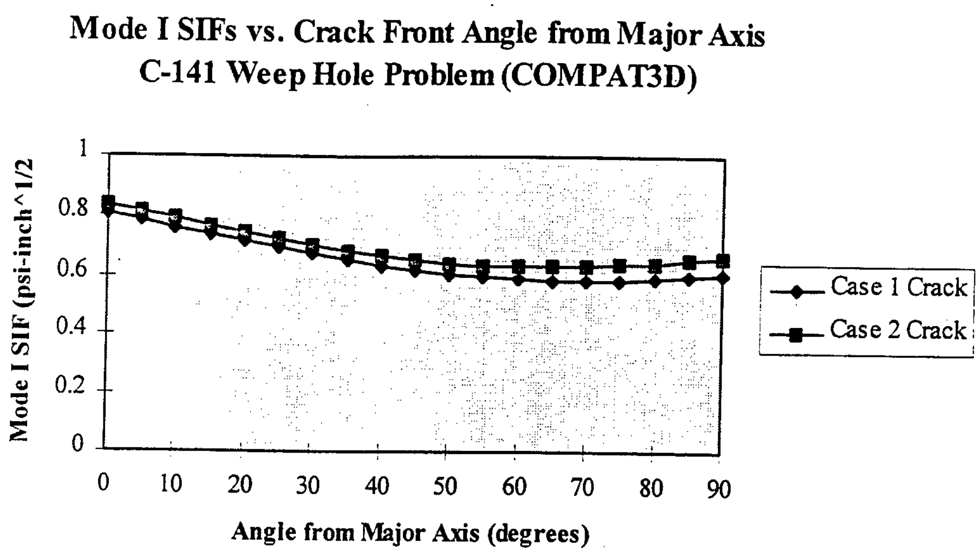

Figure 15. K_I vs. Crack Angle, θ, from Major Axis - C-141 Weep-Hole Problem.

The above results were compared to those previously reported [17], and are in very good agreement. Any minor discrepancies may be attributed to the fact that slightly different geometry dimensions were used for the weep-hole models.

5.3.2.3. T-38 Lower Wing Skin Problem

The T-38 wing skin cracking problem was not addressed with COMPAT_3D, due to time and modeling considerations. It is believed by the authors of this report that modeling this situation is possible with COMPAT_3D. However, since the version of the COMPAT_3D code that was currently available only could analyze the uncracked structure, further evaluation of COMPAT_3Ds ability to evaluate patched SIFs was considered pointless.

It was the purpose of the evaluation to evaluate the current capabilities of the individual programs. The only alternative would be to generate nodal tractions due to the presence of a composite repair patch with an external finite element code, then applying them to the COMPAT_3D finite element model. However, this approach was considered not to be within the scope of this evaluation.

5.3.3. Discussion

As stated in Section 5.1, it was impossible to calculate the repaired stress intensity factors without the existence of an external finite element code to calculate the nodal tractions within the patch area of the model using this version of COMPAT_3D. This was a disappointment, however, it does not trivialize the importance of developing this technology. A finite element alternating method-based code would be an excellent way to examine the stress intensity factors (and surrounding stress fields) of cracked, structural details. This method would be more accurate than two-dimensional finite element models and much quicker than three-dimensional finite element models.

The developers of the COMPAT_3D code have since corrected the reported bugs supplied to AS&M in the original version of the COMPAT_3D software. However, this was done after the evaluation period and funding limitations prevented the completion of the benchmark problems. Also, since that time additional bugs have been uncovered, preventing AS&M employees from successfully performing a complete analysis.

COMPAT_3D also offers both PC and UNIX-based versions allowing potential users to take advantage of their current hardware. When COMPAT_3D becomes fully capable (FEA w/ patch) it could be used to "tune" fatigue crack growth models for damage tolerance analysis or investigate the effects of patching "thick" cracked structures. In many cases, a thick structure may be only accessible by one side, therefore retarding crack growth only on the "patch-side" of the metal structure. Not knowing the thickness effects, especially under the effects of bending, could lead to an extreme overestimate of the fatigue crack-growth life and inspection intervals.

Therefore, it is the belief of this author that if the COMPAT_3D program is to be used routinely, it needs to establish at least one consistent interface package. The authors of this report realize that this is not easy or inexpensive. Also, as was mentioned before, a model that would allow a user to directly apply a patch is necessary for a full repair-code status. In addition, it is important to analyze through-cracks in structures, since these are probably the most commonly detected and repaired. Finally, the effects of thermal stresses on the patched structure would be an important capability.

5.3.4. Future Directions

There are many anticipated enhancements for the COMPAT_3D code over the next year. The code will be enhanced to include through cracks, including multiple cracks in aircraft structure. Second, the code will be enhanced to account for the effects of cold-working (plasticity), interference-fit fasteners (contact elements), and clamp up on the crack-tip stress intensity factors of part elliptical and through cracks. Third, the capability to account for the effects of thermal stresses will be incorporated into the code. Fourth, the ability to examine the effects of pre-existing disbonds on repair patch integrity, and fifth, a capability to analyze the nonlinear behavior of adhesives will be added.

There are also planned enhancements in the areas of pre/post processing and database functionality. These include developing a GUI for both UNIX workstations and PCs. A design capability to determine optimal patch geometry, design, etc., for a given repair situation is planned. A database of common material properties will also be added. A database for common finite element meshes will also be added to enhance modeling of composite patch repairs. The ability to read input files from commercially available finite element pre processors such as IDEAS and PATRAN will be enhanced. Interfaces to fatigue crack growth life prediction programs, such as AFGROW will be developed to transfer normalized stress intensity factors for fatigue crack growth analysis for non-standard geometries.

For further information regarding the COMPAT_3D code contact Dr. Daniel S. Pipkins at Knowledge Systems Inc., 81 East Main St., Forsyth, GA.

5.4. FRANC2D/L

The FRANC2D/L code was compared both to the evaluation criteria and benchmark problems. The FRANC2D/L code was the only code that was capable of solving the benchmark repair problems, even though this was within the context of a two dimensional analysis. In addition, the FRANC2D/L code was the most familiar to AS&M personnel and therefore did not impose a learning curve. However, this fact should not overshadow its flexibility and usefulness as a potential repair code. It was the intent of AS&M to give an unbiased rating to every code in this evaluation.

5.4.1. Comparison Against Evaluation Criteria

FRANC2D/L was compared to the evaluation criteria and the results are shown in Table 4.

Table 4. Comparison of FRANC2D/L to Evaluation Criteria

Primary Criteria	*Comment*
1. User Friendliness	Somewhat user friendly. More suitable to UNIX software users.
2. Reliable Crack-Growth Life Predictions	Yes (limited). Outputs file for use with external FCGR codes.
3. Damage Growth Rate Prediction in Repaired Structure	Externally. Outputs file for use with FCGR codes.
4. Compute Critical Crack Length in Repaired Structure	Externally. Outputs file for use with FCGR codes.
5. Compute Residual Strength in Repaired Structure	Externally. Outputs file for use with FCGR codes.
6. Identify Inspection Requirements for Repair	Externally. Outputs file for use with FCGR codes.
7. Account for Patch Moisture Absorption	Can incorporate through material properties.
8. Account for Thermal Mismatch Between Repair & Structure	No capability in current version.
9. Ability to Handle Complex Geometry's	Somewhat. Limited to 2D FEA (holes, loaded holes)
Secondary Criteria	*Comment*
10. Address Multiple Failure Modes	Yes. Can be done externally with FEAM results.
11. Address Multi-Site Damage	Yes. Can perform fracture analyses w/mulltiple through cracks.
12. Predict Crack Initiation Life in Repaired Structure	No capability in current version.
13. Ability to Account for Load Redistribution in Structure	Yes. Through FEA (2D).
14. Ability to Account for Corrosion Damaged Structure	No. Cannot model reduction in thickness w/ 2D model.

5.4.2. Analysis of Benchmark Problems

5.4.2.1. Academic Problem

This problem illustrates crack growth from a hole in a panel that has been patched with an adhesively bonded sheet. Analyses were performed for various crack lengths for both the patched and unpatched cases. The specimen configuration and dimensions are shown in Figure 1.

A half model was used to take advantage of symmetry (about a line in the y-direction passing through the center of the hole). The mesh used for the specimen is shown in Figure 16 and the mesh used for the patch is shown in Figure 17. Two sets of analyses were performed, one with only the plate and the other with the plate and the patch together. A unit far-field stress was used in both cases. In each set, an initial crack length of 0.1 inch was used, and analyses were performed at the initial length and at increments of 0.1 inch until the crack length was 1.6 inch in each case. Figure 18 shows the mesh at a crack length of 0.5 inch.

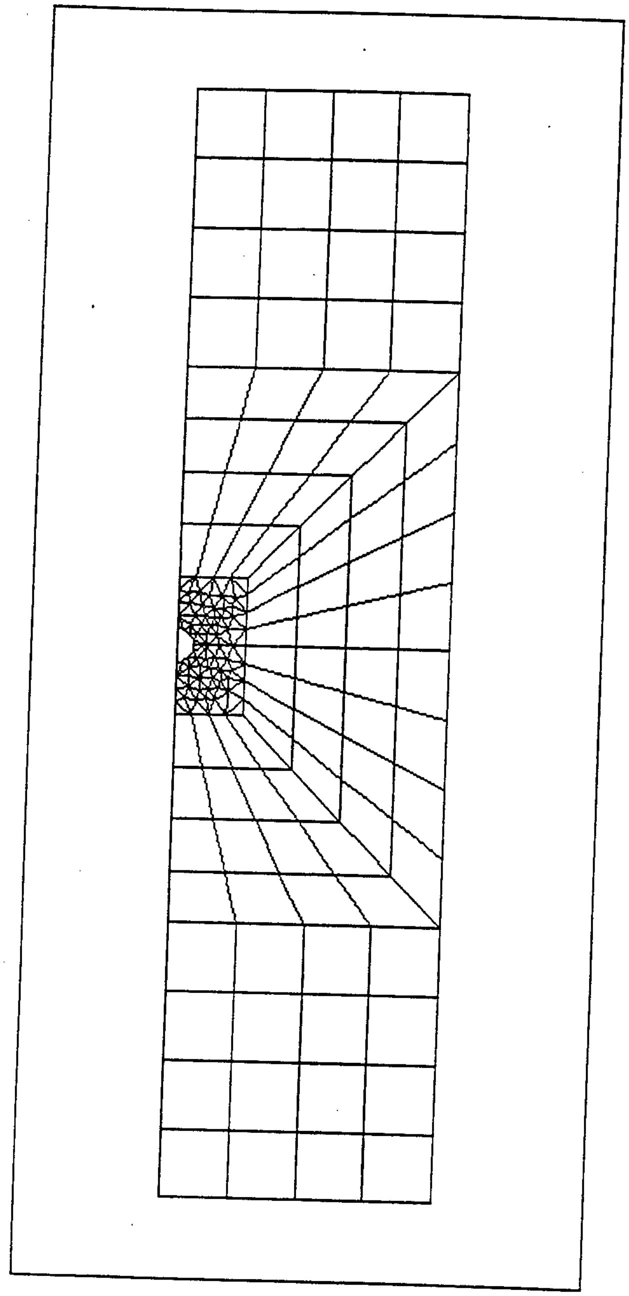

Figure 16. Mesh Used for Layer 2 Plate (Academic Problem, FRANC2D/L).

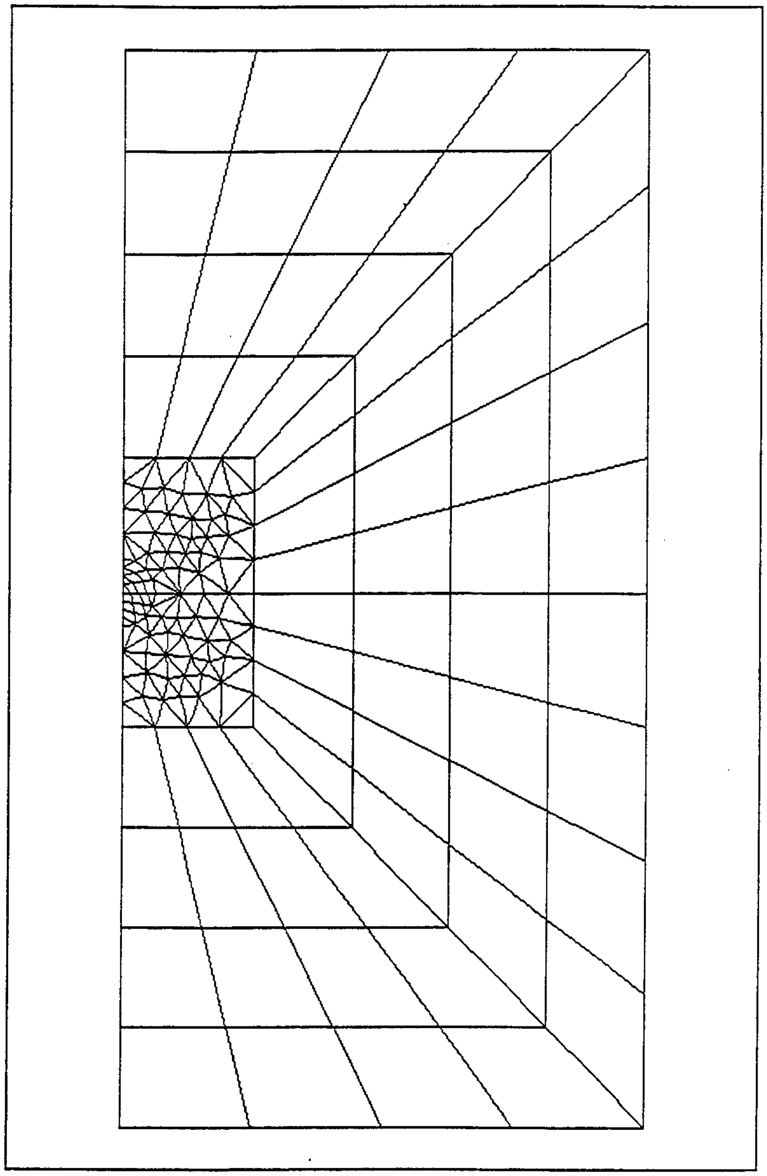

Figure 17. Mesh Used for Layer 1 Patch (Academic Problem, FRANC2D/L)

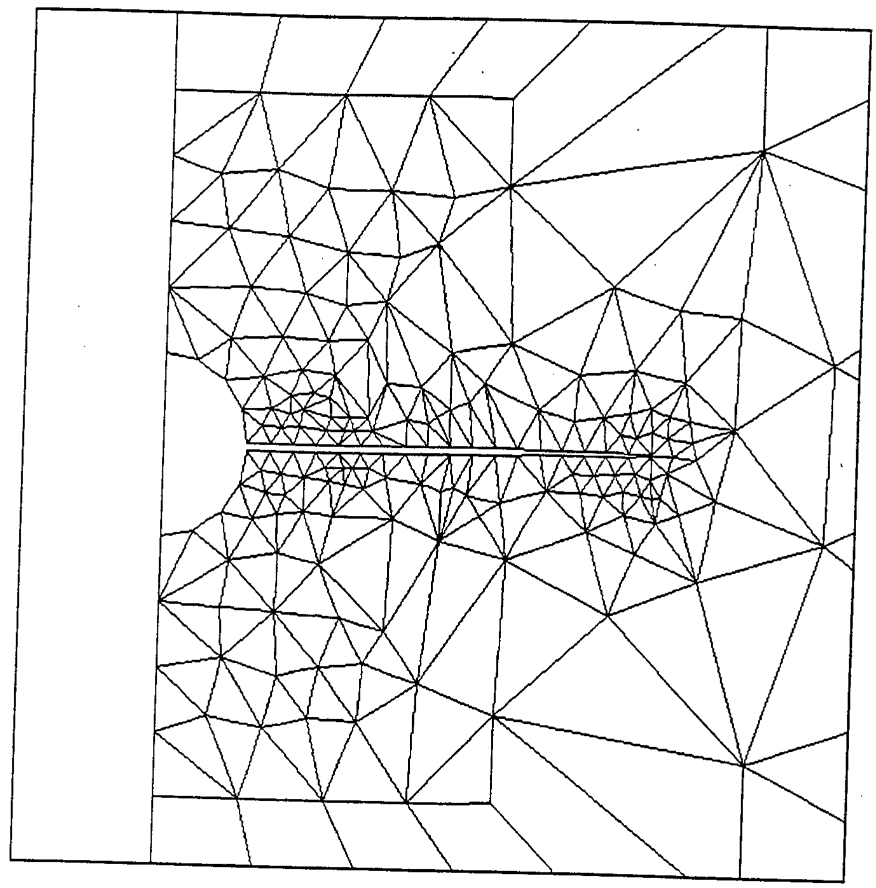

Figure 18. Layer 2 Crack Length of 0.5 Inch (Academic Problem, FRANC2D/L)

Stress intensity factors were saved at each crack increment for both the patched and unpatched model. Plots of the K_I values against crack length are shown in Figure 19. The normalized K_I values are shown in Figure 20.

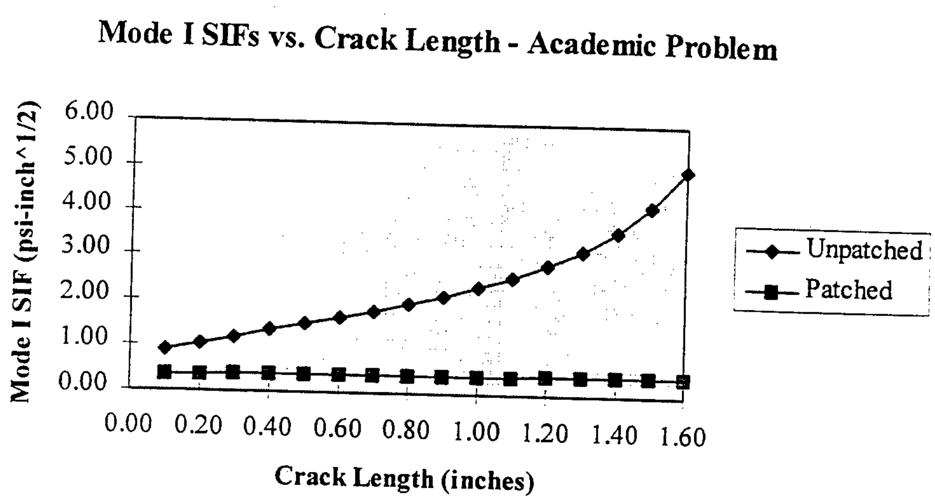

Figure 19. K_I vs. Crack Length (Academic Problem, FRANC2D/L)

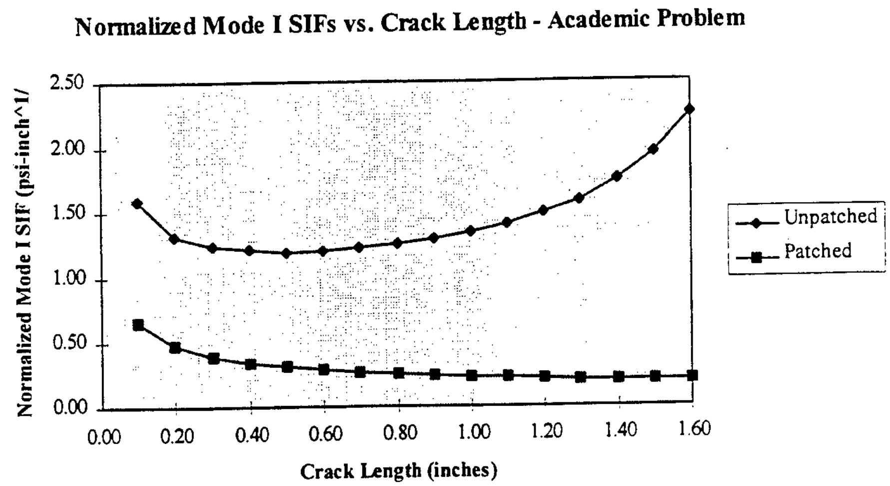

Figure 20. Normalized K_I vs. Crack Length (Academic Problem, FRANC2D/L)

AS&M personnel have previously demonstrated the accuracy of the FRANC2D/L code in accurately determining normalized stress intensity factors [18]. However, with the absence of test data, the ability to measure the accuracy of the repaired normalized stress intensity factors is premature.

5.4.2.2. C-141 Weep-Hole Problem

Several different approaches have been taken to repair the fatigue cracks initiating at weep-holes located in the risers emanating from the lower wing surface panels on C-141 aircraft. The material in this section describes a set of analyses of this problem using FRANC2D/L.

This analysis presented a challenge because it was an attempt to solve a three-dimensional problem using a code that has capabilities restricted to two-dimensional analyses. A description of the actual problem and the simplifying assumptions made to reduce the problem to a two-dimensional one follow. The problem geometry and material properties are shown in Figure 21.

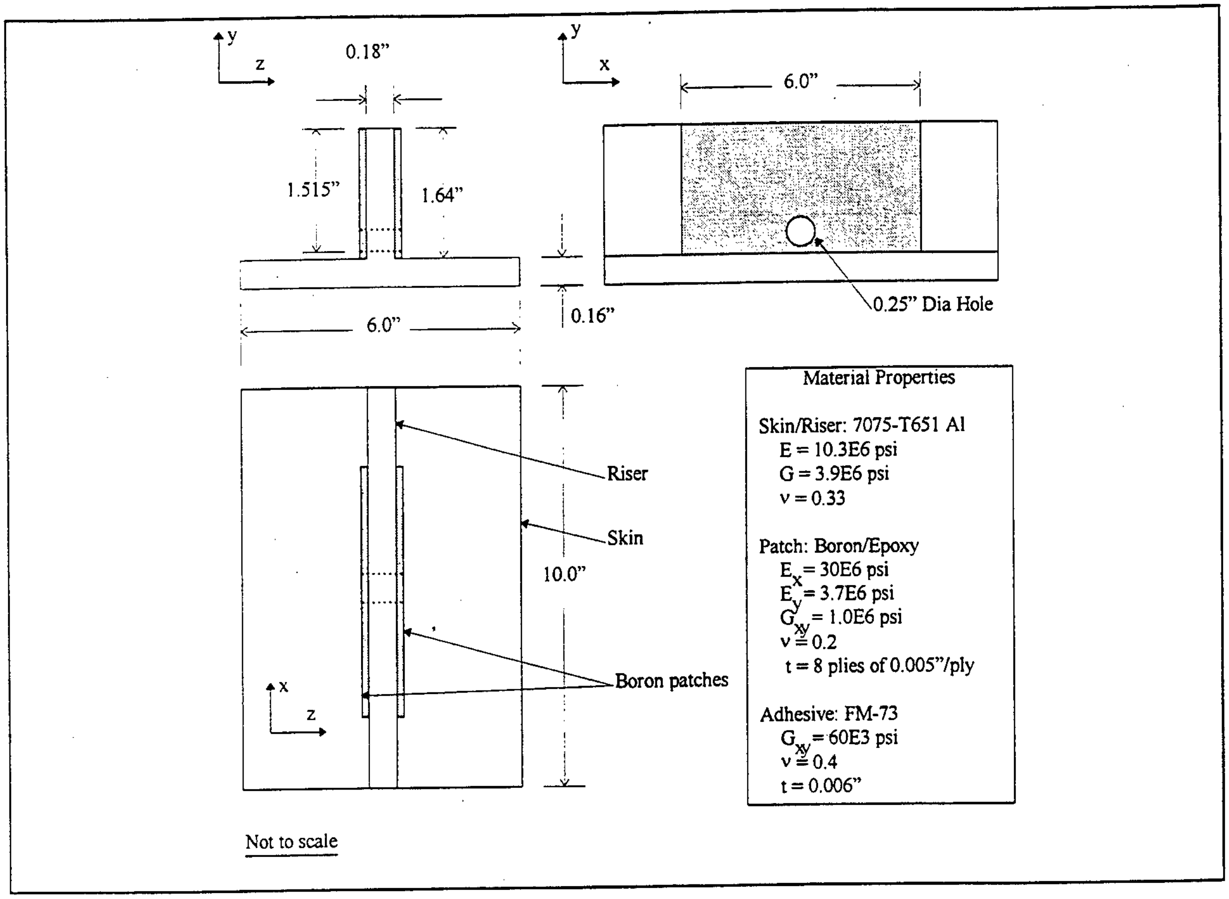

Figure 21. C-141 Weep-Hole Crack Problem (FRANC2D/L)

The three-dimensional geometry shown in Figure 21 was modeled using a two-dimensional mesh lying along the x-y plane. The different thicknesses of the skin and the riser were accounted for by specifying different element thicknesses in the respective regions.

The two primary crack locations were at the upper and lower surfaces of the weep-hole, and, due to the orientation of the risers with respect to the wing, tensile loads are predominantly in the x direction. Cracks would typically initiate as corner cracks at either the upper or lower surface of the weep-hole and then progress through the thickness of the riser to become through cracks. The cracks analyzed using FRANC2D/L were all through-the-thickness cracks as the code does not have the capability to model part through flaws.

Analyses were performed with 0.1 inch cracks at each crack location for the unpatched structure (single layer problem). These analyses were then repeated with both patches applied (three layers). The mesh used for the skin /riser is shown in Figure 22.

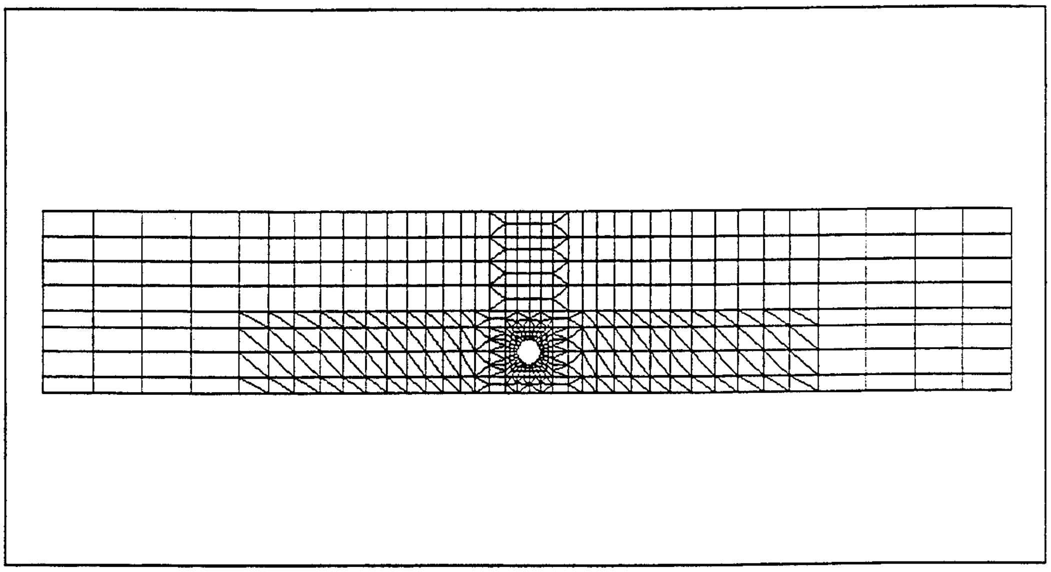

Figure 22. Mesh Used for C-141 Wing Skin/Riser (FRANC2D/L)

The following cases were analyzed:

Case 1: A 0.1 inch crack was introduced between the hole and the skin. A unit far field stress was applied in the x direction. The model was not constrained in any way except the minimum fixity conditions needed to prevent rigid body motion. This analysis yielded a normalized stress intensity factor of 0.9486.

Case 2: A 0.1 inch crack was introduced in the riser above the hole instead of below it. The normalized stress intensity factor obtained in this case was 1.3404. Figure 23 shows a detail of the crack.

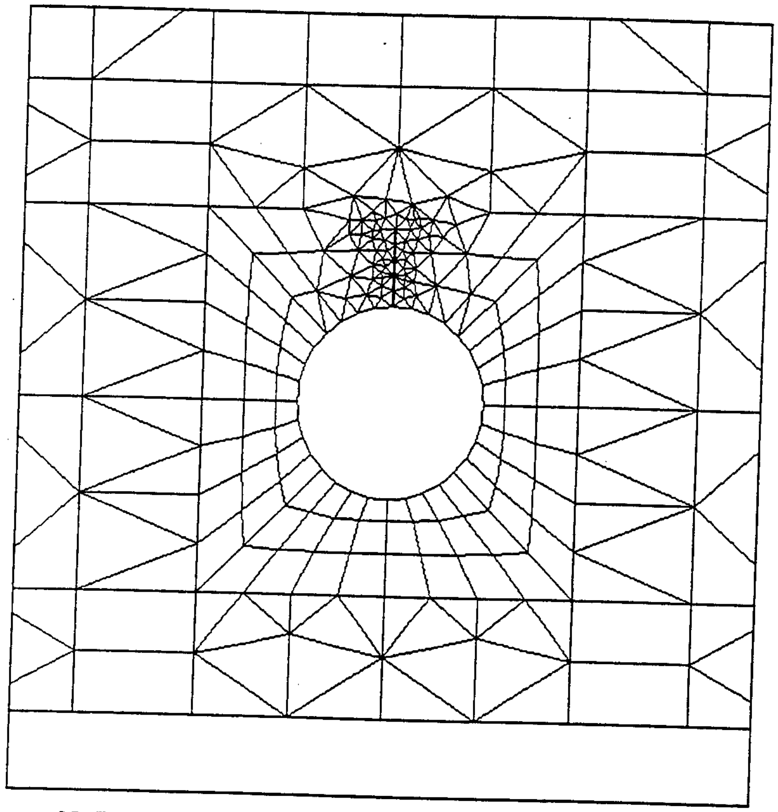

Figure 23. Detail of 'Case 2' Crack at C-141 Weep-Hole (FRANC2D/L)

Case 3: In this case, both patches were applied, making it a three-layer problem. The crack configuration and loading and boundary conditions were the same as in Case 1. A normalized stress intensity factor of 0.7625 was obtained. This is a reduction of 19.6% from Case 1. Figure 24 shows the mesh used for either patch.

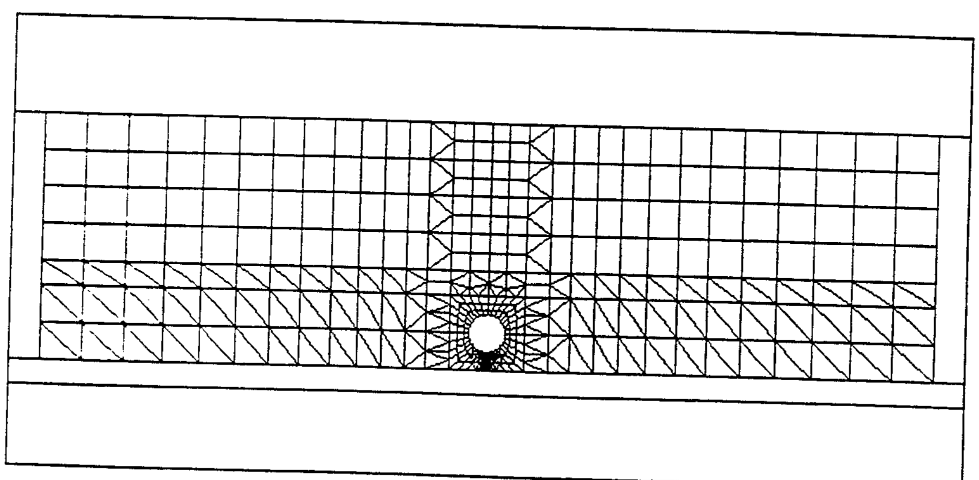

Figure 24. Mesh Used for C-141 Riser Patch (FRANC2D/L)

Case 4: The model used in Case 2 was analyzed after applying the riser patches. The normalized stress intensity factor dropped to 0.9360, a 30.2% reduction.

A summary of the results of the C-141 Weep-Hole Problem is shown in Table 5.

Table 5. Results of the C-141 Weep-Hole Benchmark (FRANC2D/L)

	Normalized SIF $(K_I/\sigma\sqrt{\pi a})$	*% Reduction*
Case 1 0.1" Crack Below Hole	0.9486	-
Case 3 0.1" Crack Below Hole (w/Repair Patches)	0.7625	19.6
Case 2 0.1 " Crack Above Hole	1.3404	-
Case 4 0.1" Crack Above Hole (w/Repair Patches)	0.9360	30.2

5.4.2.3. T-38 Problem

This set of analyses was an attempt to study the applicability of FRANC2D/L to a more complex cracking scenario than the academic problem presented earlier in this chapter. A description of the problem, along with the approach taken to modeling and analysis, is presented in this section.

Cracks were observed in the T-38 lower wing skin near the 44-percent spar between WS 72.25 and WS 76.70 at the Panel D attachment holes. A small section of the wing skin and 'D' panel were chosen for analysis using FRANC2D/L. A more detailed analysis is beyond the scope of this study. The analysis model is shown in Figure 25.

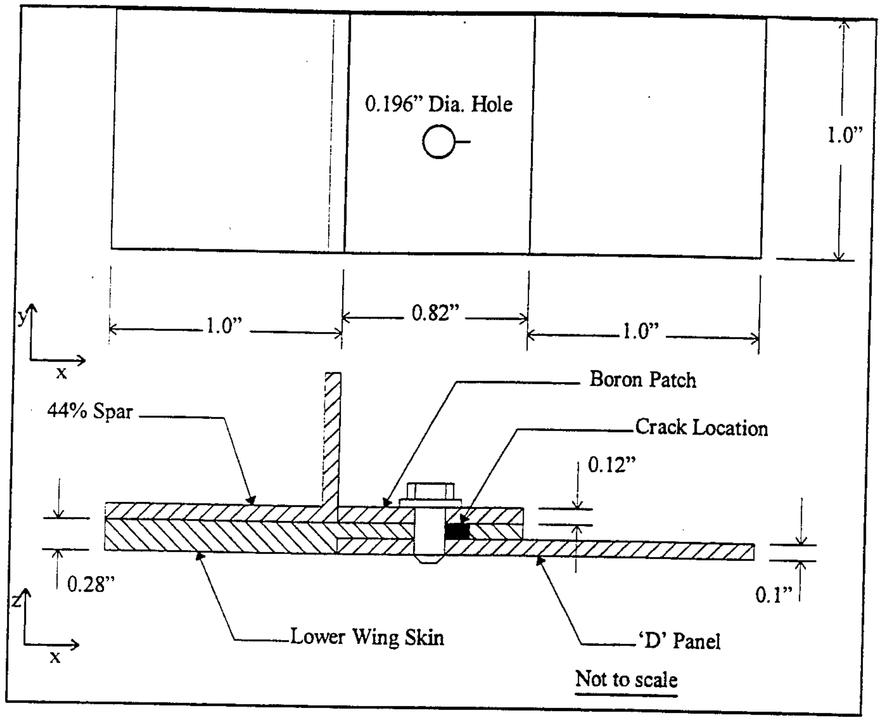

Figure 25. T-38 Lower Wing Skin Problem Geometry

The material properties of the skin, spar, 'D' panel, boron patch and adhesive are as follows:

Skin,	7075-T73 Aluminum
Spar &	E= 10.3E6 psi
D' Panel:	G= 3.9E6 psi
	ν= 0.33
Patch:	Boron/Epoxy
	Ex= 22.0E6 psi
	Ey=13.0E6 psi
	Gxy=2.0E6 psi
	ν= 0.5
	t= 0.12 " (24 Plies @ 0.005"/ply)
Adhesive:	FM-73
	Gxy= 60.0E3 psi
	t= 0.006" (2 layers of .003")

Six different cases were analyzed, the crack length used in all cases being 0.1 inch (no propagation studies were performed). In all cases, the spar was not modeled explicitly, but was considered integral with the portion of the skin that overlaps the spar. The thicknesses of the spar and the skin were added together to obtain an effective thickness in that region of the skin.

The loading and boundary conditions used in these cases, and SIF values obtained, are outlined below.

Case 1: In this analysis, only the skin (with thickness of spar added on in the appropriate region) was considered (one layer problem). A 0.1 inch crack was introduced at the hole and a unit far-field stress was applied in the y direction. A normalized stress intensity factor of 1.457 was obtained. The mesh representing the skin is shown in Figure 26.

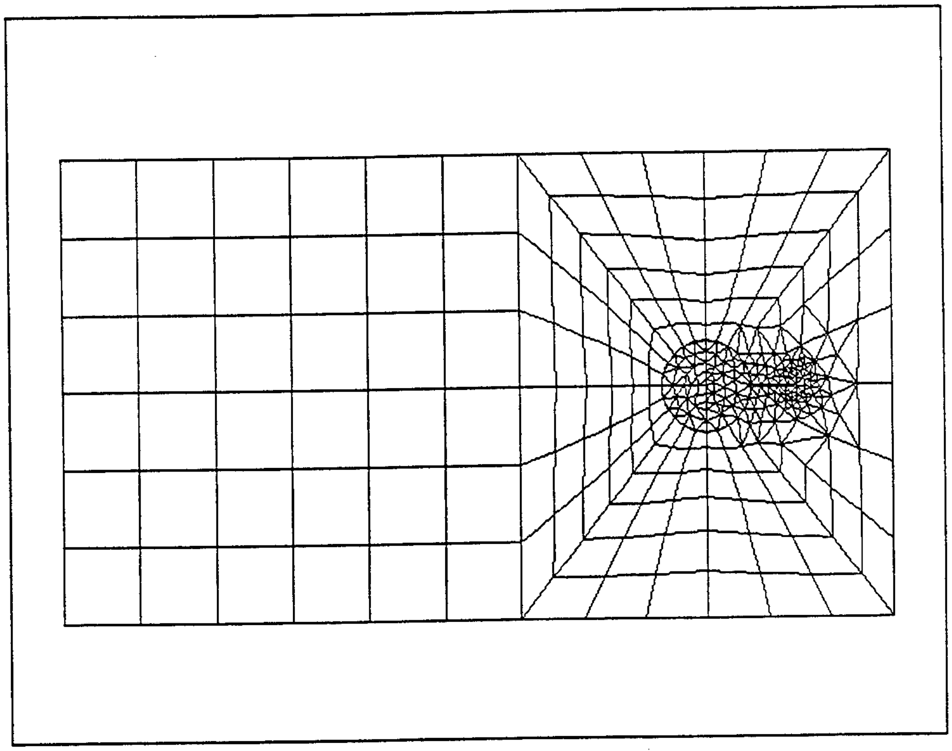

Figure 26. Mesh Used for T-38 Lower Wing Skin (FRANC2D/L).

In the mesh above, the rivet passing through the hole is modeled as a two-dimensional, circular disk, with a ring of interface elements around it. For this particular analysis, the rivet could have been omitted altogether as it does not transfer any loads to the skin. However, since the same mesh was also used in subsequent analyses, the rivet was retained.

Case 2: This case was similar to the previous case, but with the addition of the patch. The patch was modeled as layer 1 and the skin was layer 2. The rivet passing through the hole in the patch was modeled as a disk of the same thickness as the patch and was connected to the rivet elements in layer 2 by adhesive elements of stiffness equivalent to that of the rivet. The combination of these two disks and the adhesive elements behaves like a single rivet connecting both layers.

Adhesive elements were also applied between the patch elements and the skin elements. A crack identical to the one used in Case 1 was introduced at a point on the circumference of the hole furthest away from the spar. No loads or fixity boundary conditions were applied to the patch. The mesh used for the patch is shown in Figure 27.

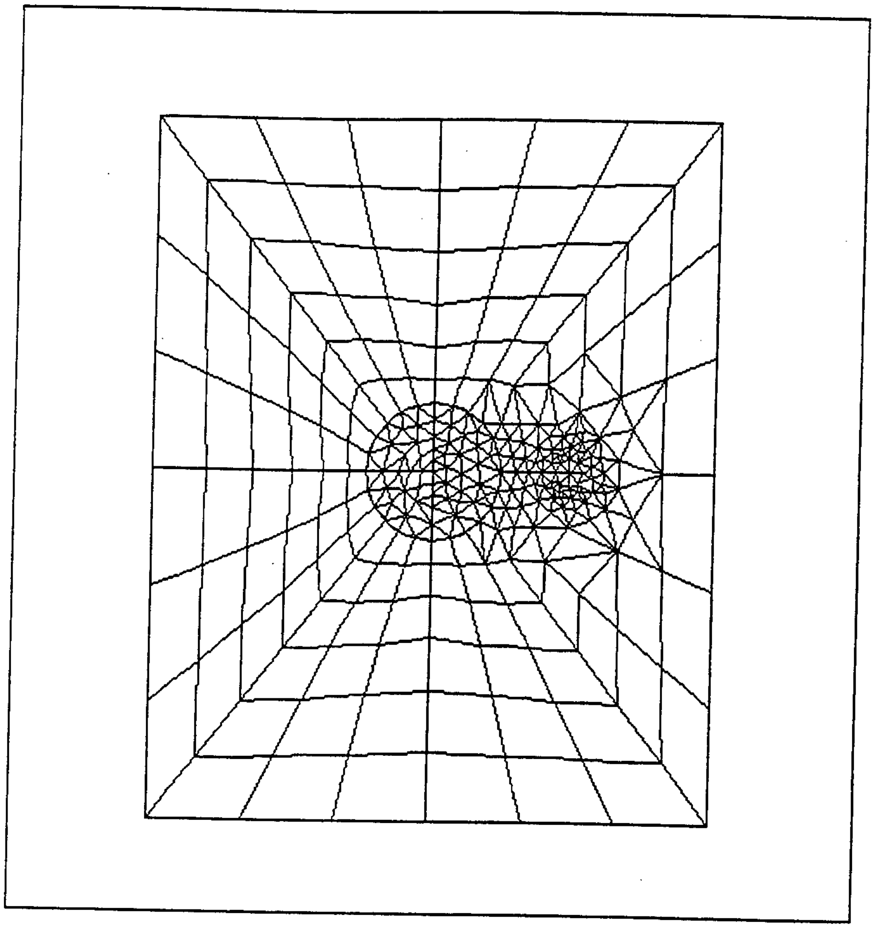

Figure 27. Mesh Used for T-38 Lower Wing Skin Patch (FRANC2D/L).

A normalized stress intensity factor of 0.8348 was obtained in this case, representing a 42.7% reduction from Case 1.

Case 3: In this case, the skin, the 'D' panel and the rivet connecting them were combined in a two-layer model. The rivet provided the only connectivity between the two layers and no adhesive elements were introduced between the skin and the 'D' panel. The rivet was modeled in the same manner as in Case 2. The mesh used to represent the 'D' panel is shown in Figure 28.

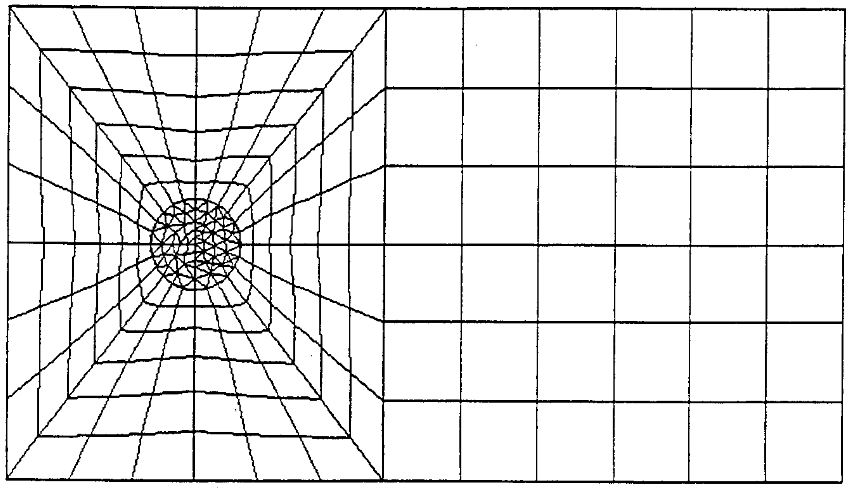

Figure 28. Mesh Used for T-38 Lower Wing 'D' Panel (FRANC2D/L).

Both the skin and the 'D' panel were fixed at the (y = 0) edge and only the 'D' panel was loaded with a unit far-field stress in the y direction. Thus, the only loads transferred to the skin were through the rivet. A normalized stress intensity factor of 0.0225 was obtained in this case.

Case 4: The analysis from Case 3 was repeated here, with the addition of the patch on the skin as in case 2. In this case, the normalized stress intensity factor obtained was 0.0139, a 38.2% reduction from Case 3.

Case 5: This case was similar to Case 3, but with the 'D' panel free at the (y = 0) edge, thereby transferring more of the load to the skin. A normalized stress intensity factor of 2.161 was obtained.

Case 6: Case 5 was repeated with the skin patch included in the analysis. The normalized stress intensity factor was found to drop to 1.5, representing a 30.6% reduction from the previous case.

A summary of the results for the T-38 Benchmark can be found in Table 6.

Table 6. Results of the T-38 Benchmark (FRANC2D/L)

	Normalized SIF $(K_I/\sigma\sqrt{\pi a})$	*% Reduction*
Case 1 0.1" Crack in Skin (Skin+Spar+Rivet)	1.4570	-
Case 2 0.1" Crack in Skin (Skin+Spar+Rivet) (w/Repair Patch)	0.8348	42.7
Case 3 0.1" Crack in Skin (Skin+'D' Panel+Rivet)	0.0225	-
Case 4 0.1" Crack in Skin (Skin+'D' Panel+Rivet) (w/Repair Patch)	0.0139	38.2
Case 5 0.1 " Crack in Skin (Skin+'D' Panel+Rivet) (Free Edge @ Y=0)	2.1610	-
Case 6 0.1 " Crack in Skin (Skin+'D' Panel+Rivet) (Free Edge @ Y=0) (w/Repair Patch)	1.5000	30.6

5.4.3. Discussion

The following paragraphs list some of the special features of FRANC2D/L that make it a useful tool for modeling fracture in two-dimensional layered structures.

The user interface is designed so that only one layer is active (visible on the display) at any time. All preprocessing and postprocessing operations are effective only on the current active layer. The user may switch to a different layer when desired, bringing the selected layer into view on the display and making it the active layer.

FRANC2D/L offers a menu-driven graphical user interface that contributes to the flexibility and ease-of-use of the code. All preprocessing and postprocessing tasks, with the exception of mesh generation, are performed interactively within the program. The accompanying CASCA program is well-suited to creating FRANC2D/L meshes, its interface being similar in functionality and appearance to that of FRANC2D/L. Graphical representation of the mesh and other input parameters facilitate model creation, and different model components may be displayed or hidden from view as desired. In addition, both CASCA and FRANC2D/L allow the user to save a model at any point, and restart the program with that model at a subsequent time.

The user may interactively introduce a crack in any of the layers by either keying in the coordinates of the crack mouth and tip or by using a pointing device. The crack face does not have to lie along element edges but may follow any arbitrary trajectory across the mesh. The software automatically remeshes the model in the vicinity of the crack face to represent the new geometry. In cases where the crack trajectory is not a straight line, it must be broken down into several straight segments that approximate the actual trajectory. The software has the capability to model both edge cracks and internal cracks. While the remeshing process is automatic, the code allows the user to control the size and locations of the newly created elements. The user may perform an analysis at any crack length, perform any desired post-processing, and then increment the crack again. The direction and magnitude of the new crack increment may be decided based on the results of the previous analysis.

Alternatively, the user may opt to propagate the crack automatically. This method requires a number of crack increments (analysis steps) and an initial increment value to be specified. The code first grows the crack by the increment value in a direction based on the current mode I and mode II stress intensity factor values. A new analysis is automatically performed after each crack increment and the new growth direction and crack increment are determined based on the new K_I and K_{II} values and the initial increment. This process is repeated until the specified number of analysis steps is completed. This feature is of great value to the analyst as it eliminates the need to create separate meshes for each crack configuration. The user models only the uncracked structure, and introduction of cracks using FRANC2D/L is a relatively simple process.

As mentioned previously, FRANC2D/L offers the capability to model layered structures such as lap joints and bonded repairs. Adjacent layers can be connected by rivet elements, adhesive elements, or a combination of both. Cracks may exist in multiple layers simultaneously. Adhesive and rivet elements are created using FRANC2D/L's preprocessing functions and not at the mesh generation stage. This is accomplished interactively by selecting the appropriate material properties and then specifying the locations of those elements.

Interface (or gap) elements are used to model contact between surfaces. The user may specify a nonlinear relationship between surface tractions and the relative displacements of the surfaces. The surface tractions are integrated to give equivalent nodal loads. The nodal loads are then included during solution.

One common use for these elements is to model contact between a rivet and a hole, where the user may need to model the rivet in greater detail than afforded by the available two-node rivet element. In this case, the rivet may be modeled as a circular disk and interface elements introduced in a ring running circumferentially around the rivet. By controlling the material properties of the interface elements, the user may specify the type of fit between the rivet and the hole (i.e., interference, exact, or clearance).

Being a two-dimensional finite element method, FRANC2D/L cannot analyze the through-the-thickness variation in a fatigue crack for thicker structures ($t > 0.125$ inches) with repair patches on one side only. It is completely up to the user to understand the FRANC2D/L code's limitations with respect to modeling repair patch problems. This could be a drawback for less sophisticated users.

Also, FRANC2D/L has options to do crack growth analysis under constant amplitude loading. The crack growth analysis is performed with the Paris model. This type of crack growth model is very simplistic and only accurately reflects limited regions of the crack growth curve. In reality, constant amplitude fatigue crack growth analysis is only of limited value, since actual structure experiences flight-by-flight loading. One may be best served by outputting normalized stress intensity factors from FRANC2D/L and importing them into more capable fatigue crack growth programs.

5.4.4. Future Directions

Currently the FRANC2D/L code is being enhanced to perform elastic-plastic finite element analysis. For further details contact Dr. Daniel Swenson, or Mr. Mark James, Department of Mechanical Engineering, Kansas State University, Manhattan, Kansas.

5.5. RAPID

Because the current version of RAPID is designed to assist with the design of bolted metal to metal repairs, it was not evaluated against the benchmark problems. However RAPID was rated against the evaluation criteria and discussed in general terms.

5.5.1. Comparison Against Evaluation Criteria

The RAPID code was compared to the evaluation criteria and the results are shown in Table 7.

Table 7. Comparison of RAPID to Evaluation Criteria

Primary Criteria	***Comment***
1. User Friendliness	Very user friendly. Excellent GUI.
2. Reliable Crack-Growth Life Predictions	Yes(limited). Performs const. amp. FCGR analysis w/Walker mthd.
3. Damage Growth Rate Prediction in Repaired Structure	Yes. Damage limited to fatigue cracking
4. Compute Critical Crack Length in Repaired Structure	Yes.
5. Compute Residual Strength in Repaired Structure	Yes. Plots residual strength vs crack length.
6. Identify Inspection Requirements for Repair	Yes.
7. Account for Patch Moisture Absorption	N/A
8. Account for Thermal Mismatch Between Repair & Structure	N/A
9. Ability to Handle Complex Geometry's	Repairs limited to square cutouts in restricted areas of fuselage.
Secondary Criteria	***Comment***
10. Address Multiple Failure Modes	N/A
11. Address Multi-Site Damage	Yes. Can perform fracture analyses w/two through cracks.
12. Predict Crack Initiation Life in Repaired Structure	No capability in current version.
13. Ability to Account for Load Redistribution in Structure	No capability in current version.
14. Ability to Account for Corrosion Damaged Structure	No. Unless repair protocol includes "cutting out" of corroded area.

5.5.2. Discussion

By examining the evaluation criteria, it could be concluded that the RAPID code best models the ideal repair/analysis code by our standards. However, the RAPID code (as most 2D codes) suffers in the respect that many critical repairs are exposed to 3D effects. Therefore, a code such as RAPID, would be excellent for a "first shot" design/analysis of bolted repairs in less critical fuselage areas. Fuselage areas that are subjected to stress raisers (doors, windows), lap joints, excessively high loads (wing carry through structure), etc., are currently off limits. Also, its applicability to wing structures has yet to be demonstrated by the developers. But the RAPID code does an excellent job of informing its users of its limitations.

There are several issues regarding the analytical capabilities of the RAPID program. The static analysis portion of this version (1.1) is based on a "strip" lap joint analysis to determine the critical fastener load, rather than the actual repair area. The current analysis assumes that all fasteners in the repair area carry the same load. This assumption then assumes the same fastener flexibility for all fasteners in the repair area and does not correlate well with experimental results [19]. Since the fastener flexibility has significant influence on the load distribution, simplified assumptions, such as this, may cause the predicted stresses to be in error by as much as 25-30%.

The analysis used to obtain stress intensity factors for the fastener loads considers displacement compatibility in one dimension and does not consider displacement compatibility in the direction perpendicular to the loading direction. This one dimensional displacement compatibility in the analysis could lead to an inaccurate analysis in complex loading situations. Also, stiffener, or substructure effect(s) are currently not taken into account in this version of RAPID.

The repairs analyzed by the RAPID code are primarily for fuselage structures and only pressure loads are considered in the analysis. There is currently no provision for flight-by-flight spectrum loading and does not take into account the effect of thermal stresses. There are also no damage tolerance analysis models to account for retardation effects. Also, there is no capability to take into consideration durability, or crack initiation life in the repaired structure. In many structures, the crack initiation life emanating from stop-drilled cracks could be significant, thereby influencing the inspection requirements of a repaired structure.

In conclusion, the RAPID code is state of the art in bolted repair/analysis codes. The complicated process of bolted metal-to-metal repair design/analysis is attempted through the preliminary version of this code. With the introduction of more sophisticate analysis techniques, complex geometry inclusions and loading situations, RAPID has the architecture of an excellent code. The next section lists, according to the developers, future enhancements to the RAPID code.

5.5.3. Future Directions

In the remaining version of RAPID (Phase I) the developers plan on including the ability to perform a repair over splice joints and implement repairs over stiffeners.

In Phase II of the RAPID program a two-dimensional finite element approach will be employed to improve the determination of fastener loads and stress distributions in the area of the repair patch. In addition, there will be included additional load spectra including "user input" and Twist spectrum. Also the fatigue crack growth module will be enhanced to perform a cycle-by-cycle spectrum analysis with retardation. The RAPID program will also include the effect of a repair in the proximity of other repairs.

Any omissions of future directions and enhancements is purely accidental. The enhancements mentioned specifically above were in regard to those objections stated in the earlier section. For further information regarding specific enhancements, future directions, etc., please contact Dr. Paul W. Tan or Dr. John G. Bakuckas, Jr. at the FAA Technical Center.

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APPENDIX A: ALC QUESTIONNAIRE

ANALYTICAL SERVICES & MATERIALS, INC.
c/o WL/FIBEC, 2130 Eight Street, Suite 1, Wright-Patterson AFB, OH 45433-7542

USAF CONTRACT F33615-94-D-3212, DELIVERY ORDER 0004
"AIRCRAFT REPAIR/ANALYSIS CODE EVALUATION"
CUSTOMER QUESTIONNAIRE

1) Who is responding to this survey?

Name: ______________________
Title: ______________________
Address: ______________________
____
____
Phone: ______________________
Fax: ______________________
E-mail: ______________________

Would you object to being contacted, if necessary, to provide additional details regarding this survey? ***(Please Circle)*** **Yes** **No**

2) Which of the following aircraft do you service on a regular basis? (Please include the Mission Design Series (MDS) and projected service life for each.)

Additional space has been provided to include the MDS of aircraft which have different service lives or other aircraft not mentioned here.

CHK (√)	AIRCRAFT	MDS	PROJECTED SERVICE LIFE
	A-7 CORSAIR		
	A-10 THUNDERBOLT		
	C-5 GALAXY		
	C-130 HERCULES		
	C-141 STARLIFTER		
	F-5		
	F-15 EAGLE		
	F-16 FIGHTING FALCON		
	F-111 AARDVARK		
	T-37		
	T-38 TALON		
	KC-135		

AS&M

ANALYTICAL SERVICES & MATERIALS, INC.

c/o WL/FIBEC, 2130 Eight Street, Suite 1, Wright-Patterson AFB, OH 45433-7542

3) Which category best describes your facility? ***(Please Circle)***

Base/Field   Depot   ALC   SPO   Other ________

4) What type of services are performed at your facility? ***(Please Circle)***

Repairs   Inspections   Modifications   Design/Analysis   Other ___________

5) What are the most common structural repairs performed at your facility? ***(Please Circle)***

Fuselage	Skins	Frames	Longerons	Other ___________
Wing	Skins	Ribs	Spars	Other ___________
Other	____________________			

6) What is the most common form of structural damage that you repair at your facility? ***(Please Circle)***

Fatigue Cracks   Corrosion   Other _____________

7) What patch materials are you currently using to perform aircraft repairs? ***(Please Circle)***

Metal (Specify)________   Boron/Epoxy   Graphite/Epoxy   Other __________

8) Which of the following approaches do you use to design repairs? ***(Please Circle)***

Static Strength   Damage Tolerance   Other ___________

9) Do you design your repairs with static strength analysis and repair the structure with the provision for subsequent damage tolerance analysis? ***(Please Circle)***   **Yes**   **No**

10) Do you specify inspection requirements based on: ***(Please Circle)***

Damage Tolerance Analysis   In-Service Experience   Other ______________

11) Briefly explain your aircraft repair design procedure:

____

____

____

____

12) What type(s) of computer facilities do you have available to perform numerical analyses? ***(Please Circle)***

Mainframe Wkstn (Type)________ PC MAC Other __________

13) What type(s) of software do you use to perform static, durability and damage tolerance analyses of repaired structures? ***(Please List, if Applicable)***

______________________________ ______________________________
______________________________ ______________________________
______________________________ ______________________________

14) Do you have on-site personnel trained in Finite Element Methods (FEM) in order to perform various structural analyses? ***(Please Circle)*** **Yes** **No**

15) Do you have loads data, fatigue spectrum and structural details readily available for performing repair design and analysis? ***(Please Circle)*** **Yes** **No**

16) Do you (or your contractors) perform structural analyses to determine if there is a significant redistribution of stresses due to the presence of a repair? ***(Please Circle)*** **Yes** **No**

17) Do you foresee a need for a (non-FEM based) software tool to aid with the design and analysis of repaired aircraft structures? ***(Please Circle)*** **Yes** **No**

18) What would you say is your major need in your repair design and analysis procedure? ***(Please List)***

____
____
____
____

19) Please include any additional comments you may have concerning aircraft repair, design and analysis procedures, etc.

____
____
____
____

** Special thanks to R-Tec for providing assistance with the preparation of this questionnaire