AGARD-R-772

AGARD

ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

7 RUE ANCELLE 92200 NEUILLY SUR SEINE FRANCE

AGARD REPORT No.772

Analytical Qualification of Aircraft Structures

(La Qualification Analytique des Structures d'Avion)

NORTH ATLANTIC TREATY ORGANIZATION

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Analytical Qualification of Aircraft Structures

(La Qualification Analytique des Structures d'Avion)

Papers presented at the 70th Meeting of the Structures and Materials Panel of AGARD in Sorrento, Italy, 1st to 6th April 1990.

North Atlantic Treaty Organization
Organisation du Traité de l'Atlantique Nord

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Preface

At its 70th meeting in Spring 1990, the Structures and Materials Panel held a Workshop to address the role of structural analysis design in relation to aircraft qualification procedures, in order to establish guidelines for the future and to seek out those areas where there exists a commonality of approach between nations.

Discussion of the papers presented and a final summary session revealed some common concerns and issues and gave rise to several recommendations.

It is hoped that the Workshop — bringing together the various views of the aircraft industry and certification agencies — has served in achieving the goal of showing a way to reduce development cost and increase reliability of structures.

Préface

Au printemps 1990, lors de sa 70ème reunion, le Panel AGARD des Structures et Matériaux a organisé une réunion de travail pour examiner le rôle de la conception analytique des structures en fonction des procedures de qualification des aéronefs. Les participants ont eu pour objectif d'établir des directives pour I'avenir et d'identifier des domaines ou une approche commune des problèmes existe au sein des pays membres de l'OTAN.

Lors des discussions qui ont suivi la presentation des communications et pendant la seance de clôture, les participants ont mentionné un certain nombre de préoccupations et de problèmes communs dans ce domaine et ils ont formulé des recommandations appropriées.

On peut espérer que cette reunion de travail, qui a permis de confronter les différents avis des représentants de l'industrie aéronautique et des organismes d'homologation, a montre les voies à suivre pour réduire les coûts de développement et pour obtenir une meilleure fiabilité des structures d'aéronef.

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A ROLE MODEL FOR QUALITY MANAGEMENT IN FINITE ELEMENT ANALYSIS.

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SUMMARY.

Many engineering companies use a Quality Management System to ISO 9001 as a means of controlling quality and standards in their products and operations. The National Agency for Finite Element Methods and Standards has recently issued a Quality Systems Supplement on the application of ISO 9001 to the use of finite element analysis in the design and validation of engineering products. The paper presents a 'role model' for a quality system designed to fulfil the requirements of that document. Quality aspects of the following topics are covered; management of the analysis operation; acquisition, development and verification of software; qualification and documentation of analysis methods; project analysis; education and training of personnel. comments are included, based on experience of implementing finite element quality procedures.

1 INTRODUCTION.

Many commercial organisations have a Quality Management System (QMS) in compliance with the requirements of ISO 9001 (ref. 1), BS 5750, AQAP1 etc. Used sensibly, a QMS provides the necessary infrastructure for improving and controlling the quality and reliability of finite element analysis. The National Agency for Finite Element Methods and Standards (NAFEMS) have produced a Quality Systems Supplement (QSS) which addresses the application of ISO 9001 to finite element analysis. The QSS (ref. 2), like all QA requirements specifications, is written primarily for assessors and in the required format of ISO 9001. This is not the most useful form for prospective assessees, who have to fit the 'jigsaw puzzle' of requirements into a quality system. The purpose of this paper is to outline a quality system model containing the main features required to satisfy the QSS. Except for items covered by the overall QMS infrastructure, the model attempts to cover every section of the QSS pertaining to product integrity analysis. In order to correlate the features of the model with the QSS requirements, references to sections of the QSS are included in the text in the form (Q,4.1.1). These relationships are also summarised in Table 1 to help identify relevant sections of the current paper when working with the QSS.

This interpretation is not unique and is only one of many systems which could fulfil the requirements. However it is hoped that, by highlighting the relevant issues, it will help others in designing QA procedures appropriate to their application.

2 FITNESS FOR PURPOSE.

QA does not require every analysis to be done immaculately to the highest degree of accuracy. The key to effective QA is to match the degree of quality control to the purpose of the analysis. To this end the QSS (Q,appendix B) defines three 'categories of importance' of analyses as either 'VITAL', 'IMPORTANT' or 'ADVISORY'. Briefly these relate to the consequences of failure of the product and the role which the analysis fulfils in the demonstration of its integrity. The degree of control in each QA function is linked to this category so that, for example, the degree of control for a 'vital' analysis is far greater than that required for an 'advisory' analysis. This categorisation also operates in the inverse mode. Where an analysis with the required degree of quality control cannot be provided for the application, then some other means of integrity demonstration, e.g. physical testing, is required. Also used extensively in the document is the 'scope of analysis' which is defined by the type of analysis, e.g. linear elastic, transient dynamic, etc and the product type e.g. pipe systems, airframe structures, etc.

As well as the definitions given above, the validation terminology of ref. 3 is used to distinguish the 'verification' of software from the 'qualification' of analysis procedures.

3 OVERVIEW.

Like any engineering endeavour, formal QA in FE isn't implemented in a day, it is a process of development built on intent and experience. The most important aspects are a management commitment to QA (Q,4.1.1) and setting up the QMS framework in which it can be implemented.

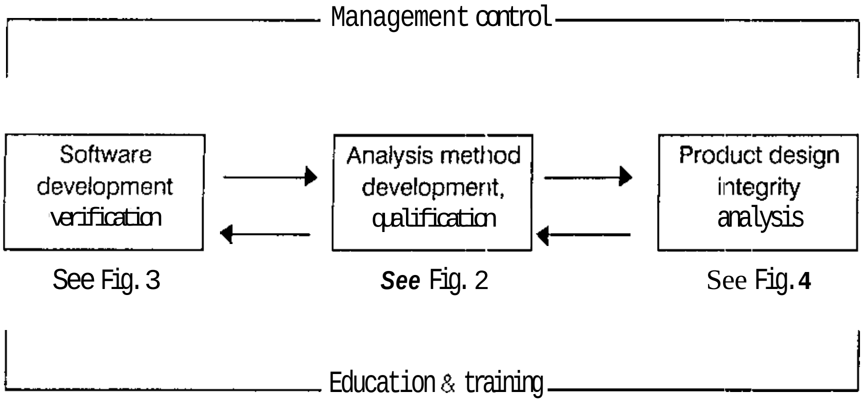

Figure 1. Quality mcdel overview.

The total analysis QA system (figure 11 is modelled in terms of:-

Three serial activities

- o acquisition, development and verification of analysis and associated software.
- o development and qualification of analysis methods, and
- o product design development and integrity analysis,

where the sole purpose of the first two activities is to ensure the efficient and robust operation of the latter, and

- o The management functions which implement and coordinate the activities
- o The education, training and experience of personnel.

Figures 2 to 4 give details of the three central items of figure 1 and can be synthesised to form a diagram of the total QA System. The asterisks * denote 'controlled documents', i.e. subject to formal issue and change control procedures.

Each of the above aspects is addressed in the following sections, starting in the middle.

4 DEVELOPMENT AND QUALIFICATION OF ANALYSIS PROCEDURES.

Analysis methods development is an activity which most analysis organisations actually do, although it may not be conspicuous or documented. The development and qualification of analysis methods is central to QA in the use of FE, it provides the link between the general purpose software and its use in the analysis of particular products. Whether it is performed within a project, or by a group set up specifically for the purpose, it is recognised as a distinct activity coordinated by appropriate technical management (Q,4.1.2.1).

The activity is driven by project analysis requirements. Its main functions are:-

- o Evaluation of software appropriate to the analysis requirements.
- o The development, qualification and documentation of analysis procedures.

Fiaure 2 represents the analysis methods development cycle. Starting from the product-analysis-requirements (2a), software is evaluated (2b) and used in the development of analysis procedures. The procedures are then qualified by validation analyses (2c) using independently derived results. After approval by the appropriate technical and project authorities, the documented qualified procedure is incorporated into the analysis procedures library (2d) and released for product analysis. Provision is made for analysis procedure error reporting and control (2e). Adjuncts to the above cycle are liaison with software suppliers on error corrections (2f) and the software acceptance procedures (2g). Further details of these functions are outlined below.

A significant part of the analysis methods development concerns the acquisition of analysis and associated software and this issue is addressed first.

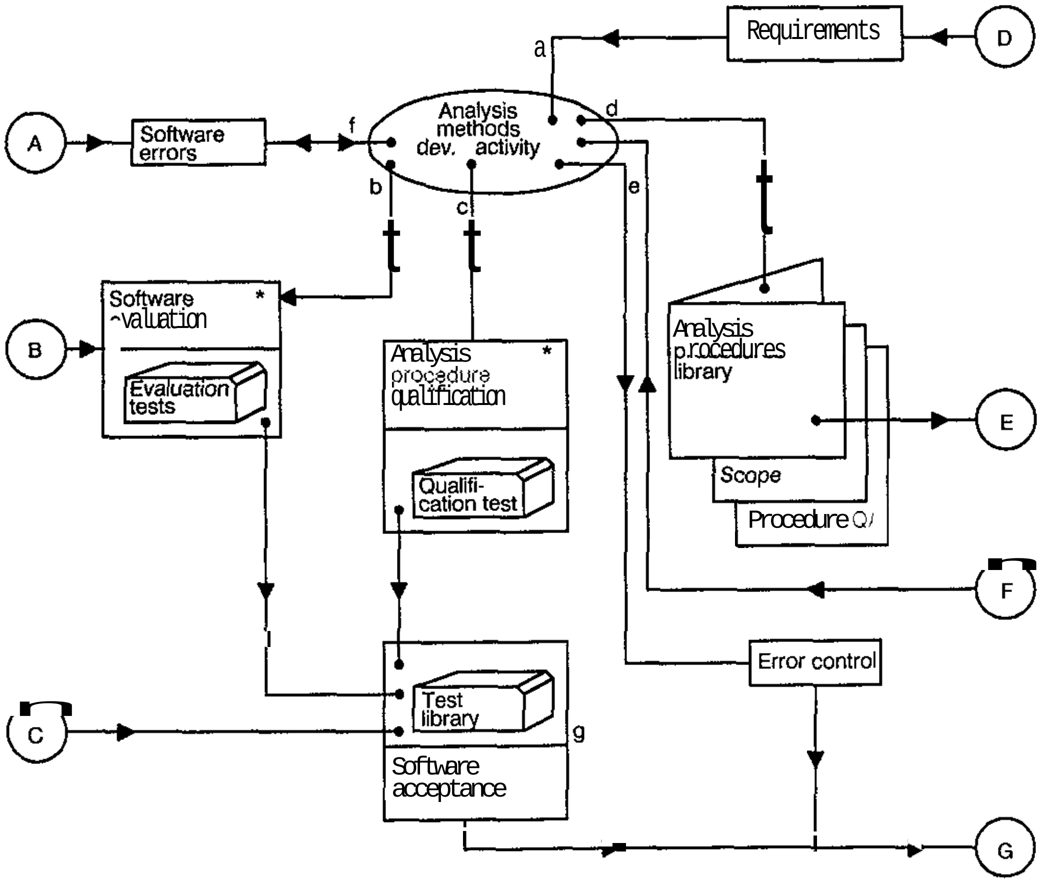

Figure 2. Development and qualification of analysis methods.

5 ACQUISITION OF SOFTWARE.

The following assumes the common situation where the majority of analysis and associated software is acquired from a supplier outside the product design organisation.

The QA issues relating to purchased software are:-

- o Evaluation of the suitability of the software to the analysis requirements of the product and the inherent limitations of the software.
- o Verification of the software against its functional specification.
- o The software suppliers software Quality Control (SQC) system.
- o The software support service.

The first two items relate to the software itself (Q,4,6,1), the latter to the organisation supplying the software (Q,4,6,2).

5.1 Evaluation and verification of software.

Distinction is made between evaluation of the software and verification of the software. Evaluation (figure 2b) is based on the software users perception of the code in respect to its prospective application. The function of evaluation is to ensure that the software is appropriate for the required application, that its inherent limitations are known and that it is not used beyond those limits. Software verification (figure 3c), on the other hand, is based on the software developers perception of the code in respect to its technical and functional specification. The purpose of verification is to demonstrate that the software does what it purports to do. Thus verification generally confirms what the software will do satisfactorily, whereas evaluation-demonstrates its applicability to the product analysis requirements.

Software evaluation is part of the product design organisation's quality system in software acquisition. The two sources of preliminary evaluation are the software theory and validation documents. Examination of the theoretical basis and numerical algorithms, together with the range of validation problems, may be sufficient to demonstrate the applicability and limitations of the software. Unfortunately software validation manuals do not generally demonstrate the limitations of the software (indeed it unusual for manuals to contain problems that perform badly) and the end user has to do it himself. Similar comments apply to tests for invalid problem or solution rejection (ref. 3).

The software is evaluated by identifying and running problems, with established solutions. which reflect the technical requirements of the product analysis. Particularly attention is given to the analysis type. One of the main activities of NAFEMS is the production of benchmark problems (ref 5 to 10) for a variety of analysis types, These provide a useful source of evaluation tests. Where these are not adequate for the purpose, it may be necessary to seek or design well specified problems which will demonstrate the software's capability with respect to the analysis requirements. Evaluation provides technical management with the required appreciation of the software limitations (Q,4.18). The evaluation tests are retained and used. in the subsequent software acceptance procedures (see 5.5).

Software verification is primarily a software suppliers function, and should be part of his Software Quality Control system (see 5.2), however it is the responsibility of the user to ensure that the verification has been performed. Evidence of satisfactory verification should be contained in the validation manual supplied as part of the software support. The verification tests must include the fundamental tests described in section 2.2 of ref. 4. Basically these demonstrate that the software has satisfied fundamental tests for soundness and convergence, for example, element tests for invariance, rigid body modes, constant stress patch tests etc. If the verification tests are not documented, the software supplier is required to provide the verification test for use in the product design organisation. The tests are examined.for complete coverage of the theoretical basis, facilities and numerical algorithms contained in the theoretical and other support documents. Selected software verification tests may be included in the software acceptance procedures (see 5.5).

In the QA system, software which does not comply with the requirements of evaluation and verification is not made available for use in analyses of high categories of importance. The importance of the theoretical and validation manuals, in software evaluation and verification, should be noted. Some guidelines to verification and evaluation testing is given in section 2 of ref. 4.

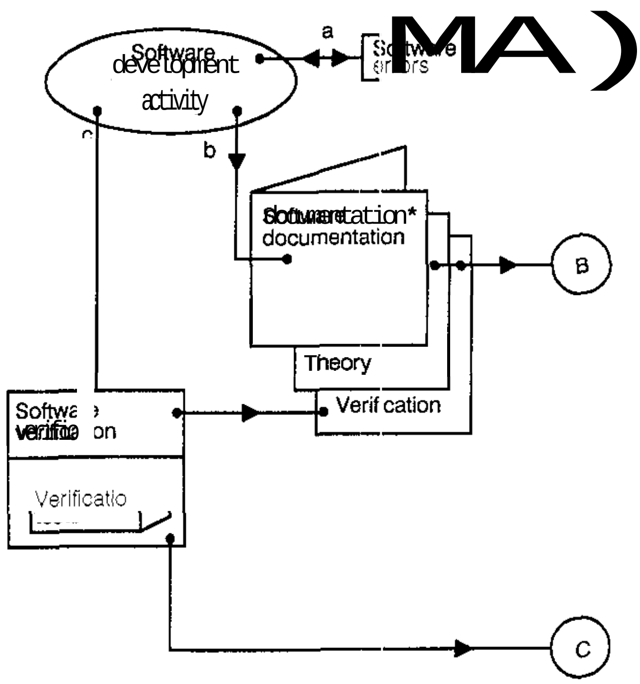

Figure 3. Software acquisition.

5.2 suppliers SQC system.

software suppliers are treated like any other suppliers of material and services and are required to demonstrate a satisfactory and effective quality system (Q,4.6.2). The quality of software, supplied for demonstrating the integrity of the product, is as important as the material supplied for its manufacture.

The most important aspects of the software suppliers SQC, relevant to the product user are

o control of the software development process,

o verification of the software (5.1), and

o the procedures for identifying, controlling and correcting software errors.

control of the software development process insures against deterioration of the software under continual enhancement and error correction. It provides some degree of confidence that the supplier can continue to provide the facilities required to support product integrity. Withdrawal of an important analysis facility can have serious repercussions on product support. Where such a risk exists, the source code may be lodged in escrow. Software quality systems vary considerably between FE software suppliers. It is the responsibility of the user to ensure that the software suppliers SQC is satisfactory. The recent TICKIT (ref. 11) initiative, on the application of ISO 9001 to software development, is useful in evaluating the supplier's quality control.

Large computer programs inevitably contain some errors and it is unreasonable to expect otherwise, however it is important that the code developer has effective procedures for controlling software errors. The SQC system must define the suppliers procedures for detecting, reporting, controlling and rectifying errors.

5.3 software support.

The requirements for support of software, used in integrity demonstration analyses, include:-

o error and correction notification,

o documentation,

o training specific to the software.

In addition to the SQC aspects of error control (5.2) the software supplier must establish effective communication of detected software errors, in particular the reporting of serious software errors so that the user can control the use of defective software (Q,4.13). Periodic supply of a 'bug list' provides the user with a useful monitor on the suppliers actions.

A guide to the documentation required to support use of the software is given in section 7.1 of ref. 4. As well as the normal 'users manual' the documentation should cover the verification, theory and numerical algorithms, demonstration problems, program structure and computer system requirements. As noted in section 5.1, the two documents which are most important to QA are the theoretical and validation manuals. Unfortunately these are the two formal documents which suppliers are least likely to provide and the user may have to exert some pressure to obtain them. The theoretical manual is the closest thing to a functional specification against which the software can be verified. The validation manual provides the primary evidence of software verification. Care may be required in differentiating between verification and demonstration problem documentation. Where necessary, access to the software suppliers actual validation test, rather than those in his 'demonstration' manual, is required.

The software supplier must provide training in the use of the software. This should not merely cover the mechanics of input, but should include such topics as the solution methods used, limitations of the theory, element formulations and algorithms, diagnostic and error messages and their meaning, outline of the program structure and operation. Specific examples should be included which demonstrate the use of the code over the range of facilities used in product analysis. The course need not cover basic FE methods, which may be provided elsewhere, but should be sufficient to enable the product user to provide in house consultancy on use of the software (8.2).

5.4 Internal Software.

Methods development inevitably involves some 'internal' software development which may not be identified as such. Many FE systems allow the user to insert subroutines or to control the execution sequence of modules. Data generation modules may be written and linked in a sequence of programs. Examples include the generation of constraint equations, element stiffnesses and load vectors etc. Use of uncontrolled software of this type is identified as a significant QA risk. All internally developed software of this type, used in qualified analysis procedures, is subject to the same QA requirements as external software, albeit in abbreviated form, in respect to documentation, verification and support. This, together with software acceptance testing, software error reporting (see 5.5) and code security procedures, forms the minimum in house software quality system (Q,4.1.2). If the product design organisation has a software development department, not particularly for FE, the SQC procedures of that organisation can be suitably adapted to cover this aspect.

5.5 Software acceptance and error control.

The purpose of the software acceptance testing is to ensure that new releases or updates of software or hardware do not invalidate current analysis procedures, qualification analyses or evaluations (Q,4.6.4). Acceptance tests are executed at initial release of the software to product analysis and for each update of the software or for major changes in hardware. In practice, after the initial trials, subsequent releases use simple 'file to file' checking software to compare the output with that of a previous satisfactory release. Only new tests, or deviations from previous results, are checked manually.

The 'software acceptance library' consists of:-

- the test problems used in the software evaluation (5.1).
- selected tests, relevant to the Project analysis requirements, taken from the software verification tests (5.1).
- the software execution decks used in qualification of current analysis procedures (see 6.3).

Where there are significant changes in the results, these are reported to the supplier and the product analysis organisation. Where necessary, the analysis procedures are amended and the qualification tests repeated. After satisfaction of the acceptance trials, the software is formally released for product analysis use with identification of the versionlrelease number, revised software documentation and a summary of the software changes.

The function of the product design organisation's error control is to liaise with the supplier on software errors. This involves logging and reporting errors to and from the supplier and disseminating information on errors and corrections to the users. Particular attention is given to controllinu the use of software containinu known errors. Users are alerted to outstanding errors by warning messages displaced at access to the programs (Q,4.13).

6 ANALYSIS PROCEDURE DEVELOPMENT.

Analysis procedure development requires a range of expertise covering product knowledge, FE modelling, software and hardware. All these must be present in the development activity. The development of qualified analysis procedures starts with a systematic review of methods in common and established use. Documentation and qualification of those methods forms the basis of the analysis procedures library. This consolidates the product analysis operation, improves its efficiency and eliminates uncertainties due to variations in practice.

6.1 Documentation of analysis procedures.

Undocumented procedures suffer from atrophy and misuse and can result in loss of experience with changes in personnel. The documentation of an analysis procedure (Q,4.2) includes the following items:-

- The output data for which it has been qualified and the order of its accuracy. This is limited solely to the data correlated in the qualification analyses (see 6.3). The order of accuracy is based on the degree of that correlation.
- The scope and limitations of its applicability, defined in sufficient detail to ensure that the procedure is not used outside that scope. Typically these include the identifying characteristics of the physical structure, loading conditions, analysis type, limitations of the behavioural modelling and theory assumptions.
- Reference to documented analyses used in qualification of the procedure (see 6.2).
- The maximum category of importance of analyses in which the procedure may be used. This is based on the degree and number of reference validation analyses. The category of importance defaults to 'advisory' if the procedure is not qualified.
- The input data required for satisfactory execution of the procedure.
- The software and facilities to be used, e.g. element, solution and loading types and options, valid constraint equations etc.
- The analyst controlled procedures, e.g. the geometric representation, mesh size and configuration, treatment of sub-scale features, representation of loading, boundary conditions and constraints, material modelling, solution procedures, stress output interpolation, interpretation to the physical problem etc. A comprehensive list of these items is given in sections 3 to 5 of ref. 4. The level of detail must be sufficient to enable an analyst, of the relevant level of competence (see 8.2), to execute the analysis in a satisfactory manner.
- The QA checks to be exercised within the procedure; pre-analysis checks such as element distortion, compatibility and connectivity; analysis execution checks such as numerical stability, convergence in iterative solutions, residuals, etc; post-analysis checks such as stress continuity, total load, small displacements etc. Examples of this type of in procedure checks are given in ref. 12 and sections 4.3 and 4.4 of ref. 4.
- Identification of the 'procedure owner' responsible for action on errors, omissions and queries.

6.2 Qualification of analysis procedures.

Analysis procedure qualification is performed by analyses of realistic engineering structures for which results, appropriate to the purpose of the analysis, can be confirmed by some independent means. Confirmatory results are typically obtained from physical tests supporting integrity demonstration of previous products, service experience, failure investigations, alternative analyses or third party assessments. Many test results can be found in the open literature. The qualification tests are executed to the defined procedure, preferably by an independent analyst of the appropriate level of competence, to ensure that the procedure documentation is adequate. The documentation of a qualification analysis is sufficient to enable it to be repeated without reference to any other documentation so that, if the procedure is amended, the qualification can be repeated with a minimum of delay. Qualification analysis reports are subject to formal issue and change control. The software execution decks, used in the qualification analyses, are added to the test library (5.5) for software acceptance.

In assessing the qualification in respect to the category of importance, both the degree of correlation and the reliability of the confirmatory values are taken into account. The more qualification tests and correlations available, then the greater the confidence in the procedure and the higher the category of importance of analyses in which the procedure may be used. Qualification of analysis procedures provides project management with an appreciation of the inherent assumptions and limitations of the analysis method (Q,4.18).

7 PRODUCT ANALYSIS.

Figure 4 shows the project analysis cycle. The product analysis is controlled by the project analysis manager through the project analysis plan (4a). The individual analysis tasks are allocated (4b) through analysis specifications (4c) which use the analysis procedures qualified in the methods development activity. After checking of the results (4d) each analysis is documented (4e), approved by the analysis project manager and used to update or amend the plan (4f). Details of the individual items are given in the following sections.

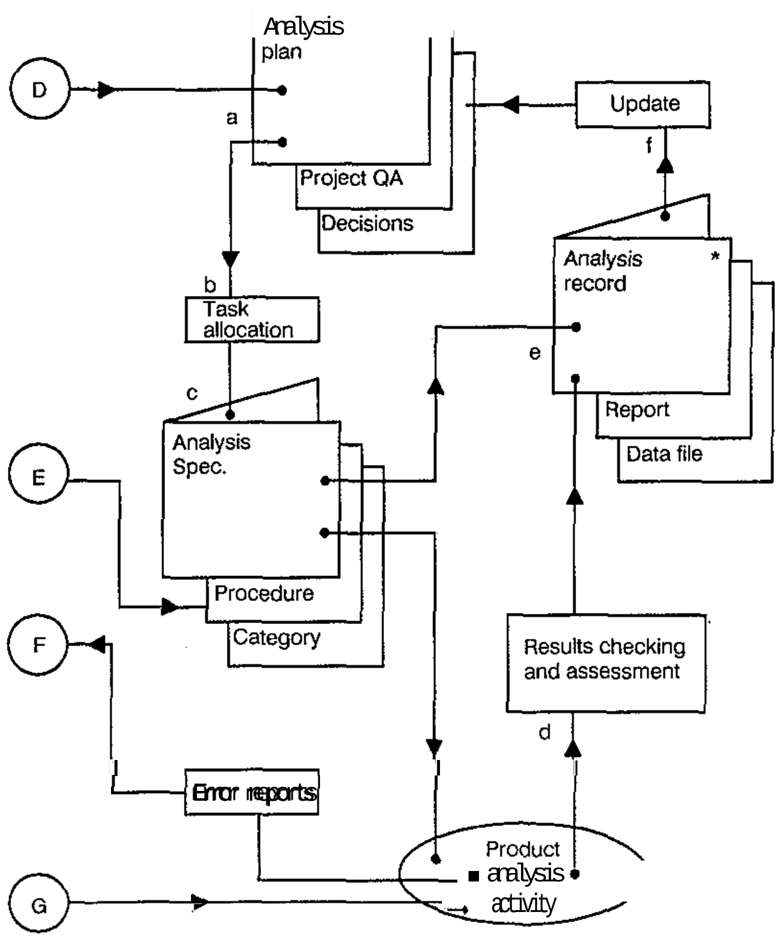

Figure 4. Product integrity analysis.

7.1 Project Analysis Plan.

The project analysis plan is a dynamic document which starts as a broad outline and becomes more detailed as the design evolves. It is updated and amended as analysis results are obtained and may involve tasks in the methods development activity. At completion of the project the analysis plan becomes a record which correlates the individual analysis reports. The analysis plan is a controlled document subjected to periodic reissue and change control procedures.

Particular QA features of the plan are (Q,4.4.2):-

- The identification of decisions or reviews, based on stsuctural features and analysis results, which may result in updating the plan and redirection of analysis activities.
- Quantitative assessments, estimates, design reviews and correlations between analyses to be used in checking the results from individual analyses.
- The scope and category of importance of each analysis.

Based on the scope of the analysis and it:; category of importance, each analysis task is allocated to an analysis team (supervisor, analyst and software consultant1 who, collectively, fulfil the requirements of experience and expertise lsee 8.31.

7.2 Analysis Specifications

The analysis team first prepares an analysis specification which is agreed by the project manager before the analysis proceeds. The purpose of the specification is to ensure that the analysis is sufficient to fulfil its purpose, that the appropriate input data is available and that the results are relevant to the project needs. The specification includes:-

- the purpose of the analysis and output data required,
- the sources of authentic data for input,
- the qualified analysis procedure to be used,
- the input and results checking procedures to be invoked.

Where a qualified procedure is not available the procedure itself is outlined in the specification and the results used only for advisory purposes.

7.3 Results checking

At completion of the analysis the results are checked and assessed. This involves:-

- Confirming that all the analysis procedure QA checks (6.1) have been executed satisfactorily.
- Comparison of the results with the estimates and correlations identified in the analysis plan (7.1). Quantitative estimates are obtained from traditional simplified analyses, ball park and eyeball values based on product knowledge, comparison with coarse mesh analysis, formal assessment or review etc. Checks like total mass, load balance between substructures and other global quantities provide useful indications of the validity of the results. Some useful results checks are included in ref. 12.
- Assessment of the results based on knowledge of the physical problem. For analyses of high category of importance, an independent assessment is performed by a qualified individual who is not a member of the analysis team. A guide to results assessment is given in section 5.2 of ref. 4.

7.4 Analysis documentation

The controlled documentation of the analysis consists of an analysis report, the analysis record and project computer data files.

The analysis report provides only the information relevant to product design and integrity. It includes:-

- the purpose of the analysis,
- an outline of the representation of the physical structure by the analysis model,
- summary, discussion and accuracy assessment of the principle results,
- relevance of results to the engineering problem and design recommendations,
- references to enable further details to be obtained from the analysis records.

Summaries of analysis reports, including the scope of analysis, category of importance and identification of analyst, supervisor and software consultant, are stored on a database. This is used for subsequent task allocation decisions (see 8.3).

The analysis record includes:-

- o the analysis specification (7.21,
- o the key input data, in terms of the physical structure and model representation, and the sources of that data,
- o selected output relevant to the purpose of the analysis,
- o a summary of QA checks,
- o location of the input in the project computer files and the version/release number of the software used.

The analysis record, together with the stored input files, should be sufficient to enable the analysis to be repeated or updated reliably with a minimum of effort. Guidelines to the content of analysis reports and records are given in sections 6.1 and 6.2 of ref. 4.

The project computer files contain the program input, and where appropriate, output relevant to the analysis. Where a number of design iterations are involved, only that germane to the definitive version is stored. The data to be retained is defined in the job closedown procedures (Q,4.15) and is stored in secured files.

8 QUALIFICATION AND TRAINING OF PERSONNEL.

The QSS addresses the qualification of personnel on the basis of collective accreditation, rather than at the level of individuals. This provides flexibility in the movement and advancement of personnel and avoids any contentious issues associated with unsolicited personal accreditation. The requirements are that a 'team', who collectively provide the necessary expertise, be allocated to each analysis task . An analysis team consists of a supervisor, analyst and software consultant. One individual may fulfil more than one role, provided the requirements are satisfied.

8.1 Personnel requirements

The requirements are quantified in ref. 13 in terms of:-

- o formal academic or professional qualifications,
- o product analysis experience, not necessarily in FE,
- o FE modelling and problem solving relevant to the scope of the analysis, and
- o FE software application experience.

The required degree of training and experience varies with the category of importance of the analysis and must be relevant to the scope of the analysis. Formal training in FE methods and software application contributes to personnel accreditation. Job experience, accumulated in one category, contributes to qualification for tasks of a higher category. Thus an analyst may perform low category tasks, under supervision, until sufficient experience is gained to perform such tasks unsupervised. He may then move to a higher category under supervision, and so on. In cases where the required software application expertise is not available within the product design organisation, it may be necessary to contract suitable personnel from the software suppliers organisation. Similarly if the organisation is subcontracting analysis of an unfamiliar product, the necessary product expertise may be provided by the contractor.

8.2 Provision of training.

Training is provided by a combination of formal tuition and on job training. It is effective to use two types of course, one in general FE methods and others in the use of particular FE software packages. This leads to a better appreciation of the analysis technology and maintains flexibility in the use of different codes.

Lecture courses in FE methods are provided for all analysis personnel, except for those who qualify by virtue of experience alone. In a large organisation it is economic to run such courses in house using an external specialist. The course follows the NAFEMS recommended syllabus (ref. 14). This syllabus also provides a basis for evaluating training courses. As a follow up to the teaching, suitable 'starter packs' of product oriented problems are made available. Self study is encouraged through subsidising purchase of the NAFEMS primers (ref. 15). Training courses in the use of particular software should be provided by the software supplier. The content of such courses 15.3) is reviewed and neaotiated, with the supplier, in respect to the product design organisation's needs. The supplier may require some encouragement to emphasise the limitations of the software. Courses of both types are provided when the requirement is identified (see 8.31.

On job training is controlled by the task allocation process itself (see 8.3 and 7.11. The requirements for experienced personnel in high category of importance tasks, forces inexperienced personnel to be used in low category tasks until sufficient experience has been gained to undertake higher category work.

8.3 Personnel Records

The following personnel information is extracted from various sources:-

- Professional qualifications and years of engineering experience from the company personnel records.
- Training course attendance from the training records.
- The category of importance, scope of analysis, identification of the supervisor, analyst and program consultant, from the analysis report summaries (7.4).

This information is coordinated in a data base and used in a variety of ways:-

- In task allocation. Given the category of importance, scope of analysis and a list of available personnel, all possible teams which fulfil the requirements are returned. This provides a management aid in task allocation and job scheduling. Note that this does not prevent the project manager from rejecting any team which, based on other considerations, is deemed to be inadequate.
- In the provision of training courses. Given the training course type, returns a list of personnel that require the training in order to progress to higher category of importance tasks. Provides a means of scheduling courses.
- In on job training. Given personnel identification, returns list of analysis scope, category of importance and roles, required to advance to higher category tasks. Provides a means of monitoring the advancement of individuals.

9 DISCUSSION.

The real quality of the analysis is determined by technical considerations. Operational QA procedures are only necessary to ensure that quality controls are actually performed. The QA system should be a tool to help you do a better job, not some bureaucracy you have to overcome. A QA system, which is dominated by procedural aspects is ineffective. The emphasis should be on technical documentation, which generally speeds up the operation, with procedural QA documentation, which appears to slow it down, kept to a minimum. It is important to use experienced technical staff in determining quality procedures, using a QA consultant only in an advisory capacity. Unfortunately not all organisations use their most imaginative people for QA activities. Bureaucrats generate bureaucracy which is a burden to managers already working under project pressures. A recent survey by Zins (ref. 12) shows that the majority of analysts are actually in favour of technical quality control procedures.

Too much control can stifle innovation. A good QMS shouldn't stop you doing anything except endangering the product or the company. Control should be aimed mainly at trapping errors as soon as possible after they occure. In this context, quantitative checking is identified in six progressive phases backing product integrity demonstration.

- In the software verification (5.11,
  i.e. is the software correct?.
- In the software evaluation (5.11,
  i.e. is the software appropriate to the application?.
- In the analysis qualification (6.21,
  i.e. is the procedure for using the software correct?.
- In the software acceptance trials (5.5).
  i.e. has the software changed?.
- During execution of the analysis procedure (6.11,
  i.e. are the software and analysis procedure behaving well in the particular analysis?.
- Post analysis results checking (7.3),
  i.e. are the analysis results credible?.

The recommended first steps in implementing a FEQA System are to:-

- constitute a technical body with responsibility and authority for the analysis quality system,
- agree responsibilities, within the organisation, for each activity required in the QSS,
- review current practices against the requirements.

Experience of implementing finite element quality assurance (ref. 16) reveals that it is mainly a process of unifying, formalising and reinforcing existing practices and procedures.

TABLE I Correlation of the Quality Model and the QSS requirements.

QSS requirement	see section.
4.1.1	3, QMS
4.1.2	5.4, QMS
4.1.2.1	4, 7, 7.1
4.1.2.2	7.3
4.2	6.1, QMS
4.3	7.1
4.4.2	7.1
4.4.2.1	7.2
4.4.3	7.2
4.4.4	7.4
4.4.5	5.1, 5.2,
4.4.6	Figures 2, 4
4.5	6.1
4.5.1	5.4, 6.1, 6.2, 7.1, 7.4
4.5.2	6.2, 7.4
4.6.1	5, 5.1
4.6.2	5, 5.2
4.6.3	5.3
4.6.4	5.5
4.8	5.5, 7.4, QMS
4.9.1	7.2
4.9.2	6.1
4.10.1	5.1, 5.5
4.10.3	7.3
4.10.4	5.1, 6.1
4.13	5.1, 5.5, 7.2
4.15.1	5.4, 7.4
4.17	8.3, QMS
4.18	8, 5.1, 6.2

QMS denotes that the requirement is covered by the parent Quality Management System in which the FE aspects are implanted.

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VALIDATION DES LOGICIELS D'ORIGINE INTERNE ET EXTERNE A L'AEROSPATIALE
VALIDATION OF IN-HOUSE AND EXTERNAL SOFTWARE SYSTEMS AT AEROSPATIALE

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SUMMARY :

In the aeronautical industry, the concept of numerical simulation is implemented, at first, in the preliminary design stage, and then during the certification activities with respect to the airworthiness regulations. This implies the development and the use of calculation software, of which it is necessary to improve the validation at different levels.

The purpose of this paper is to describe the different features of these validation tests in the design, use and evaluation of these software systems. the experience of the engineer then plays a major role in obtaining high quality models in the design of aircraft Structures.

1. INTRODUCTION

In the framework of aircraft project development, finite element analysis is performed at each step of the structural design process.

In the preliminary design stage, the results of finite element analysis may be used for pre-sizing different structural components.

Additional calculations requiring the stress analysis background of structural design specialists, may be required from finite element results.

During the progress of the structural design, several revisions of the analysis model are made, in order to optimize design on the basis of strength and weight criteria.

Finally, a stress report according to airworthiness regulations, is written; it is based upon the analysis of the finalized design, and correlated with results of experimental data obtained in laboratories.

In the past, the number of analyses performed with finite element tools was limited, because of the lack of user-friendly interface foe pre- and post-processing (consuming extensive manpower), and because of the law performance of computing resources.

Nowadays, if communication between designer (with his C.A.D. facilities) and analyst (with his Finite Element software system) is still not perfect, the number of iterations in the design process increases and the complexity of real structures is represented by more and more refined models.

The fact still remains that many assumptions arc made to reduce this complexity to a manageable level in the finite element model, for the best compromise between time saving and result interpretation, which is not easy for designing details.

In addition, new domains are opened by the Finite Element Approach. *So*, this approach, in the field of idealization, must be based on specific design check goals : the suitability of the design *must be* judged on a variety of criteria (static, dynamic, stability features, in linear and non-linear domains).

All these remarks show the complexity of structural modelling which involves validation of the software itself, organization of work and associated teams, and high qualification of engineers.

2. IN-HOUSE SOFTWARE SYSTEMS

At Aerospatiale (Aircraft Division), finite element calculations are performed by an in-house software system called A.S.E.L.F. which provides capabilities of modelling conventional linear phenomena (static, buckling analyses, ...), and non-linear aspects (large displacements, post-buckling ...), optimization strategy under static and dynamic constraints, and finally identification techniques (correction of models on the basis of test results). Sub-structuring techniques enable very large problems : resolution of linear systems with two hundred thousand unknowns for completely modelling an aircraft for flutter control studies; to be dealt with by the new generation of computers (Cray X.M.P.).

If we limit our study to the problem of validating the software itself, excluding modelling and data problems, a distinction must be made between two separate aspects : the first concerns examination of the finite element library itself, and the second, the software (other than the finite elements).

Addition of a new type of finite element always corresponds to a need either for new calculation possibilities, or for more accuracy or reliability.

When the need appears, the first Step of the research process consists in studying the bibliography on the subject and finally in choosing the right formulation according to industrial utilization : cost and performance, robustness, facility of interpretation of the results ...

Once the development of the element inside the software has been done, its validation then comprises two phases. The first consists in checking that we indeed get the expected results by making a comparison with results known from other Sources, either analytical, or published by other authors with similar types Of elements through other finite element software systems.

The second consists in determining the validity envelope of the element : limitation of geometrical shapes, sensitivity to distortion. The practical validation is then obtained by comparison between calculations and experimental results of specimen tests or existing Structures, by modifying the associated model. The interpretation of the results must take into account modelling features of the Structure : mesh, boundary conditions, loads ... Generally, these comparisons are permanent, in order to achieve accuracy of structural modelling.

The second aspect of software validation is to check the code for which, apart from the numerical approximations due to computer calculation errors, the approximations are due to the algorithms used. It is then necessary to isolate, a8 far as possible, the verification of these algorithms.

Prom a data processing point of view, modular architecture makes tests easier. An algorithm is checked similarly to a type of finite element. First of all, calculations are made according to known results : analytically or in simple and well identified situations. Comparisons between calculations and tests are also performed. If we take the example of the vibration modes of an aircraft, the frequencies and the associated modal shapes are checked with those of the corresponding vibration tests, and with analysis of the considered structural model.

Another factor of software evolution is the continuous development of tho finite element method itself, and the progressive extension of its scope. Research is still very active in this field. But from research to industrial application, there is a long process that can be qualified as validation in the broad sense of the term, which takes some time in some cases, for new analysis domains opened. Correlations between calculations and experiments may offer some information to the critical mind. For example, take the post buckling analysis of a stiffened panel. Interpretation of the results of the experiment shows that correlations can only be made by taking the initial construction imperfections and initial stresses into account. The practical modelling then consists in introducing into the model, a very small geometric imperfection resembling the deformed shape observed during testing or, for specific loading conditions. That raises the problem of previsional calculations, without performing tests. Future developments, of course, will probably permit such calculations.

General steps of validation are available for the development of other structural analysis methods. In the particular case of the boundary element method, some correlations with the finite element approach are fruitful. In a more general way, comparisons with different methods are interesting from the safety of results point of view; especially when well-suited experimental data are lacking, for investigation of new analysis domains.

3. EXTERNAL SOFTWARE SYSTEMS

In the context of European cooperation, industrial work-sharing between each partner implies responsibility for the design and manufacture of respective sections, and consequently, finite element modelling, with a given software system. It can be software available on the market (NASTRAN) or in-house software. Some studies (static load conditions or flutter calculations) require modelling the whole aircraft.

The exchanges of models between each partner raise tte problem of consistency between element libraries. Each partner is responsible for establishing the correspondence between the finite elements used in the Other Software system8 and those of the software he used himself.

Exchanges of simple representative modelling data are then made to check that the correspondences are correct. If not, the development of new types of element can be necessary, for different reasons : formula or physical property representation ...

When data about aircraft sections are exchanged, a cheek is also made an significant load conditions to make sure that good agreement is obtained with the target, to within a few percent. Validation of transfers of information, made by specialized interface software systems, is necessary; not only on the types of finite elements, but on the way in which boundary conditions, relations between degrees of freedom (different from one software system to another) are made.

The concept of Standardization of methods for transferring data between finite element software systems is now advancing, via an extension of the Aerospatiale standard S.E.T., used before strictly far exchanges between C.A.D. software systems. An interface between S.E.T. and A.S.E.L.F. has been developed; interfaces with other commercial software systems are going to be launched.

The above remark illustrates the desire to provide the structural design teams with a simulation tool that has been validated at all levels, both from the data processing point of View, for total reliability of operation, and from the point of view of formulation, the associated algorithms and the permanent comparisons with other software systems (not only based on the finite element method, such as the boundary element approach for example). The verdict of all these studies is the full scale test (ground and flight conditions).

4. DESIGN PROCESS AND VALIDATION

The Structural Department is in charge of validating design office drawings, concerning product definition and associated sizing.

If a great effort is made to find the best integration between their respective (and validated) tools, i.e : Finite Element Analysis and C.A.D. facilities, special attention is focused on work organisation in order to optimize final product definition.

For the Structural Department, in order to provide quantified information to design office teams, different kinds of finite element models are involved in the iterations between analysis and design, during the general design process.

Simplified models based on simplified geometry at the preliminary design stage, useful for first loops of optimization tools, giving, in particular, sensitivity analysis; then more and more refined models, either for the whole aircraft, or for a structural component. Detailed analyses are necessary to verify structural behaviour for the more complex cases. Updates of the drafts are made on the basis of interpretation of results according to background and experience of engineers.

Haw can interpretation be validated ? The interpretation involved in generating a suitable finite element idealization performed in a specific analysis goal, or interpretation of the model results according to assumptions made to reduce complexity of the real Structures.

Idealization means geometrical simplifications, decisions for assigning physical properties to the model, choices for joint modelling and reflection upon boundary conditions.

Some of the concepts behind various steps of interpretation, which seems justified for the analyst may lead to results, that are hard to interpret for the designer.

Developments around post-analysis of the finite element tool, by additional calculations, arc sometimes necessary, if rules can be well defined. Such is the case for practical design of fastening between panel, frames and stiffeners.

Only training, understanding of structural mechanisms under the given load, experience and knowledge of used finite element software features (such as : finite element type behaviour) can provide satisfactory results for a successful and reliable analysis, during design and certification of structures.

5. CONCLUSION

Correct adaptation of structural analysis software to technical needs, associated with training end qualification of design teams, implies a necessary and permanent effort, which is a consumer of resources; therefore a certain investment, the cost-effectiveness of which is ensured by the quality of the software product, the suitability to requirements and their evolutions.

If well-suited training is a solution to the satisfactory practice of conventional structural analysis, the problem of handling new complex calculation (non linear ...) is raised. Perhaps, in the future, development of monitoring systems implemented inside software, based on experience and knowledge of technical development teams may provide some help for that question. Evolution of work organization will probably be necessary. Reflection must Start.

From this point of view, it is extremely advantageous to participate in workshops for sharing experience.

ANALYTICAL CERTIFICATION OF AIRCRAFT STRUCTURES

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ABSTRACT

Analysis is expected to play an important role in the design and validation of future aircraft structures. This paper points out a need for the development of professional standards in order to implement the concept of analytical certification. Standards for analysis and criteria for model definition are discussed in some detail.

1. Introduction

The development of a new aircraft or a new derivative is an expensive and time consuming process. The aerospace industry around the world has built many successful aircraft over the years. Much of the past success was due to extensive test programs conducted during the development. The aircraft development cycle generally includes a myriad of test programs: coupon tests, component tests, full scale static and fatigue tests, ground vibration tests, wind tunnel tests and flight tests. These tests are the primary contributors to the ever increasing development costs. A rational approach to control costs is to increase emphasis on analytical certification with necessary (but limited) experimental validation.

The recent proliferation of commercial and public domain software, for analysis particularly, points to the advantages of analytical certification. Finite element programs such as NASTRAN, ANSYS, ABACUS, ADINA and ASKA are powerful software systems that permit modeling of complex aircraft and spacecraft structures. They are being used extensively by the aerospace industry around the world. The new structural optimization systems, ASTROS (USA), LAGRANGE (Germany), ELFINI (France), STAR (UK), SAMCEF (Belgium), etc. are in intense development, and they provide new opportunities for the design of ultra-light weight structures with stringent performance requirements. Most of the modern structural analysis software is based on a finite element formulation, and they can be used with common databases and pre and post processor software.

These analysis and optimization systems are merely sophisticated tools. The success of their application is contingent on the user's understanding of the physical system being modeled and the limitations of th-esoftware system. Availability of these programs alone does not guarantee accuracy of the analysis or reduction in computational cost. The technical community must make a concerted effort and a long term commitment to promote analysis as a reliable certification tool. This approach is particularly appropriate now because of the rapid developments in both low and high end computers and the consequent reduction in computational costs. If all the benefits, such as, shorter schedules, facility for parametric studies and the potential for technology transfer are added up, there is no question that analysis will play a key role in total quality management. Analysis can aid in two important areas of technology transfer. Transition of knowledge and tools from the research laboratories and the methods development groups to the applications and design offices is one form of technology transfer. An equally important technology transfer involves transmission of lessons learned from one system development to the next. In both cases analytical approaches offer significant cost and schedule advantages.

Analysis as a certification tool becomes a reality if the technical community establishes credible requirements in the form of

- Professional standards for analysis
- Criteria for model definition
- Certification of commercial software systems
- Benchmarking for validation
- Training - Technology Transfer
- Communication between analysis and experiments

The scope and limits of these requirements are certainly difficult to define and need careful deliberation before recommending them to the community. They should remain as voluntary guidelines and should not abridge creative applications.

The technical committees (TCS) of the AIAA (American Institute of Aeronautics and Astronautics) are addressing some of these issues. The Dynamics TC has a subcommittee on technology transfer and training. The charter of this subcommittee is to develop effective means for transferring technology developed in the research laboratories and the universities to industry where product development takes place. Similarly the Multidisciplinary Design Optimization TC has a subconimittee on benchmarking and validation. The charter of this subcommittee is to develop benchmark problems of varying degree of complexity and to provide guidelines for modeling the physics of the system and its environment. A few years ago the National Agency for Finite Element Methods and Standards (NAFEMS) in the UK had undertaken the admirable task of developing standards for finite element modeling. It has already published a number of books and primers and continues to orchestrate modeling issues in a lively magazine called "Bench Mark".

The purpose of this paper is to initiate a dialogue and to identify some of the issues of importance in analytical certification. It is just a beginning and by no means a well researched thesis to make specific recommendations at this time.

2. Standards for Analysis

The discussion in this section is in the context of finite element analysis (FEA) as applied to aircraft structures. Why finite element analysis? Aircraft structures are generally built up of many structural elements such as panels, beams and joints. They are articulated and consist of a complex arrangement of spars, ribs, skins, spar caps, rib caps, stiffeners and longerons. Before the advent of finite element analysis aircraft designers made gross approximations, such as representing lifting surfaces by equivalent beams or plates and the fuselage by beams. A representation with rods and shear panels (in the context of a multi-cell box beam) was the most sophistication that was available before the era of general purpose finite element codes like NASTRAN.

Aircraft structures are too complex or cluttered to be represented by single continuum models. These simple models do not provide enough accuracy and detailed strength and stiffness information to design modern aircraft where the performance and weight requirements are extremely stringent. The behavior of plates, beams, and rods from which aircraft structures are constructed is governed by one or more differential equations, and they can be solved with strict assumptions of continuity and complex boundary conditions.

However, when they all come together at the joints, with their differential equations, it is difficult to establish compatability and make a meaningful analysis. Finite element analysis, on the other hand, allows modelling these discrete structures by approximating the differential equations by algebraic equations which do not normally require continuity and compatability beyond the first level. Also, it is easy to represent complex boundary conditions in simple terms in a finite element analysis. An even more important consideration is that the algebraic equations can be solved very efficiently on modern digital computers. In response to this facility and versatility numerous finite element analysis codes were developed during the 60s and **70s** for public domain or commercial purposes. They are used extensively for the analysis of aerospace, mechanical, civil and marine structures. A partial list of frequently used finite element codes is: NASTRAN, ANSYS, ABACUS, ADINA and MARC. Emphasis in the 1980s has been on the development of multi-disciplinary preliminary design programs such as ASTROS. They also are based on finite element analysis. In addition, they have extensive optimization capability. When these systems are fully operational, they can really bring the impact of modern super computers to the design office in an unprecedented way in order to improve the performance at a minimum cost.

The purpose of a finite element analysis is to determine the performance of aerospace structures. The strength, stiffness, and static and dynamic aeroelastic properties can be estimated quite accurately by analysis with finite element models. When the physics of the problem is well defined by appropriate elements, boundary conditions (geometry) and loading conditions (flight environment), a finite element analysis can be very reliable and cost effective. The cost of testing can be significantly reduced by promoting high quality analysis.

The objective of analysis cannot be achieved without a disciplined and well defined approach for developing finite element models, tracking input data for a given software system and validating the output of the analysis. The present analysis practice is unstructured, and it is often difficult to verify its validity. In 1987 a preliminary Data Item Description (DID) wa-sproposed in an ASIP (Aircraft Structures Integrity Program) Conference(') for the delivery of finite element models. The DID calls for three requirements for presentation of finite element models for verification. The general requirements deal with the development of a narrative of the analysis problem. The analysis data requirements are developed for five types of analysis. The output requirements are for validation of the analysis. The contents of this DID are given next.

2.1 General Requirements

The finite element data supplied must accompany a problem narrative. This narrative must include the following items:

- Configuration version.
- Identification of the documents and/or drawings from which the model was generated. Copies of these documents must be provided if they are not available to the government.
- **A** key diagram showing the location of the component being modeled in relation to the rest of the structure.
- **A** brief description of the physical phenomena being modeled.
- A discussion on the coarseness/fineness of the grid selected.
- **A** rational explanation for the elements selected for the model.
- An explanation of the boundary conditions.

- Materials - Identification of the Mil Standard from which the mechanical properties were derived. Reasons for any deviations from the standard properties.
- A complete description of the flight maneuvers for which the loading conditions are attributed.
- Planform used for aerodynamic analyses showing all important dimensions.

2.2 Analysis Data Requirements

The finite element analysis models are classified into the following five categories:

1. Static Analysis Models
2. Dynamic Analysis Models
3. Aeroelastic Analysis Models
4. Heat Transfer Analysis Models
5. Acoustic Cavity Analysis Models

2.2.1 Static Analysis Model Requirements

A static analysis basically requires a good stiffness representation. However, when gravity loading or inertia relief conditions are specified, a good mass representation is also required. This mass representation must include both structural and nonstructural mass distributions. The finite element models for static analysis must consist of the following items as a minium.

1. Geometry - (as appropriate)

   Grid Point Coordinates
   Element Types
   Element Connections
   Coordinate Systems

2. Element Properties - (as appropriate)

   Thicknesses
   Cross-sectional Areas
   Moments of Inertias
   Torsional Constants
   Fiber Orientations
   Other properties as required for special elements.

3. Material Properties - (as appropriate)

Isotropic
Anisotropic
Fiber Reinforced Composites
Temperature Dependent Properties
Stress Dependent Properties
Thermal Properties
Damping Properties
Other properties as required for special problems.

4. Boundary Conditions - (as appropriate)

Single Point Constraints
Multipoint Constraints
Partitioning for Reduction or Substructuring

4. Loading - (as appropriate)

Static Loads
Gravity Loads
Thermal Loads
Centrifugal Loads
Other loading conditions as required for special simulations.

For buckling or nonlinear analysis additional information is required on the following items:

- How the nonlinear matrices are derived.
- The method of solution for the nonlinear problem.
- A description of the method in the case of an eigenvalue analysis.

2.2.2 Dynamic Analysis Models

The dynamic analysis models require i) geometry, ii) element properties, iii) material properties, and iv) boundary conditions as described for the static case. In addition an accurate nonstructural mass and damping representation is required. Generally five types of dynamic analysis are contemplated.

- Normal Modes Analysis or
- Complex Eigenvalue Analysis
- Frequency Response Analysis
- Transient Response Analysis
- Random Response Analysis

In the first two cases only the method of eigenvalue analysis and the frequency (modes) range of interest need be specified. For frequency response analysis the frequencies of interest must be specified. For transient response analysis the dynamic: load must be defined as a function of time or must be provided as tabular values. For random response analysis the statistical nature of the input (such as PSD, Auto Correlation) and the statistical quantities of the output desired must be specified. In addition all the information on dynamic reduction and/or modal reduction must be specified.

2.2.3 Aeroelastic Models

An aeroelastic analysis requires mathematical models of the structure and the aerodynamics. The structure is generally represented by finite element models (FEM). The requirements for the structures models are as specified under static and dynamic analysis. They include mass, stiffness and damping representation. Both structural and nonstructural mass distributions shall be included in the mass model. The aerodynamic models are generally based on paneling or equivalent methods. The requirements of the aerodynamic models are those of the panel geometry which covers all the lifting surfaces including the control surfaces, the empennage (horizontal and vertical tails) and canard surfaces. The fuselage slender body and interference panels shall be modeled to represent the flow-field adequately. The altitude (air density), mach number and other relevant aerodynamic parameters must be specified. The details of the aerodynamic theory and the limits of its validity must be clearly defined. In addition, data for the force and displacement transformations from the structural grid to the aerodynamic grid (and vice versa) shall be included in the aeroelastic models. Two types of aeroelastic analysis are contemplated. Both deal with the phenomenon of aeroelastic stitbility. The real eigenvalue analysis is the basis for determining the static aeroelasti-cstability. There are a number of methods for determining dynamic aeroelastic stability (flutter analysis), and the details of the method (references) and the necessary data shall be provided with the models. Flutter analysis is generally an iterative process and can also involve more than one flutter mechanism. There are often special techniques associated with the flutter analysis, and they can be defined in terms of the ranges of the aerodynamic: parameters. Such data shall be included in the aeroelastic models. In addition, provisions must be made to include the effects of the rigid body modes on the flutter model (body freedom flutter). If it is anticipated that these models will be used for aeroservoelastic analysis, then the data shall be provided for a state space formulation. Also sensor actuator locations and their range of operation and/or limitations shall be included in the data. In addition, a flight control system block diagram shall be provided with sufficient information to define all transfer functions and gains using S-domain variables for analog systems or 2-domain variables for digital systems. The units of important parameters shall be provided.

2.2.4 Heat Transfer Analysis Models

There are three elements to heat transfer models: the heat conducting medium, the boundary conditions and the heat sources and/or sinks. The data requirements for the heat conducting medium are similar to those defined for static and dynamic analysis. For instance the geometry definition includes the grid point coordinates, element types, element connections and coordinate systems. Elements can be classified into volume heat conduction and surface elements. The element type designation for the volume heat conduction element is generally derived from the degree of approximation of its shape functions. The surface elements are used to model a prescribed heat flux, a convective flux due to the difference between the surface temperature and the recovery temperature or local ambient temperature, and radiation heat exchange. Appropriate material properties, single point and multipoint boundary conditions and description of the heat sources (applied forces) have a similar correspondence in the static and/or dynamic analysis. The surface heat convection or radiation details shall be provided (through surface elements) as appropriate. The response variables in heat transfer analysis are generally grid point temperatures or the temperature gradients and heat fluxes within the volume heat conduction elements and the heat flow into the surface elements. Four types of heat transfer analysis are contemplated:

1. Linear Steady-State Response Analysis
2. Linear Transient Response Analysis
3. Nonlinear Steady-State Response Analysis
4. Nonlinear Transient Response Analysis

It is often necessary to adopt special techniques for obtaining stable solutions, particularly in the last two cases. The data pertaining to these special techniques and the limitations of the nonlinear algorithms shall be fully identified.

2.2.5 Acoustic Cavity Analysis Models

Basically there are three elements in acoustic cavity analysis models: the acoustic medium, the boundaries, and the sources of excitation. The acoustic medium model shall consist of grid points and acoustic elements connecting these grid points. The response variables are generally the pressure levels and the gradients of the pressures (with respect to the spatial variables) at the grid points. So for a general three dimensional acoustic analysis there will be four degrees of freedom per node (corresponding to four response variables) in an acoustic-medium model. The properties of the acoustic medium can vary with the temperature and pressure distribution and density. The boundaries of the acoustic model can be solid walls, flexible walls, openings in the walls and walls with acoustic material which can be represented as a complex acoustic impedance. For complicated boundary conditions separate finite element models may be necessary in order to derive the boundary conditions for the acoustic model. These finite element models are based on solid mechanics and their data requirements are similar to those described for the static and dynamic analysis earlier. The acoustic excitation source model shall have information on the spatial distribution and the statistical properties (in terms of the frequency content) of the noise. For a deterministic case, however, definition of the forcing function includes the magnitude, phasing and frequency along with the spatial distribution. The acoustic excitation is generally given as velocity or pressure applied to the medium over prescribed surfaces or at gird points. If the disturbance is from mechanical sources, separate finite element models of the sources shall be supplied as required. These models are also generally solid mechanics models and their requirements are similar to static and dynamic analysis models. Generally three types of acoustic analysis are contemplated.

- Eigenvalue Analysis
- Steady-State Solution
- Nonlinear-Analysis

In the eigenvalue analysis the acoustic frequencies and mode shapes are determined. The purpose is to compare the natural frequencies of the cavity with those of the forcing function and estimate the resonance effects, and to compare the natural frequencies to the resonant frequencies of any structure which may be placed in the cavity. This analysis provides useful information for design changes in the cavity either by altering the overall dimensions or by introducing noise suppression mechanisms such as baffles or by adding noise suppression material to introduce acoustic wall impedance. This analysis (does not require explicit definition of the forcing function. The steady-state solution gives the response of the cavity to a given excitation. This analysis can be in the time or frequency domain. The nonlinear analysis involves an iterative solution when the properties of either the cavity or the acoustic medium vary significantly with the pressure levels and/or temperature.

2.3 Other Requirements

In addition to the input data a summary of output results (such as deflections, stresses, frequencies, etc. at critical areas) shall be provided for future validation of the models. Also a brief description of how these results were used to satisfy a specific design criteria.

3. Criteria for Model Definition

The criteria for model definition is discussed in the context of design optimization using software systems such as ASTROS. Similar observations are valid in the case of just an analysis as well.

The designer is often in a quandry as to what level of detail (finite element model) is appropriate for optimization in preliminary design. The distinction between global and local (detailed) models is the key to answering this question. The global models are intended for predicting the overall response such as the stiffness (displacements), the dynamic characteristics (such as natural frequencies), the static and dynamic aeroelastic response, etc. The stress information derived from such models is approximate and represents only an average over a region. In spite of this limitation it is not advisable to include the details of cutouts, connections, exact-stiffener configurations etc. in the global models. Although these details are important in structural life prediction, their effect on the overall response is not significant enough to justify their inclusion in the global :models. Moreover, their presence disrupts the overall load path selection in optimization and produces unrealistic (undesirable) sculptured designs. The local models, on the other hand, are more appropriate for predicting the effects of stress concentration, and fatigue and fracture properties, etc. However, these details cannot be ignored in preliminary design, because optimization becomes counter productive if the design requires significant modification afterwards. The purpose of this paper is to outline a procedure to account for the local effects in a global design optimization by modifying the strength criteria. As an example, the procedure is outlined in the context of the local buckling of metal and composite panels. Similarly the impact damage in composites and fatigue and fracture in metals can be brought into the preview of the definition of strength criteria.

3.1 Panel Buckling Analysis

The mathematical model of an airframe structure consists of a number of panels. Each of these panels is modeled by one or more finite elements. Optimization of the structure using a finite element model requires the definition of a strength constraint. The object is to define this constraint in such a way that the panel does not buckle under the service load. Two aspects of buckling are excluded from this discussion. A detailed buckling model is considered impractical in a global optimization model. Similarly, allowing for buckling and possible reconfiguration of the load paths for examining post buckling behavior is not a part of this discussion.

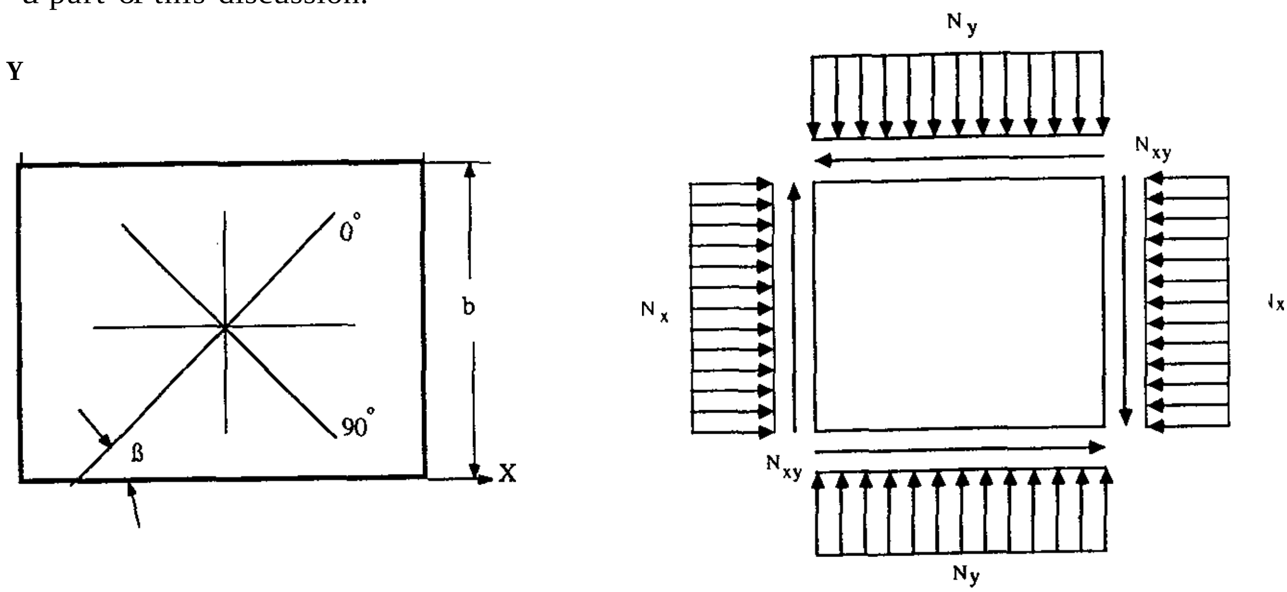

Fig. 1: Rectangular Panel and Loading

An anisotropic rectangular plate supported on one or more sides and subjected to the loading in Fig. 1 is likely to buckle. The buckling condition is governed by the partial differential equation[2,3]

$$D_{11}\frac{\partial^4 w}{\partial x^4} + 4D_{16}\frac{\partial^4 w}{\partial x^3 \partial y} + 2(D_{12} + 2D_{66})\frac{\partial^4 w}{\partial x^2 \partial y^2} + 4D_{26}\frac{\partial w}{\partial x \partial y^3} + D_{22}\frac{\partial^4 w}{\partial y^4}$$

$$= N_x\frac{\partial^2 w}{\partial x^2} + 2N_{xy}\frac{\partial^2 w}{\partial x \partial y} + N_y\frac{\partial^2 w}{\partial y^2} \tag{1}$$

where w is the transverse displacement of the plate when it buckles. N_x, N_y and N_{xy} are the stress resultants at the edges. The coefficients D are the elements of the flexural rigidity matrix shown in the relation

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \\ \hline M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \left(\begin{array}{ccc|ccc} A_{11} & A_{12} & A_{16} & B_{11} & B_{12} & B_{16} \\ A_{12} & A_{22} & A_{26} & B_{12} & B_{22} & B_{26} \\ A_{16} & A_{26} & A_{66} & B_{16} & B_{26} & B_{66} \\ \hline B_{11} & B_{12} & B_{16} & D_{11} & D_{12} & D_{16} \\ B_{12} & B_{22} & B_{26} & D_{12} & D_{22} & D_{26} \\ B_{16} & B_{26} & B_{66} & D_{16} & D_{26} & D_{66} \end{array}\right) \begin{Bmatrix} \epsilon_{x0} \\ \epsilon_{y0} \\ \epsilon_{xy0} \\ \hline k_x \\ k_y \\ k_{xy} \end{Bmatrix} \tag{2}$$

The mid-surface strains, ϵ, and the curvatures, K, are related to the stress and moment resultants through the A, B and D matrices and their elements are given by

$$A_{ij} = \int_{-\frac{h}{2}}^{\frac{h}{2}} Q_{ij}\, dz \tag{3}$$

$$B_{ij} = \int_{-\frac{h}{2}}^{\frac{h}{2}} Q_{ij}\, z\, dz \tag{4}$$

$$D_{ij} = \int_{-\frac{h}{2}}^{\frac{h}{2}} Q_{ij}\, z^2\, dz \tag{5}$$

The Q matrix relates the stresses and strains through the generalized Hooke's Law.

The relationship between the stress resultants and the strains is through the A matrix, while the moment resultants are related to the curvatures through the D matrix. The coupling between stretching and bending is through the matrix, B. For a symmetric laminate B is a null matrix which signifies uncoupled behavior The phenomenon of buckling **as** discussed in this paper is only relevant in the latter case. Only the elements of the D matrix are relevant in determining the buckling loads on the plate. For a layered composite plate computation of the D matrix involves integration over the thickness of the individual layers and the transformation of the elastic constants matrices to a common reference axis. Point stress analysis programs such as SQ 5[4] have such a facility, or they can be generated with relative ease.

When the principal directions of the symmetric laminate (directions of the maximum and minimum stiffness) coincide with the axis of the rectangular panel (parallel to the edges), the flexural rigidity coefficients D_{16} and D_{26} would be zero and equation **1** reduces to the well known biharmonic equation

$$D_{11}\frac{\partial^4 w}{\partial x^4} + 2(D_{12} + 2D_{66})\frac{\partial^4 w}{\partial x^2 \partial y^2} + D_{22}\frac{\partial^4 w}{\partial y^4} = N_x\frac{\partial^2 w}{\partial x^2} + 2N_{xy}\frac{\partial^2 w}{\partial x \partial y} + N_y\frac{\partial^2 w}{\partial y^2} \tag{6}$$

For an isotropic plate, the coefficients are given as

$$D_{11} = D_{22} = D_{12} + 2D_{66} = \frac{Et^3}{12(1-\nu^2)} \tag{7}$$

There are a number of methods available for the approximate solution of Eq. **1** in order to obtain the buckling load. Galerkin's method, **a** series solution, and the finite difference method are appropriate methods for the solution of the differential equation. Rayleigh-Ritz and the finite element method are also appropriate when the formulation starts with the potential energy of the system. The aspect ratio, the boundary (support) conditions of the plate, the thickness to one of the sides ratio and the material mechanical properties are factors affecting the buckling load. In the case of layered composite plates the stacking sequence of various ply orientations and their percentages also affect the magnitude of the buckling load. For a given aspect ratio and boundary conditions, the buckling load is a function of the elements of the D matrix.

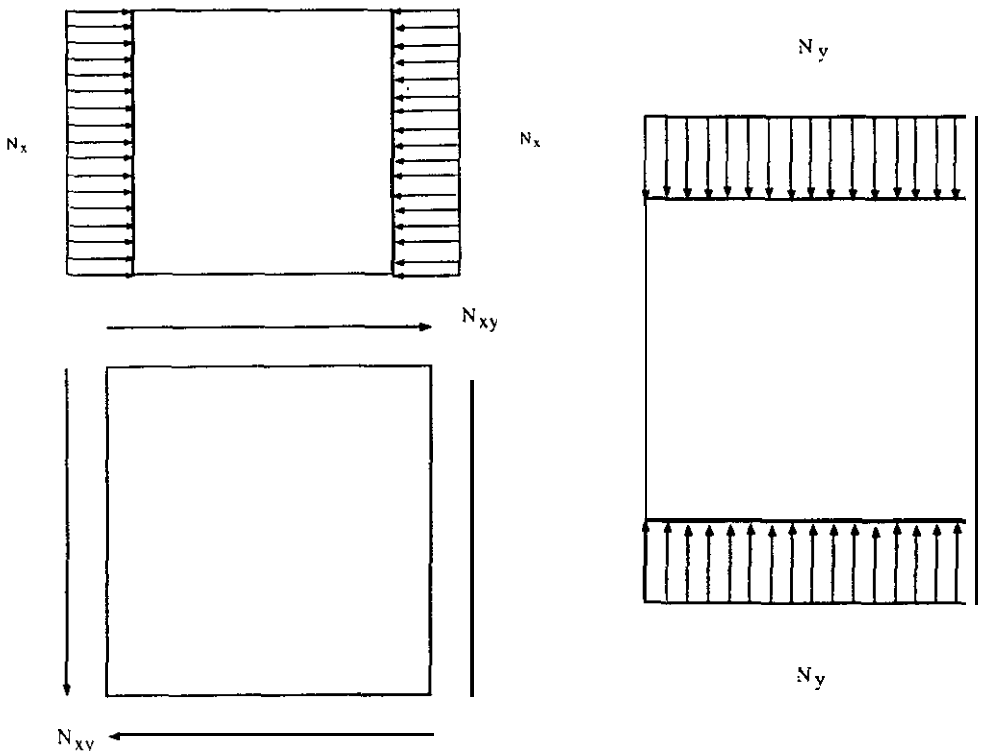

Fig. 2: Independent Edge Loads

The solution of Eq. **1** can be obtained for separate edge loads as shown in Fig. 2 and then they can be superimposed in the sense of an approximate interaction formula discussed in the next section. Alternatively, a combined solution can be obtained by assuming a proportional change in $N,$, N_y and N_{xy}. The proportionality relationship is derived from the global finite element analysis (discussed in the next section).

3.2 Relationship Between the Global Analysis and Panel Buckling

Analysis by programs such as ASTROS and NASTRAN is based on finite element models. The panel in such a model consists of one or more finite elements covering the area between major stiffening components such as spars and ribs or longerons and transverse frames, etc. Panel buckling analysis is generally based on idealized square or rectangular plates, subject to uniformly distributed edge forces as in Fig. **1.** The panels in real structures are not necessarily idealized rectangular plates, and also the internal loads from the finite element analysis (the displacement method in particular) do not come as distributed edge forces. The natural element force output of a finite element analysis is a set of discrete nodal forces or average stresses in the elements. It is necessary to make some approximations in order to establish a correspondence between the finite element model and the usual panel buckling analysis. An alternative to this procedure is to make a panel buckling analysis using local (refined) finite element models. Panel buckling analysis using a finite element model is an unnecessary complication and involves needless expense in terms of modeling time and computational cost.

A procedure for deriving an approximate rectangular panel from a non-rectangular panel is shown in Fig. 3.

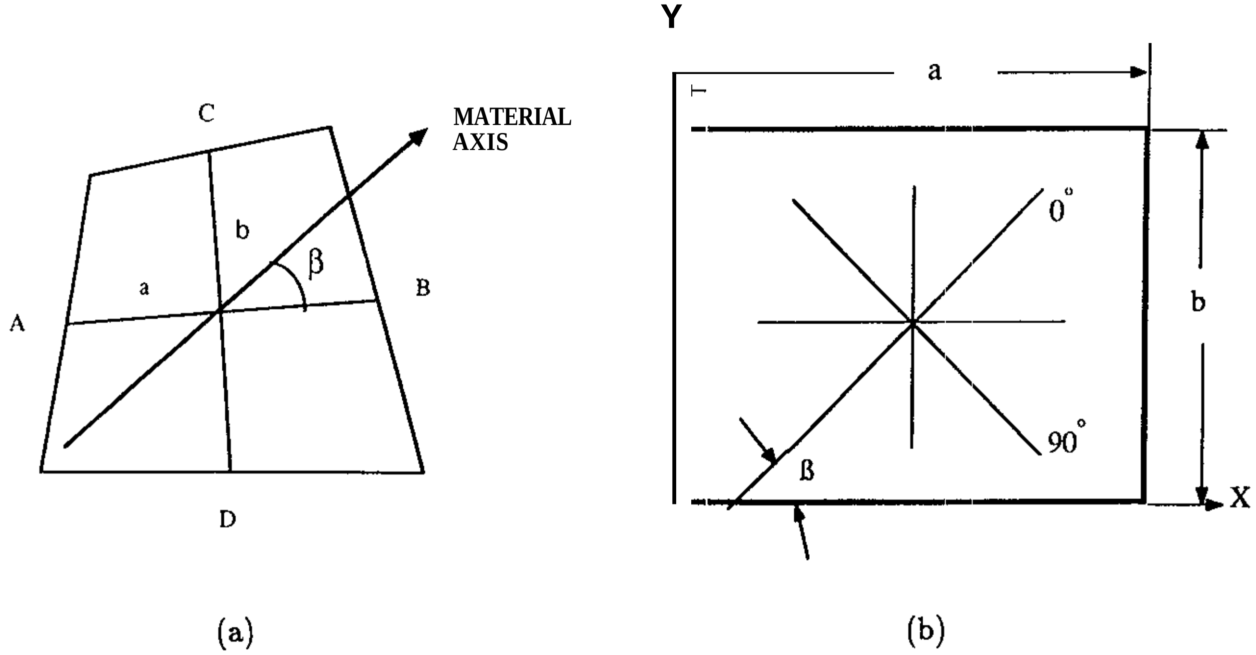

Fig. 3: Panel Idealization

The points **A,** B, C and D are the mid-points of their respective side. The longest of the two lines **AB** and CD is designated as the panel length, “a”, and the shorter length is then “b”. The approximate rectangular panel with sides a and b is shown in Fig. 3b. The quality of this approximation deteriorates as the angle between the two lines **AB** and CD deviates from 90^o. The new reference axes for the panel is the line **AB** and its perpendicular line.

From the finite element analysis, the averagestrains in the quadrilateral panel (Fig. 3a) can be obtained in the element local coordinate system, and they are designated by the strain matrix, ϵ. This strain matrix can be transformed to ϵ' with respect to the new coordinate system by using the relationship (Molir’s circle transformation)

$$\epsilon' = r\epsilon \tag{8}$$

where r is the rotational transformation matrix.

The corresponding approximate edge forces can be determined from the relation

$$\begin{pmatrix} N_{x'} \\ N_{y'} \\ N_{xy'} \\ N_{xy'} \end{pmatrix} = \begin{pmatrix} A_{11} & & A_{13} \\ A_{12} & A_{12} & A_{23} \\ A_{13} & & A_{33} \\ A_{13} & A_{23} & A_{33} \end{pmatrix} \begin{pmatrix} \epsilon_{x'} \\ \epsilon_{y'} \\ \epsilon_{xy'} \\ \epsilon_{xy'} \end{pmatrix} \tag{9}$$

for a layered composite plate the matrix A is the same as that indicated in Eq. **2.**

The buckling loads for the three separate edge loads can be expressed as a function of the elements of the matrix D. The three buckling load designations are shown in Fig. 2. The effective stress-ratio corresponding to the buckling of the panel can be written as

$$ESR = \left(\frac{N_{x'}}{\bar{N}_{x'}}\right) + \left(\frac{N_{y'}}{\bar{N}_{y'}}\right) + \left(\frac{N_{x'y'}}{\bar{N}_{x'y'}}\right) \tag{10}$$

The stress resultants $N_{x'}, N_{y'}$ and $N_{x'y'}$ are determined from Eq. 9, and the critical buckling stress resultants &%, $\bar{N}_{y'}$ and &$'_{x'y}$, for the three edge loads in Fig. 2 are determined from the panel buckling formulas available in the literature.

The ESR computation in Eq. 10 would not be valid when one or two of the buckling loads in the denominator are very small compared to the largest buckling load. In such cases the terms corresponding to the lower buckling load should be neglected.

If the value of the ESR is greater than 1, the panel is assumed to be buckled and otherwise not. If the panel has buckled, then the new strength constraint for the panel can be defined as

$$X_{\text{new}} = \frac{X_{\text{old}}}{\text{ESR}} \tag{11}$$

where X_{OLD} is the strength allowable defined in the previous optimization **run.** This process of redefining the strength allowable is repeated for all the panels. Then the optimization is repeated with the new strength allowables.

This process becomes even simpler for metal panels, since the *A* and D matrices can be written as explicit functions of the two elastic constants and the thickness of the plate. Simple panel buckling formulas are available for metal panels with a variety of edge boundary conditions and aspect ratio values. The column buckling formulas are even simpler. They can be checked in a similar fashion, and strength constraints can be redefined if necessary.

The procedure outlined here for checking the local buckling effects is quite simple and can be implemented with relative ease in a standard finite element analysis.

A similar procedure will be presented in Ref. 5 to account for the requirements of fatigue, fracture, and crack propagation. Basically, this procedure outlines a simple way to define the strength constraints based on durability and damage tolerance requirements.

The essential point of this section is to promote the concept of separate global and local models in the interest of analysis reliability. The communication between the models can be established indirectly. More details of this approach are forthcoming in Ref. 5.

4. Certification of Commercial Software

The concept of software certification arouses serious controversy, and the commercial developers will strongly resist third party intervention. Fear of abridgement of their prerogatives and the bureaucracy generally associated with any certification process are the underlying cause for this aversion. Their argument is that natural selection in an open market environment well weed out the substandard codes. Although there is a strong case for this argument, it takes time, and often damage is done before this happens. It is not unreasonable to demand from commercial vendors that they spend at least a fraction of their marketing effort in identifying the limitations of their codes. Controversy is only part of the impediment to certification. A more difficult issue is how and who is qualified to do such certification. Technical Societies are probably best qualified to address this issue.

5. Benchmarking for Validation

The purpose of benchmarking is to develop standard test cases for verification of new analysis methods and/or software systems. The test cases should be designed to highlight the basic physics of the system and its environment and not the special features of a particular software system. Some of the AIAA coinmittees, with the oupport of the research laboratories, the universities and the aerospace industry, are interested in the development of benchmark problems.

6. Conclusions

Analytical certification offers the best opportunity for reducing development costs and improving the quality of the system. The ongoing efforts by the technical societies and the research laboratories should yield effective guidelines for increasing the role of analysis in product development.

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NONLINEAR ANALYSIS OF COMPOSITE SHEAR WEBS WITH HOLES AND CORRELATION WITH TESTS

by

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SUMMARY

Since the use of composite materials is largely extended in aircraft construction, it is necessary to develop analytical calculations to avoid the present dependency on structural tests.

With that objective a test plan of 25 representative specimens of 9 different shear webs geometries with accessing, inspectioning or lightening holes, enveloping the A320 tailplane design, has been performed, including simulated defects and impacts for two environmental conditions.

Finite element linear and nonlinear analysis, using a very refined mesh, has been performed to correlate test results. Very good correlation has been found even in the post buckling behaviour of the structure

This analysis allows the prediction of the postbuckling capability of these structural elements, and the derivation of a failure criteria.

1. INTRODUCTION

As a part of the development tests for designing the Airbus 320 Horizontal Tailplane completely made in carbon fiber, tests were performed in 1.985 and 1.986 on representative specimens of structural elements, designed to withstand in-plane shear loads only, and containing large holes for accessing, inspectioning or lightening of structure. These structural elements are spar webs and ribs (see Fig. 1.1).

The philosophy for designing spar webs and ribs With large holes is different because spar webs are highly loaded structural elements whilst ribs are low loaded. Therefore, to design ribs it is sufficient to develop buckling criteria though incontrastto design spars webs, it is necessary to develop strength and buckling criteria.

On the other hand, for designing both spar webs and ribs, environmental conditions and damage tolerance must be considered.

The main objectives of these tests were:
1. To determine the validity of stress/strain analysis procedures taking into account environmental conditions and damage tolerance.
2. To derive failure criteria for both structural elements taking into account environmental conditions and damage tolerance.

Tests performed on spar web specimens show a very important nonlinear behaviour in the vicinity of the buckling load. Analysis performed with finite element models (F.E.M.) and analytical calculations show good agreement even in non-linear zone. Furthermore, it has been possible to develop a failure criteria considering strength and stability aspects for these elements.

On the other hand, tests performed on rib specimes show a good correlation with buckling load predicted by F.E.M. allowing derivation of failure criteria from buckling load.

Therefore, the studies allow the design of these types of structural elements without the need to perform more structural tests.

2. TEST DETAILS

SPECIMENS DESCRIPTION:

The test specimens were defined for the 3 most critical areas of both front and rear spars, and for 3 ribs, summing 25 specimens, summarized in table 2.1.

They were made from Hexcel T300/F593 graphite/epoxy plain woven cloth. Its properties are listed in table 2.2.

Geometries, thicknesses and ply layups of these specimens are shown in figures 2.1 to 2.9. In these figures it can be seen that specimes, representing front and rear spars have a plain hole and those of the ribs have a flanged hole, which was expected to give the shear panel better buckling resistance. It can be seen in fig. 2.9 that R10 have a special design with cut-outs which allow the continuity of the skin stiffen ers in the real structure; therefore, the existing geometric stress concentration in those cut-outs, has been considered with these representative specimes although this design aspect is nos handled in this document.

All web layups are symmetric laminates and therefore specially orthotropic.

Some specimens have stiffeners which were designed to give simple support condition to the webs. The stiffeners are integral co-cured parts with the web.

LOADING AND SUPPORT:

Specimens RS1213 and FS1112 (according to table 2.1) were defined as a beam loaded in 3 - point bending in which the web was the actual test specimen (see Fig. 2.10).

The rest of the specimens were defined to be loaded in a picture frame as is shown in figure 2.11.

The choice of text fixture was determined by specimen aspect ratio. Specimens with aspect ratio greater than 3.0 were tested as beams in order to prevent load introduction problems such as concentration in the corners.

Loading of the picture frame was performed by applying tension to diagonally opposite corners, using a hydraulic jack. The beam specimens were simply supported at both ends and loaded at, or near, the center by a single hydraulic jack as is shown in figure 2.10.

INSTRUMENTATION:

Specimens were mounted with back-to-back strain gauges located at critical positions. The generalized installation plan is shown in the figures 2.12 and 2.13.

CONOITIONING:

The influence of defects and moisture content was investigated.
Some of the rear spar and rib specimens (RS0203-3,4,5 ; R10-5;6), as is indicated in table 2.1, were fabricated including artificial defects, and impacts were performed at more critical locations.

Figures 2.14 and 2.15 show the introduced damages arid defects.
Furthermore, these specimens were conditioned reach'ng the moisture content corresponding to the equilibrium for 70% RH to achieve its end-of-lifetime moisture content (0.88% normalized to a resin content of 40% by volume) and were tested at 70 ºC (hot/wet condition).

3. PRESENTATION OF TEST RESULTS

SPECIMENS TESTED IN RT/DRY CONOITION YITHOUT DEFECTS:

Figures 3.1, 3.2 and 3.3 show the different behaviours of the specimens tested. They show the variation of axial strain gauge readings versus applied load by the hydraulic jack, according to figures 2.12 and 2.13.
Figure 3.1 presents exactly the results of the specimen RS0203-1 which are similar to the rest of rear and front spar specimens except for RS1213-1, FS1112-1 and FS1112-2. The behaviour of these 3 latter specimens is given by the figure 3.2 which corresponds exactly with the FS1112-2 specimen. The difference between these two behaviours is because in RS1213-1, FS1112-1 and FS1112-2 a point of strain divergence appears clearly as is shown in figure 3.2 located as "estimated buckling load".
Figure 3.3 shows exactly the strain gauge readings of the specimen R7-1 which are similar to the rest of rib specimens. This figure reveals a behaviour similar to figure 3.2 for the gauge readings located near the flanged area of the hole, but in the flat area of the panel (see Fig. 2.13 to locate the strain gauges). The strain gauge readings at the edge of the hole are different since the hole flange does not lie in the same geometric plane as the remaining panel, and the applied shear moving from panel plane to flange plane causes a moment around the flange.
The specimens failure mode is sketched in figures 3.4 and 3.5 which show the crack positions of the panels.
Failure data of rear and front spar specimens are summarized in table 3.1. Table 3.2 gives failure data of rib specimens.
Estimated buckling data, according to figures 3.2 ard 3.3, are included in tables 3.3 and 3.4 for spar and rib specimens, respectively.
It should be emphasized that specimen FS0607-1 did not fail because this test was stopped due to frame problems, and that the beams specimens RS1213-1 and RS1213-2 failed prematurely due to flange crippling and, consequently, did not fail at the hole edge.

SPECIMENS TESTED IN HOT/WET CONDITIONS YITH DEFECTS:

Table 3.1 gives the results for the rear spar specinens. The greatest strength reduction from the RT/DRY without defect tests is approximately 30% but the 'Failure mode and type of behaviour described before did not change. It is not clear whether the higher failure loads of specimens RS0203-4 and RS0203-5 were due to the lower moisture content or the difference between defect types, delaminations or scratches.
The results for specimens R10-5 and R10-6 are summarized in table 3.2. It can be seen that no influence of hot/wet conditons together with defects can be found clearly on the specimen failure load. Futhermore, it can be seen that influence of vertical stiffeners on rib strength is insignificant.

4. ANALYSIS

The analysis have been performed developing analytical calculations and finite element models (F.E.M.), handling linear, buckling and geometric non-linear analys'is.

FINITE ELEMENT MODELS:

Figure 4.1 shows the F.E.M. for the specimen RS0203 (F.E.M. type 1). The F.E.M. shown in figure 4.2 have been used for the rest of rear and front spar specimens, altering the dimensions and thicknesses to represent the different configurations (F.E.M. type 2). Fgure 4.3 shows the F.E.M. for specimen RS0203 modified changing the plain hole by a flanged hole to study this concept of hole design which corresponds to R5, R7 and R10 specimens (F.E.M. type 3). Furthermore, two F.E.M. were performed corresponding to specimens R-10, with and without stiffeners, to study their buckling behaviour which are shown in figure 4.7.
All models consist of cuadrilateral and triangular 2 - dimensional plate elements only. Also, all were constrained representing simple-support condition and were loaded at the edge nodes simulating a uniform shear flow.
Linear and buckling analyses were performed using NASTRAN F.E.M. program, and geometric non linear analyses were done using ARGUS F.E.M. program.

The linear analysis performed on type 1 and 2 F.E.M. reveals maximum concentration of cincunferential stresses at 45º, 135º, 225º and 315º as is shown in figure 4.4. The maximum value of the concentration factor is in the range 6.5 to 7.25 for the indicated F.E.M.. These values were obtained extrapolating results of plate F.E.M. elements near the hole edge. Also, the analysis reveals that the larger the holes, the larger the affected area. However, considering the distance from the hole edge at which the axial stress returns to its nominal value (1.0), it can be seen that this occurs at approximately 50% of the hole diameter. In addition, the distance at which the concentration factor became 2 is aproximately 25% of the hole diameter. Also, this analysis reveals concentrations of in-plane shear stress at 0º, 90º, 180º and 270º, as is shown in figure 4.4. Maximum concentration factor for these F.E.M. is 1.8-2.1. Again, larger holes affect a bigger area. However, the peak values occur at approximately 25% of the diameter for P.E.M. and these concentrations don't decay rapidly. This analysis for the type 3 F.E.M. gives complex results because the hole flange does not lie in the same plane as the rest of F.E.M. and consequently moments appear around the flange. The figure 4.5 shows the different concentrations.

Buckling analysis carried out with type 1, 2 and 3 F.E.M. reveals the buckling modes shapes shown in figure 4.6. Various modes are shown together with its load level factor refered to the first mode. It has been found that the panel with flanged hole has approximately (4% difference) the same buckling load as the same panel without hole, whereas the panel with plain hole shows 40% reduction in the buckling load level as respect the same panel without hole. As the panel with flanged hole is not symmetric, it could be expected to obtain different results depending on the sense of the load but results have shown that the buckling load level is approximately the same (3% difference). Also the mode shape is the same, as can be seen in the figure 4.6.

Buckling analysis performed with F.E.M. corresponding to Rib-10 specimens with and without stiffeners reveals the buckling mode shapes shown in figures 4.7. It can be seen that F.E.M. with stiffeners show buckle wave on the hole, whilst F.E.M. without stiffeners show buckle waves between holes.

The buckling analysis is needed in order to specify properly the load increments for the geometric non-linear analysis, to avoid numerical problems near the buckling load. Geometric non-linear analysis was performed for the F.E.M. corresponding to specimens FS1112, RS0203 and RS0405. The results are presented in detail in chapter 5.

ANALYTICAL CALCULATIONS:

Stress Concentration:

Analytical solution of stress concentration around all elliptical opening in an infinite elastica1 anisotropic plate under shear stresses can be found in chapter 6 of reference 1:

$$\sigma_\theta = -\tau\frac{ab}{l^2}sen2\theta + \frac{\tau}{l^2}Re\{\frac{ie^{-\theta i}}{(asen\theta - \mu_1 bcos\theta)(asen\theta - \mu_2 bcos\theta}$$

$$\cdot[\mu_1 a + \mu_2 a - i\mu_1\mu_2 b)a^3sen^3\theta + (2 - \mu_1\mu_2)a^2b^2sen^2\theta cos\theta +$$

$$+i(1 - 2\mu_1\mu_2)a^2b^2sen\theta cos^2\theta + (a - i\mu_1 b - i\mu_2 b)b^3cos^3\theta]\}$$

where $l^2 = a^2 + b^2$

τ, θ, a and b are given in figure 4.8

μ_1, and μ_2 are the roots of the following equation when the plate is specially orthotropic. as it is in our cases:

$$\mu^4 + \left(\frac{E1}{G} - 2\nu_1\right)\mu^2 + \frac{E1}{E2} = 0$$

Properties relative to laminate principal directions which must coincide with ellipse directions (see Fig. 4.8)

This formula has been computerized. and for our case gives:

$$\sigma_\theta = f_\tau\tau; \qquad |f_\tau| = 5.448$$

and it will be,

$$\varepsilon_\theta = \frac{f_\tau\tau}{E_\theta}$$

In the next chapter we can see the correlation of the tests with this formula. It should be emphasized that these results do not take into account the finite aspect ratio of the panel.

Buckling:

To estimate the buckling shear flow of a specially orthotropic plate with a circular opening loaded uniformly by a shear flow q and simply-supported as it is shown in Fig. 4.9. it is suggested to apply the following expresion:

$$q_{CR} = K1\ K2\ K3\ \frac{t^3}{b^2} \qquad (q_{CR} = q_{BUCKLING})$$

K_1 is a constant for an infinite length panel which is a function of the material and the stacking.
K_2 is a constant to take into account the finite aspect ration a/b and it is a function of the material, stacking and the aspect ratio a/b.
K3 is a constant to quantify the effect of a large centrally located plain hole which is a function of the ratio Ø/b.
K_1 and K_2 can be found applying the energy method given in chapter 14 of reference 1 assuming the following expression, in the form of a series, of the out of plane displacements:

$$w = \Sigma\Sigma A_{mn} \sin\frac{m\Pi x}{a} \sin\frac{n\Pi y}{b}$$

which satisfy all edge conditions, and minimize the potential energy of bending, plus the work done by the shear forces. The solution of this problem has been obtained with numerical methods which give, for K_1 and K_2 for our material, figures 4.10 and 4.11 respectively.
K_3 has been taken from reference 2 which gives:

$$K_3 = (1 - Ø/b)\ (1 - Ø/2b)$$

for our case of all laminate 100% ± 45Q is:

$$K_1 = 188750\ N/mm^2$$

5. ANALYSIS / TEST CORRELATION

The analysis procedures developed in chapter 4 have been used to attempt correlation with the test results.

Analysis/test correlation is presented in figures 5.1, 5.2 and 5.3 for RS0203, RS0405 and FS1112 specimens, which show all spar web specimens behaviour. Moreover, buckling analysis is correlated with Rib10 specimens.

Before commenting this correlation, it should be pointed out that important factors affect the real behaviour which have not been considered in the previous analysis:

1.- The influence of low interlaminar shear stiffness:
This effect is difficult to quantify. It is fundamentally a resin property. So therefore, it will be influenced by environmental conditions and internal defects. It can be important when the plate is relatively weak in the transverse direction and when the plate response is sensitive to the transverse stiffeness as in buckling. More inforination about this factor can be found in the reference 3.

2.- Geometric imperfections:
The difference in behaviour for perfect and imperfect panels is illustrated in figure 5.4, taken from reference 4. This reference provides more details about this factor.

3.- Edge support conditions:
True edge support conditions generally lie between the simple supported and clamped limiting cases.

4.- Eccentricity:
In reality, the aplication of load and its reaction in the panel are not in the same plane, out-of-plane twisting apperaring during test.

Furthermore, we must consider that the strain gauges are located in a position of severe strain gradients and installation position tolerance will be a significant factor.

DISCUSSION:

From Figure 5.1:

1.- Analytical calculations of hole edge strain and buckling of the panel do not agree with the results, because the local reinforcement around the hole, which has this spec'imen, is not large enough for these calculations, which consider a flat panel, to be applied.

2.- The geometric non-linear analysis by which F.E.M. accurately predict the hole edge axial strain. The built-in offset reinforcement that has this specimen clearly drives the non-linearity.

3.- The shear flow buckling predicted with F.E.M. is beyond the point of strain divergence explained in chapter 3.

4.- This specimen cannot reach their theoretical buckling load due to the non-linear compression strain response which cause the ultimated failure.

From Figure 5.2:

1.- Analytical calculations of the hole edge strain appear conservative to predict the average behaviour. However, it has to be considered that strain gauges readings can vary appreciably with its location near the hole and that this prediction does not lake into account the finite aspect ratio of the panel.

2.- The geometric non-linear analysis with F.E.M. predicts quite well the point of strain divergence although it also gives conservative results. Noting that in this specimen, without local reinforcement, geometric non-linear analysis cannot predict the initial non-linear behaviour of the specimen, which would be due to the imperfections or twisting. However, near to the Strain divergence point, geometric non-linear analysis approaches this behaviour quite well because, when, out-of-plane displacements reach sufficient value to be important, these displacements drive the non-linearity, as was shown in figure 5.4.

3.- Analytical calculations of shear flow buckling give slightly conservative values of strain divergence point .

4.- The shear flow buckling predicted with F.E.M. is beyond the strain divergence point. However this predicted value is when strain starts to rise sharply which can be considered a better value of inestability.

5.- This specimen cannot reach their theoretical buckling load.

From Figure 5.3:

1.- Analytical calcultations of the hole edge strain, approach conservatively the average strains near the hole.

2.- These specimens reach and clearly exceed their theoretical buckling load.
The geometric non-linear analysis approaches the strain divergence point very well and shows good correlation within the postbuckling range, despite the initial non-linear behaviour that cannot be predicted.

3.- Analytical calculations of shear flow buckling give conservative value of point of strain divergence and those predicted by F.E.M. is higher than this point but really represent the strain divergence.

From Buckling Analysis with **F.E.H. for** Rib 10 Specimens:

Very good correlation has been found between the estimated buckling load given in table 3.4 from tests for Rib 10-1; 2 specimens with stiffeners and the buckling load found by F.E.M. shown in figure 4.7:

Average Test Result	: 52.8 N/mm
F.E.M. Result	: 49.2 N/mm

On the other hand the buckling load found by F.E.M. for Rib-10 specimens without stiffeners was 34.2 N/mm.

Comparing these results, with and without stiffeners, with its respective failure load, it can be concluded that despite the fact that specimens with stiffeners have higher buckling loads than specimens without stiffeners, these latter specimens have more postbuckling capability. (Approximately 35%). For this reason the ribs of A320 tailplane have not stiffeners.

6. FAILURE CRITERIA

Spar webs without defects and RT/DRY condition: The failure mode of spar webs is rupture in compression in the hole edge due to stress concentration, before or after buckling.

There is a correlation between τ test/τ buck vs tweb shown in Fig. 6.1, which can be adjusted by a linear relationship, line A, taking into account that all test results lie on or above a parallel line which gives 90% of τ test/ τ buck called 0.9 A. This line establishes that panels with tweb less than 3.5 mm. have postbucking capability and panel with tweb greater than 3.5 mm. will not achieve the theoretical buckling load.

In figures 6.2, 6.3 and 6.4 are represented the maximum compression strain at hole edge for test specimens RS0203, RS0405 and FS1112 (which resume all spar web specimens behaviour) vs the percentage ot the panel theoretical buckling load, % q_{CR} which is calculated according to analytical method given in chapter 4 but k3 has been modified correlating with F.E.M. results to give better approach of buckling load. The new formula is:

$$K_3 = .94\ (1 - \varnothing/b)$$

In figure 6.5 those curves are jointed in the range of $-5000\mu\varepsilon$ and $-10000\mu\varepsilon$. It can be seen that type 1 and 2 specimens have very similar behaviour, whereas type 3 specimen is quite distinct. Both type 1 and 2 specimens have web thicknesses greater than 3.5 mm. (ie 8.88 and 5.76 mm. respectively), whereas type 3 specimen has web thickness less than 3.5 mm. (ie 2.4 mm.).

Based on the observed behaviour, a failure criteria has been defined which states that the "usable buckling load" see Fig. 6.6, is dependant upon the material compression strain allowable and upon the panel thickness. It therefore represents a combination of both "strength" and "stability" aspects of panel stress analysis.

Spar webs with defects and hot/wet conditions:

Of the 3 specimens tested with defects and in hot/wet condition (see table 2.1), specimen RS0203-3 had the lowest failure load level. This specimen was fully aged, and had simulated delaminations at each of the maximum stress concentration areas. The hole edge compression strain level is shown as a function of the theoretical buckling load in Figure 6.7. In this case, the theoretical buckling load has been calculated using aged material properties. The hole edge strain of this specimen is shown together with the present failure criteria in figure 6.8. It can be seen that excellent correlation exists. The maximun difference in % q_{CR} for a given value of compression strain is approximately 2%. It can therefore be concluded that the present hole edge strain limitation criteria is valid for the moisture **andteniperaturerequirements.**

Rib Webs:

All specimens tested which are representative of rib designs have presented their failure after reaching buckling load. Therefore, a simple failure criteria based in the buckling load can be derived:
Non-buckling at Ultimate Load.

7. CONCLUDING REMARKS

Results have been presented of tests done on shear webs with two designs of large holes: plain and flanged.

Flanged holes reveal better buckling resistance than plain holes, but they present the most severe stress concentration, therefore, they should be used in shear webs, under low levels of load, designed for buckling requirements.

All specimens present non-linear behaviour. Linear analysis cannot predict this behaviour and consequently non-linear analysis is need to approach accurately that loading response.

However, analytical calculations have been presented to get a preliminary estimation for the average strains at the hole edge and for the buckling load.

Geometric non-linear analysis have been done by finite elements method for "plain holes" and the results reveal very good agreement with the tests near to the buckling load and even in the postbuckling range.

Buckling analysis has been done by the finite element method for both designs, plain and flanged holes, obtaining very good correlation with test results.

Failure criterion have been derived for two design:;.

It should beemphasizedthat the conclusions given 'in this paper can only be considered valid for these particular types of shear panels construction, with geometric characteristics lying within the range of those panels tested and analysed.

8. ACKNOWLEDGMENTS

The work presented in this document was supported by CONSTRUCCIONES AERONAUTICAS, S.A.. The authors would like to acknowledge the technical support provided by Mr. A. E. Davie. Also, services provided by Testing Department, Design Department and Stress Department are acknowledged.

9. REFERENCES

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2.- A. E. Davie; "TORQUE BOX SPARS: BUCKLING ANALYSIS"; Technical Document NO. 35 D 551 80023; C.A.S.A.; 1986.

3.- M. Stein; EFFECTS OF TRANSVERSE SHEARING FLEXIBILITY ON POSTBUCKLIEIG OF PLATES IN SHEAR. AIAA Journal Vol. 27, No. 5; 1988.

4.- NON-LINEAR EQUILIBRIUM OF IMPERFECT, LOCALLY DEFORMED STRINGER STIFFENED PANELS UNDER COMBINED IN-PLANE LOADS. Computer & Structures Vol. 27, No. 4 PP 519-539; 1987.

DESIGN CONCEP	AREA REPRESENTED OF A320 TAILPLANE SPECIMEN CODE	NO. OF SPECIMENS	ARTIFICIAL DEFECTS AND IMPACTS (*)	CONDITIONING (**)	SCHEMATIC FIGURE
REAR SPAR	BETWEEN RIBS 2 AND 3 RS0203-1÷5	2	NO	RT/DRY	Fig. 2.1
		3	YES	HOT/WET	
	BETWEEN RIBS 4 AND 5 RS0405-1÷2	2	NO	RT/DRY	Fig. 2.2
	BETWEEN RIBS 12 AND 13 RS1213-1÷2	2	NO	RT/DRY	Fig. 2.3
FRONT SPAR	BETWEEN RIBS 4 AND 5 FS0405-1÷2	2	NO	RT/DRY	Fig. 2.4
	BETWEEN RIBS 6 AND 7 FS0607-1÷2	2	NO	RT/DRY	Fig. 2.5
	BETWEEN RIBS 11 AND 12 FS1112-1÷2	2	NO	RT/DRY	Fig. 2.6
RIBS	RIB 5 R5-1÷2	2	NO	RT/DRY	Fig. 2.7
	RIB 7 R7-1÷2	2	NO	RT/DRY	Fig. 2.8
	RIB 10 R10-1÷6	4(***)	NO	RT/DRY	Fig. 2.9
		2(****)	YES	HOT/WET	

(*) ACCORDING TO FIGURES 2.14 AND 2.15

(**) RT/DRY : ROOM TEMPERTATURE / DRY (AS RECEIVED)
HOT/WET : 70° / MOISTURE CONTENT : SPECIMEN RS0203-3 : .90%
SPECIMEN RS0203-4 : .71%
SPECIMEN RS0203-5 : .71%
SPECIMEN R10-5 : .88%
SPECIMEN R10-6 : .88%
(DATA NORMALIZED TO A RESIN CONTENT OF 40% BY VOLUME)

(***) R10-1 : 2 WITH STIFFENERS
R10-3 : 4 WITHOUT STIFFENERS

(****) R10-5 : 6 WITHOUT STIFFENERS

TABLE 2.1.- SUMMARY OF TEST SPECIMENS

TEST		VALUES	
TYPE	TEMP./COND.	ALLOW B	MEAN
TENSILE STRENGTH (0°) (N/mm²)	RT/DRY	475	
	HOT/WET	450	
TENSILE MODULUS (90°) (KN/mm²)	RT/DRY	—	52
	HOT/WET	—	52
COMPRESSION STRENGTH (90°) (N/mm²)	RT/DRY	460	
	HOT/WET	450	
COMPRESSION MODULUS (90°) (KN/mm²)	RT/DRY	39	47
	HOT/WET	34	45
ILSS STRENGTH (90°) (N/mm²)	RT/DRY	63	
	HOT/WET	60	
POISSONS RATION (-)	RT/DRY	—	0.05
	HOT/WET	—	0.04
IN PLANE SHEAR STRENGTH (N/mm²)	RT/DRY	104	
	HOT/WET	86	
IN PLANE SHEAR MODULUS (KN/mm²)	RT/DRY	—	3.1
	HOT/WET		2.7

Prepeg areal weight : 345 g/m²
Prepeg resin content : 44% (+/- 2%) (weight)
Fibre areal weight : 193 g/m²
Fibre : T300 3K
Resin : Hexcel F-593
Material designation : W3T-282-42-F593-14
CASA code : Z-19.776

TABLE 2.2.- MATERIAL PROPERTIES

SPECIMEN LABEL	SPECIMEN NUMBER	LOAD (KN)	SHEAR FLOW (N/mm)	SHEAR STRESS (N/mm²)	$\mu\epsilon_C$ (*) $\mu\epsilon_{CBIS}$ $\mu\epsilon_{CAV}$	$\mu\epsilon_T$ (**) $\mu\epsilon_{TBIS}$ $\mu\epsilon_{TAV}$	REMARKS
RS0203	1	403.34	647.2	72.9	- 12000 - 4500 - 8250	8100 7900 8000	
	2	389.50	625.0	70.4	- 11500 - 4900 - 8200	- - -	
	3	272.93	438.0	49.3	-	-	WITH DEFECTS
	4	299.64	475.4	53.5	-	-	
	5	363.39	583.1	65.7	-	-	
RS0405	1	392.59	577.5	100.3	- 10200 - 6300 - 8250	9300 8900 9100	
	2	401.87	591.1	102.6	- 10000 - 6700 - 8350	9500 9500 9500	
RS1213	1	66.84	163.9	68.3	- 8700 - 2900 - 5800	7000 5500 6250	
	2	61.64	151.7	63.2	- 7000 - 6000 - 6500	5000 3600 4300	
FS0405	1	333.94	491.8	97.6	- 10300 - 5600 - 7950	7300 6200 6750	
	2	293.61	432.3	85.8	- 9600 - 7100 - 8350	11100 8600 9850	
FS0607	1	238.39	374.0	74.2	- 10800 - 3800 - 7300	9900 8900 9400	
	2	295.23	463.2	91.9	- 9300 - 8000 - 8650	10200 7100 8650	
FS1112	1	40.62	158.5	73.4	- 20400 3200 - 8600	8200 -	
	2	38.22	149.4	69.2	- 15531 600 - 7400	8800 -	

(*) $\mu\epsilon_C$: MAXIMUN COMPRESSIVE STRAIN FOUND IN ANY OF THE HOLE EDGE GAUGES.
$\mu\epsilon_{CBIS}$: STRAIN IN REVERSE FACE GAUGE CORRESPONDING TO $\mu\epsilon_C$.
$\mu\epsilon_{CAV}$: AVERAGE STRAIN. $(\mu\epsilon_C + \mu\epsilon_{CBIS})/2$.

(**) $\mu\epsilon_T$: MAXIMUN TENSION STRAIN FOUND IN ANY OF THE HOLE EDGE GAUGES.
$\mu\epsilon_{TBIS}$: STRAIN IN REVERSE FACE GAUGE CORRESPONDING TO $\mu\epsilon_T$.
$\mu\epsilon_{TAV}$: AVERAGE STRAIN. $(\mu\epsilon_T + \mu\epsilon_{TBIS})/2$.

TABLE 3.1 - FAILURE DATA OF BOTH REAR AND FRONT SPAR SPECIMENS

SPECIMEN LABEL	SPECIMEN NUMBER	LOAD (KN)	SHEAR FLOW (N/mm)	SHEAR STRESS (N/mm²	REMARKS
RIB5	1	41.45	72.5	50.4	
	2	44.10	77.1	53.5	
RIB7	1	78.30	127.7	75.5	
	2	68.60	111.9	66.6	
RIB10	1	56.35	93.4	55.6	WITH STIFFENERS
	2	61.05	101.2	60.2	
	3	62.18	103.1	61.4	WITHOUT STIFFENERS
	4	59.72	99.0	58.9	
	5	55.81	92.6	55.1	WITHOUT STIFFENERS WITH DEFECTS
	6	64.72	107.3	63.9	

TABLE 3.2.- FAILURE DATA OF RIB SPECIMENS

SPECIMEN LABEL	SPECIMEN NUMBER	LOAD (KN)	SHEAR FLOW (N/mm)	SHEAR STRESS (N/mm²)	$\mu\epsilon_C$(*) $\mu\epsilon_{CBIS}$ $\mu\epsilon_{CAV}$	$\mu\epsilon_T$(**) $\mu\epsilon_{TBIS}$ $\mu\epsilon_{TAV}$	REMARKS
RS1213	1	56.35	138.2	57.6	-6500 -3000 -4750	4600 5700 5150	
FS1112	1	29.40	115.00	53.2	-7400 -3000 -5200	4400 -	$\phi = 50$ mm
	2	26.95	105.35	48.8	-7200 -2600 -4900	5800 -	

REST OF SPECIMENS DO NOT SHOW BUCKLING BEHAVIOUR

(*)(**): LIKE TABLE 3.1

TABLE 3.3 - ESTIMATED BUCKLING DATA OF BOTH REAR AND FRONT SPAR SPECIMENS

SPECIMEN LABEL	SPECIMEN NUMBER	LOAD (KN)	SHEAR FLOW (N/mm)	SHEAR STRESS (N/mm²)	REMARKS
RIB 5	1	14.71	25.7	17.85	
	2	INSUFFICIENT STRAIN GAUGE DATA			
RIB 7	1	34.32	56.0	33.34	
	2	INSUFFICIENT STRAIN GAUGE DATA			
RIB 10	1	29.37	48.7	28.98	WITH STIFFENERS
	2	34.29	56.9	33.87	
	3	47.06	INSUFFICIENT STRAIN GAUGE DATA		WITHOUT STIFFENERS
	4	50.96			
	5	46.05			WITHOUT STIFFENERS WITH DEFECTS
	6	41.15			

TABLE 3.4.-ESTIMATED BUCKLING DATA OF RIB SPECIMENS

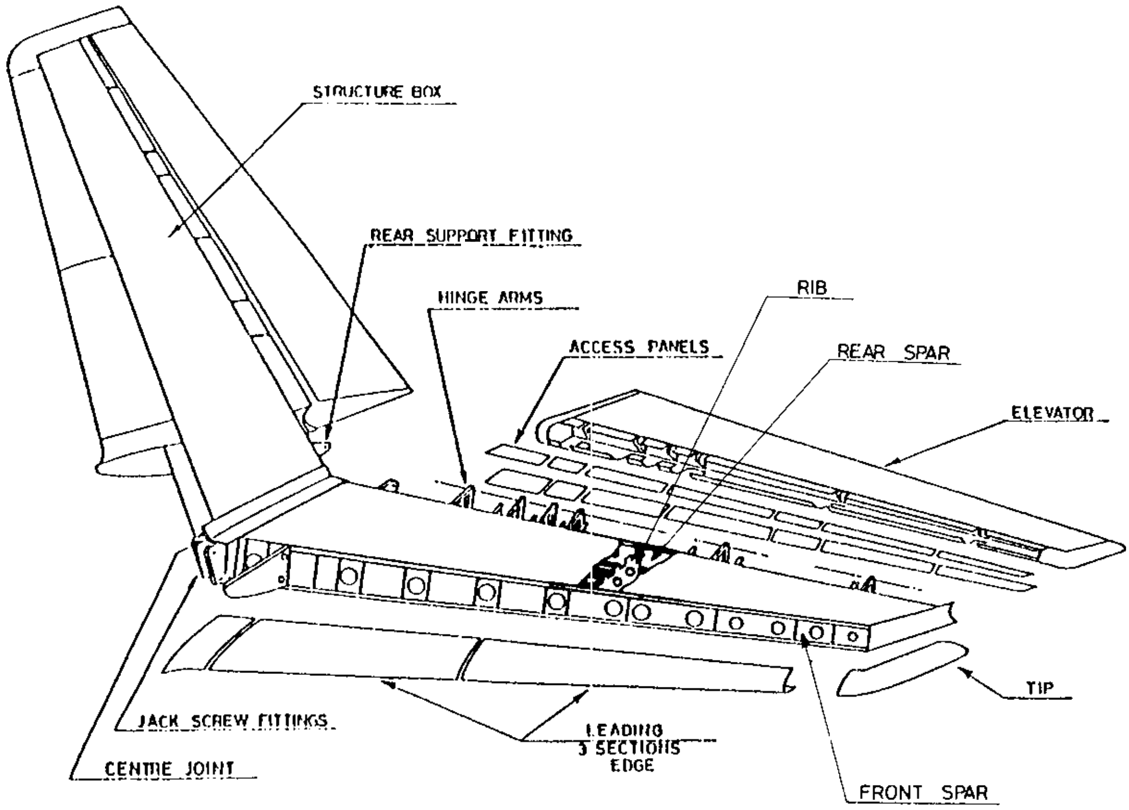


FIGURE 1.1.- A320 HORIZONTAL TAILF'LANE GENERAL DESCRIPTION

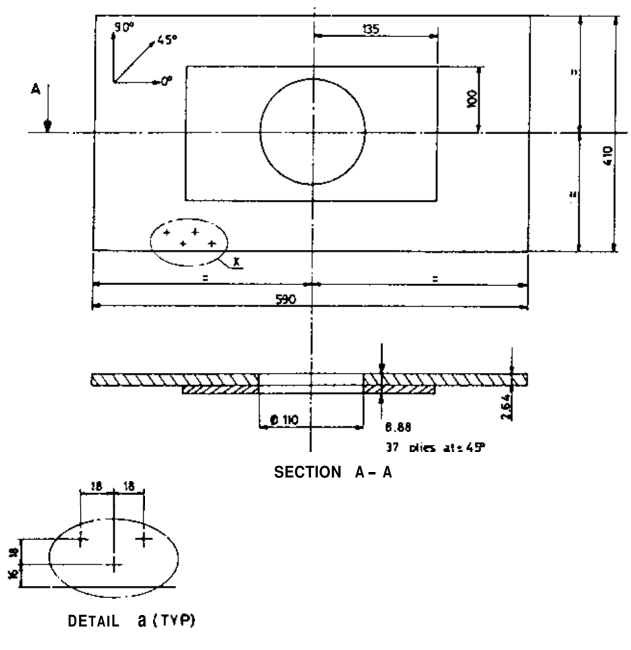


FIGURE 2.1.- SPECIMEN RS0203

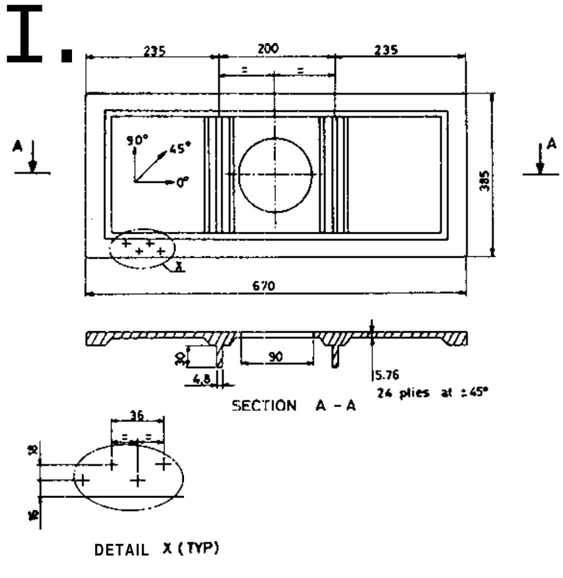


FIGURE 2.2 .- SPECIMEN RS0405

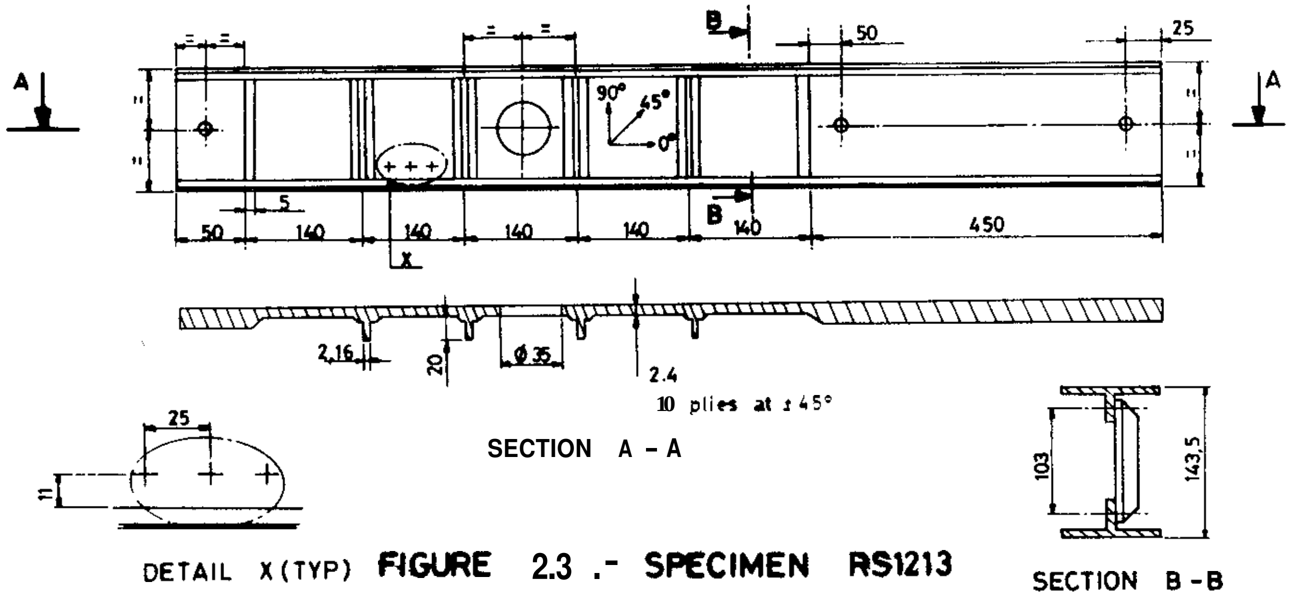


FIGURE 2.3 .- SPECIMEN RS1213

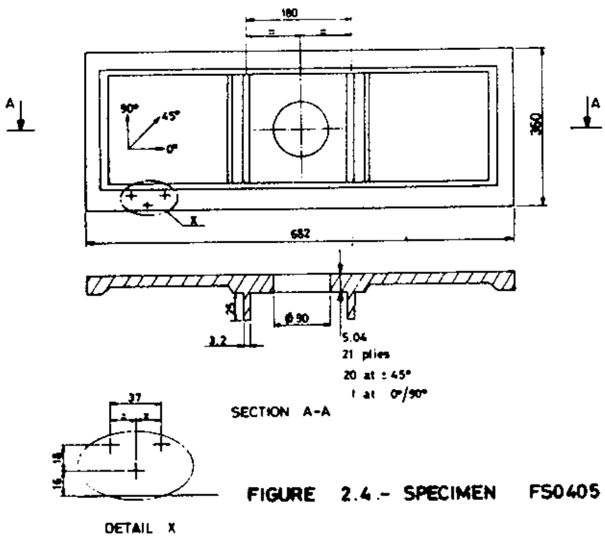


FIGURE 2.4 .- SPECIMEN FS0405

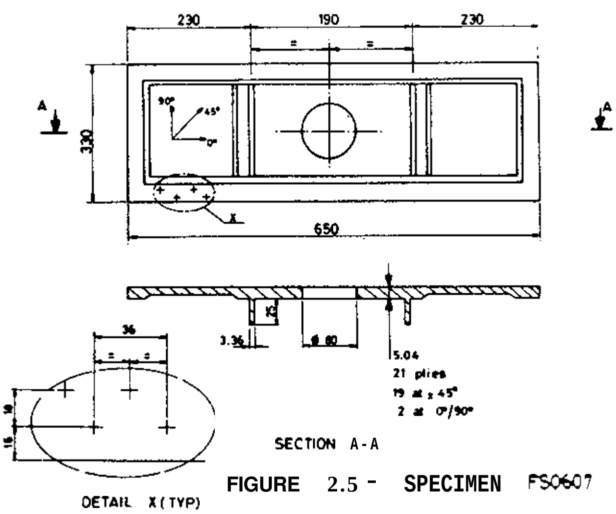


FIGURE 2.5 - SPECIMEN FS0607

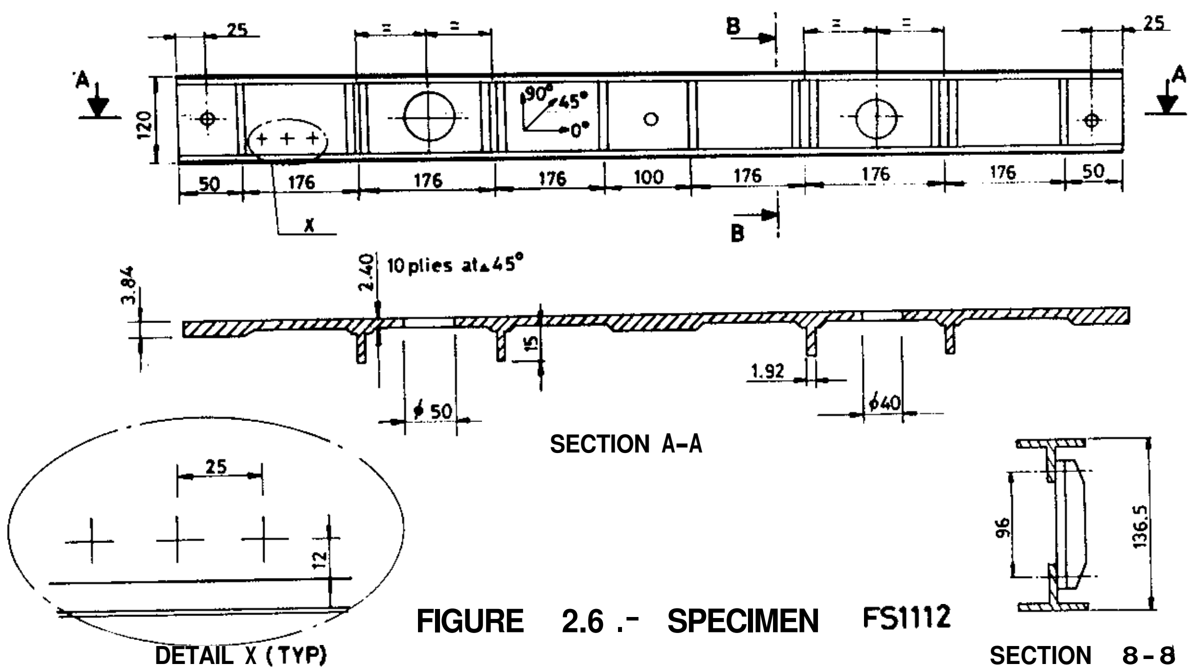


FIGURE 2.6 .- SPECIMEN FS1112

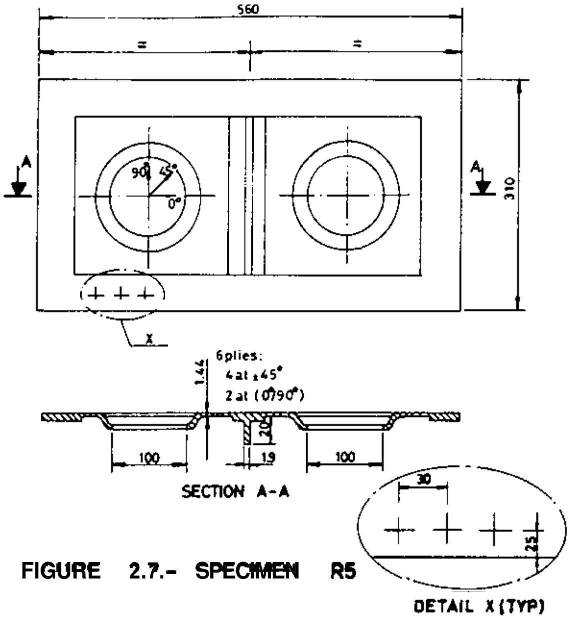


FIGURE 2.7.- SPECIMEN R5

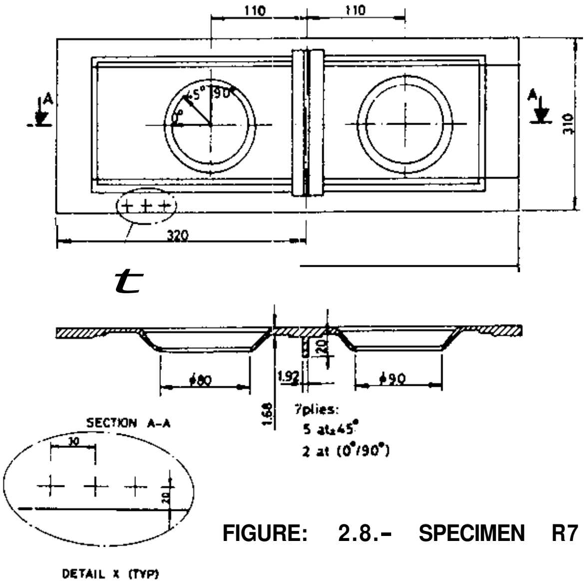


FIGURE: 2.8.- SPECIMEN R7

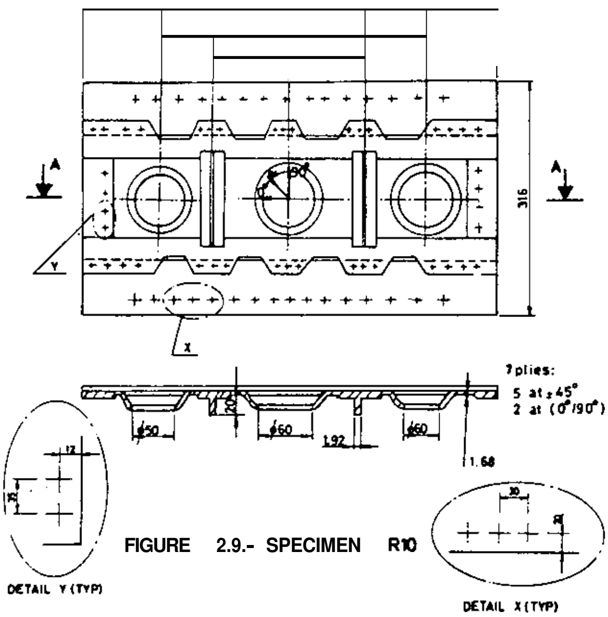


FIGURE 2.9.- SPECIMEN R10

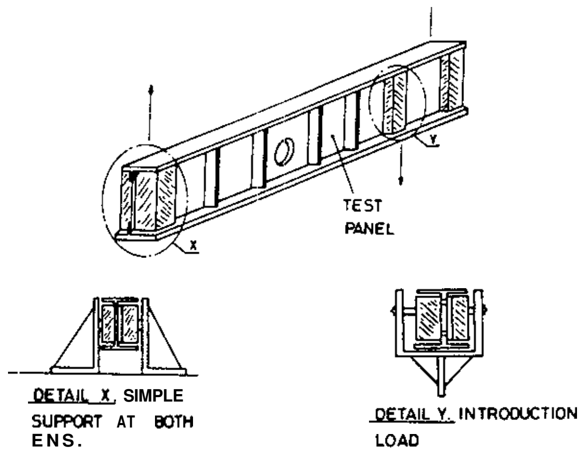


FIGURE 2.0.- THREE POINT BENDING TEST

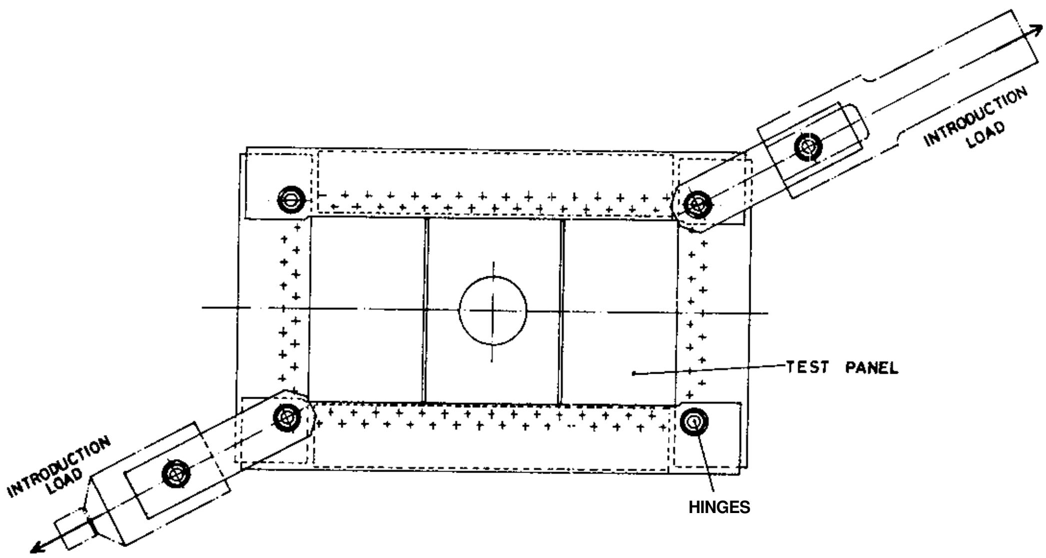


FIGURE 2.11 .- PICTURE FRAME TEST

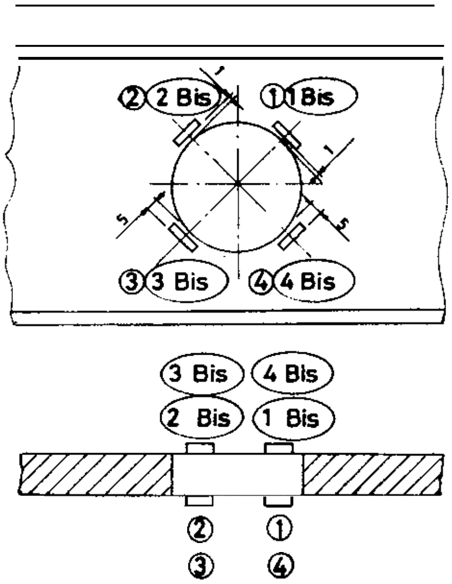


FIGURE 2.12.- STRAIN GAUGE INSTALLATION PLAN APPLICABLE TO ALL SPAR WEB SPECIMENS.

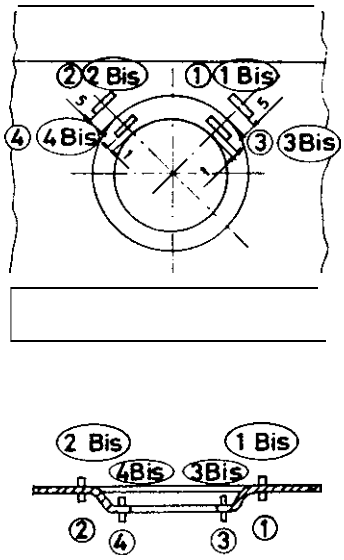
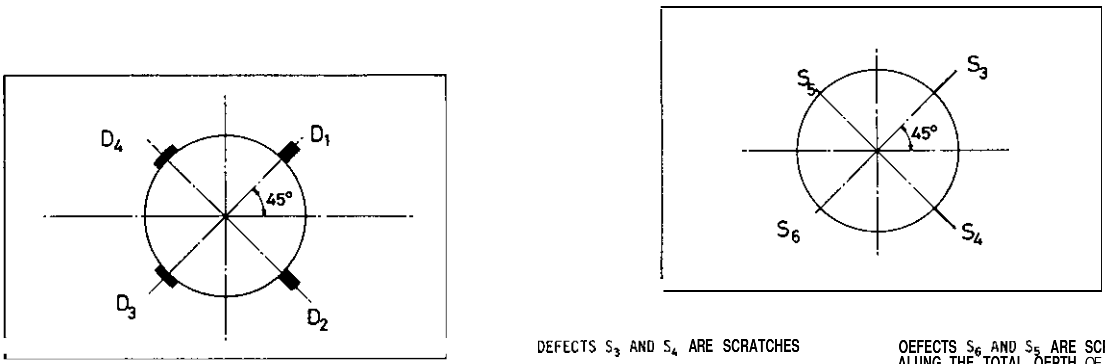


FIGURE 2.13.- STRAIN GAUGE INSTALLATION PLAN APRICABLE TO ALL RIB SPECIMENS.



TEFLON OR TITANIUM SHIM; t= 0,1 mm : BETYEEN PLIES N° 34 AND 35 FROM FLAT TOOL FACE

D_1 ; D_2 ; D_3 ; D_4: AREA 36 mm² (10.0 × 3.6)

SPECIMEN RS0203-3

DEFECTS S_3 AND S_4 ARE SCRATCHES 0,5 mm OEEP (2 PLIES), OUTER WEB FACE

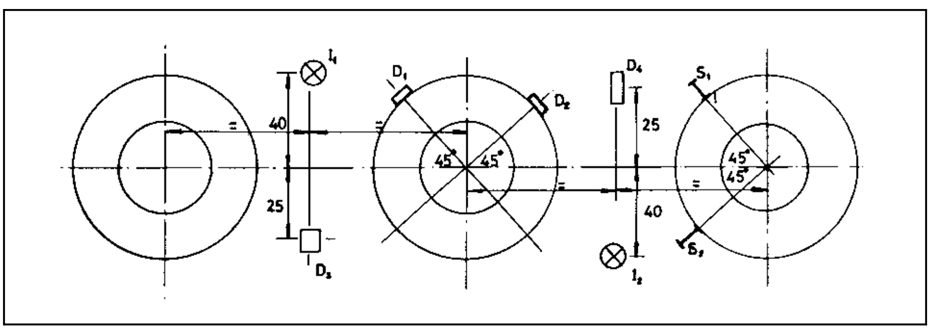
LENGTH = 25 mm
WIDTH = .5 mm

OEFECTS S_6 AND S_5 ARE SCRATCHES ALUNG THE TOTAL OEPTH OF THE HOLE BORE

LENGHT = .5 mm
WIDTH = .5 mm

SPECIMEN RS0203-4 ; RS0203-5

FIGURE *2.14.* SPECIMEN RS0203 DEFECTS



KEY: (1.) DEFECTS: D_1 6 D_2 ; AREA 36 mm²: (10.0 × 3.6) (2.) IMPACTS 5J, ø 25 mm

0.1 mm TEFLON OR Ti SHIM, BETWEEN PLIES 5 & 6 FROM TOOL FACE.

D_3 ; AREA 225 mm²: (15 × 15) (3.) SCRATCHES S_1 , S_2

LENGTH = 10.0 mm

D_4 , AREA 225 mm²: (25 × 9) DEPTH = 0.5 mm (≈ 2 PLIES)

SPECIMEN R10-5 ; R10-6

FIGURE 2.15. SPECIMEN R 10 DEFECTS

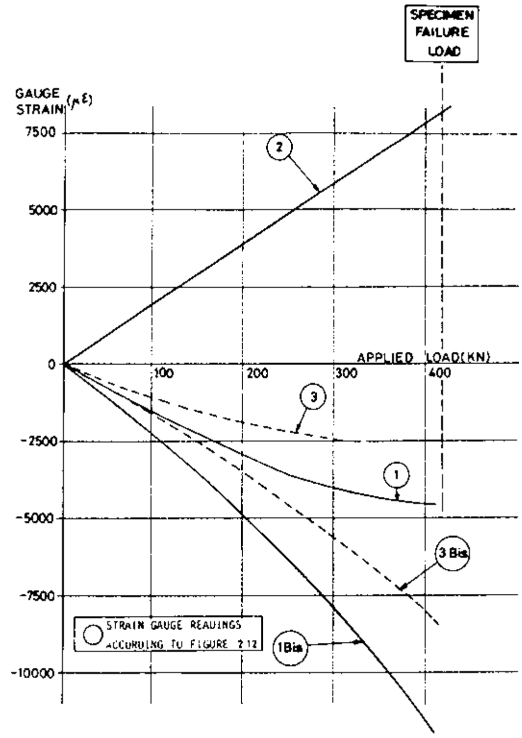


FIGURE 3.1.- STRAIN-LOAD HISTORY OF SPECIMEN RS0203-1

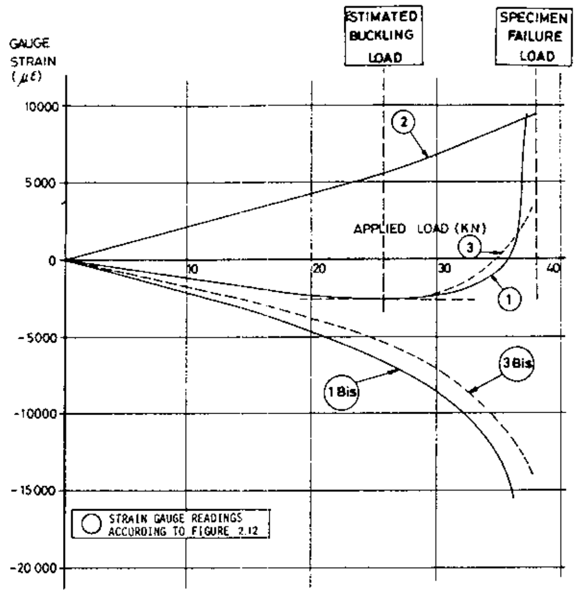


FIGURE 3.2.- STRAIN-LOAD HISTORY OF SPECIMEN FS1112-2

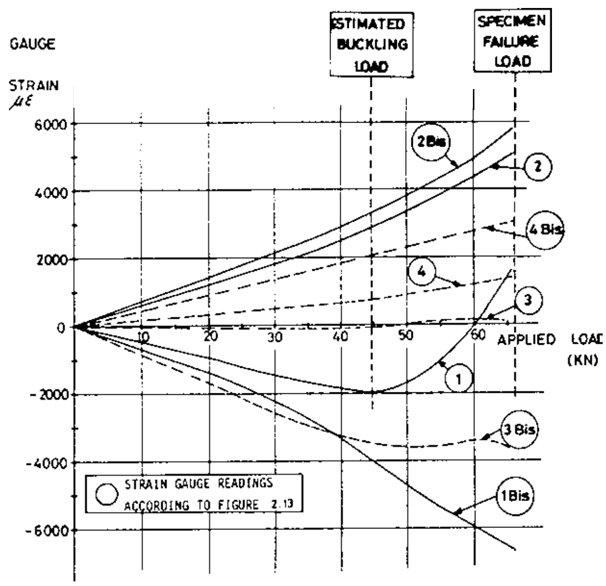


FIGURE 3.3.- STRAIN-LOAD HISTORY OF SPECIMEN R7-1

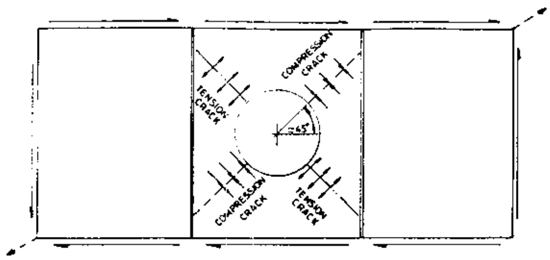


FIGURE 3.4 - FAILURE MODE OF SPAR WEB SPECIMENS.

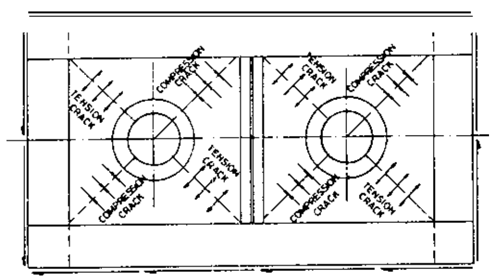


FIGURE 3.5.- FAILURE MODE OF RIB SPECIMENS.

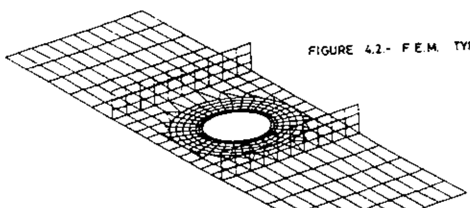


FIGURE 4.2- F.E.M. TYPE 2.

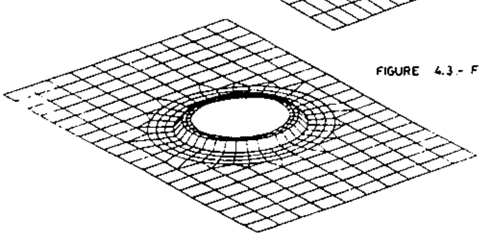


FIGURE 4.3.- F.E.M. TYPE 3.

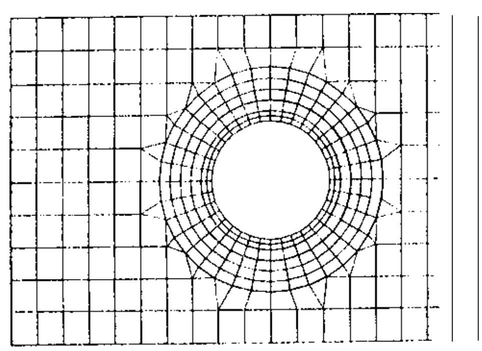


FIGURE 4.1.- F.E.M. TYPE 1

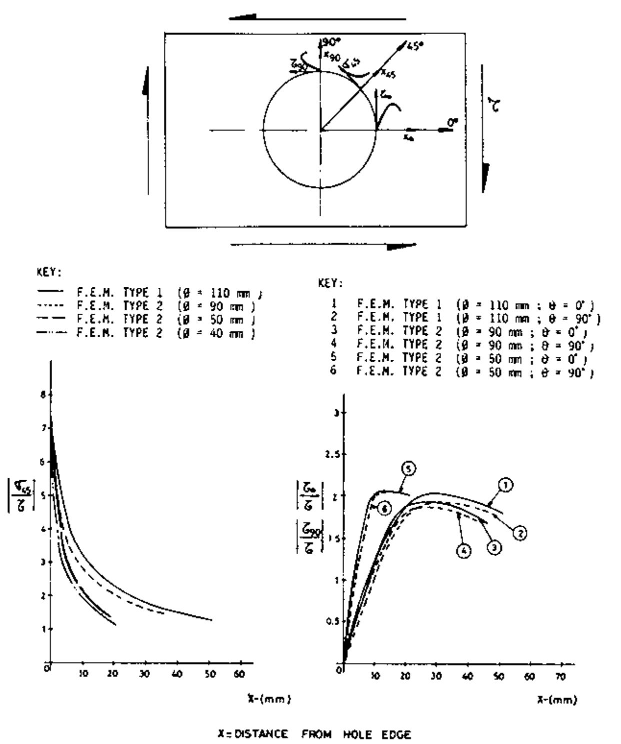


FIGURE 4.4.- IN-PLANE HOOP AND SHEAR STRESS CONCENTRATIONS FOR PLANE HOLE.

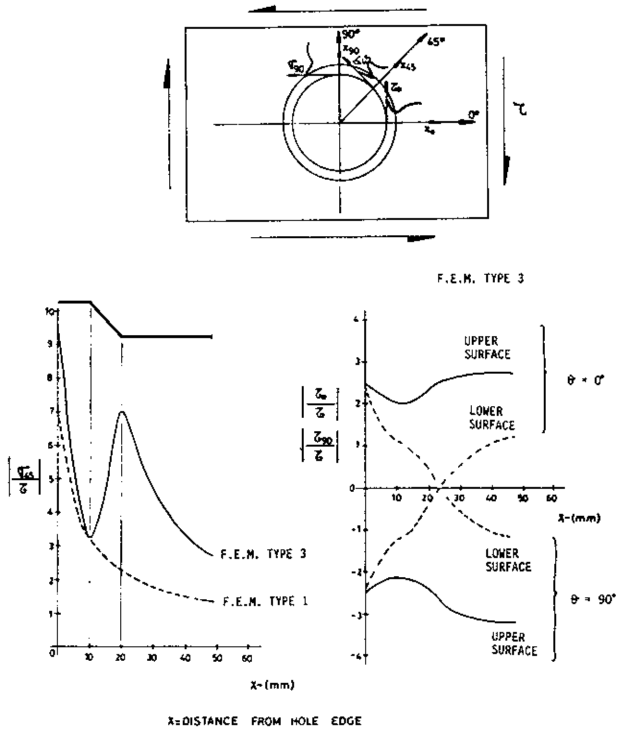
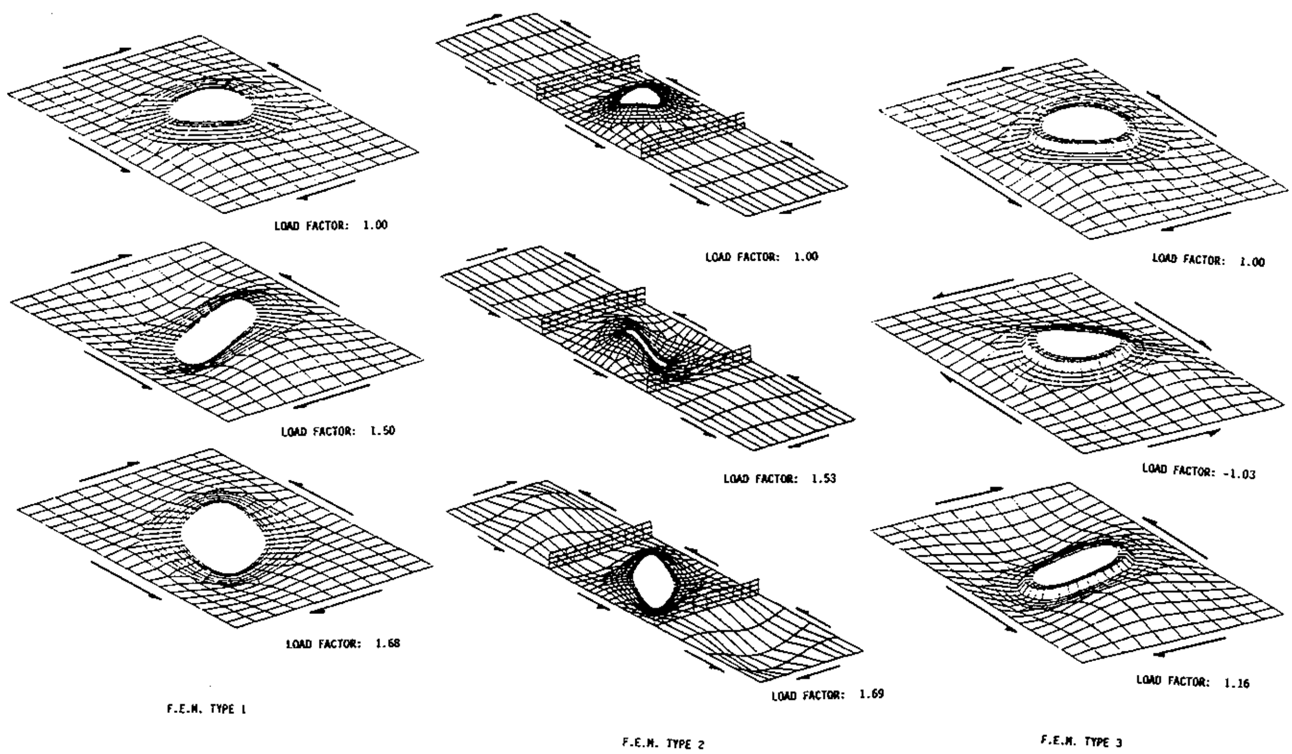


FIGURE 4.5.- IN-PLANE HOOP AND SHEAR STRESS CONCENTRATIONS FOR FLANGED HOLE.



FIGURES 4.6.- BUCKLING MODES SHAPES. F.E.M. TYPES 1,2 AND 3.

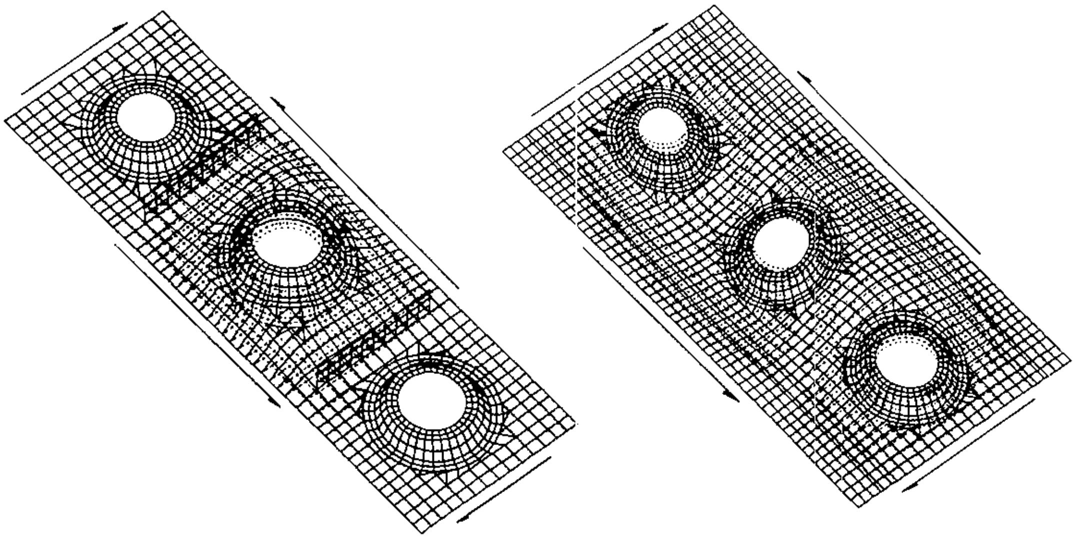


FIGURE 4.7.- BUCKLING MODES SHAPES RIB XI F.E.M

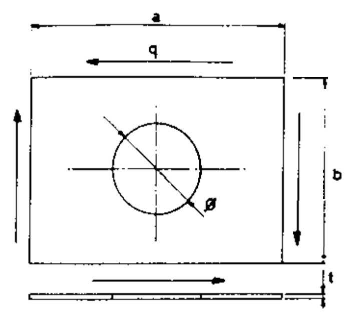


FIGURE 4.9.- PANEL WITH A CIRCULAR OPENING.

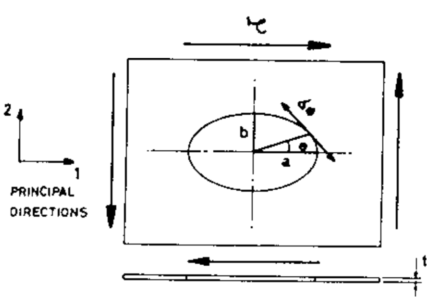


FIGURE 4.0.- STRESS CONCENTRATION AROUND AN ELLIPTICAL OPENING.

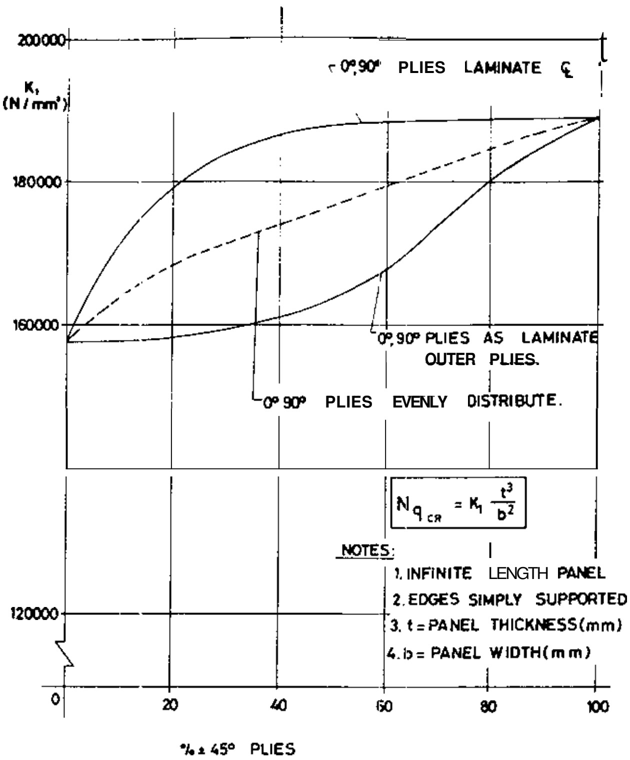


FIGURE 4.10.- ORTHOTROPIC SHEAR BUCKLING COEFFICIENT FOR C.A.S.A MATERIAL Z19.776 (P/W FABRIC)

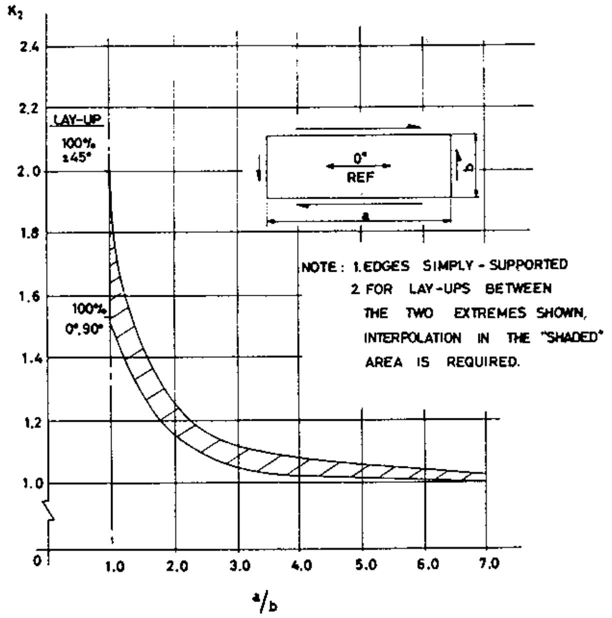


FIGURE 4.11.- ASPECT RATIO COMPENSATION FACTOR FOR ORTHOTROPIC SHEAR BUCKLING : C.A.S.A MATERIAL 219.716.

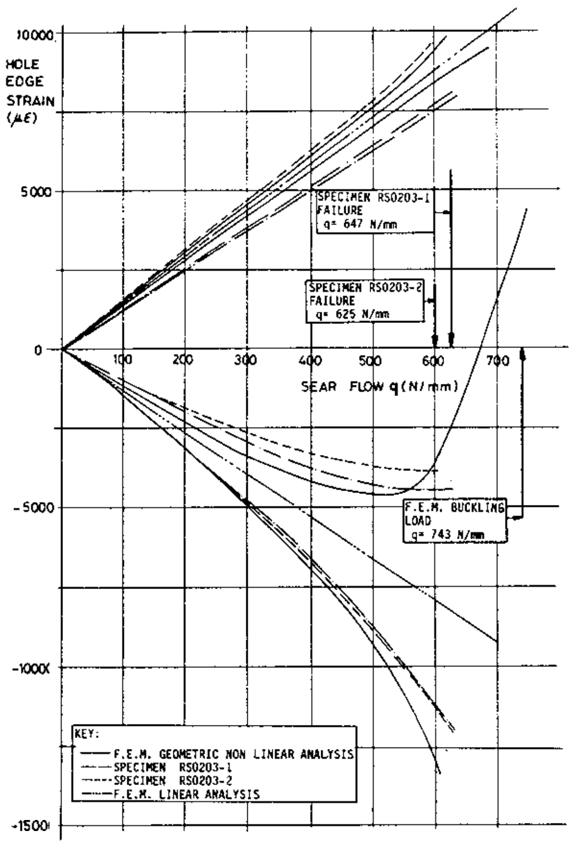


FIGURE 5.1.- RS0203 - 1 ; 2 ANALYSIS / TEST CORRELATION

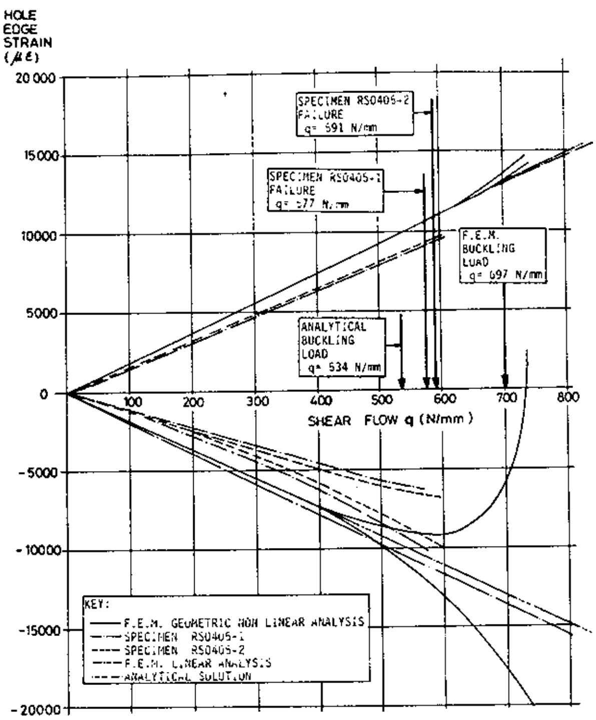


FIGURE 5.2.- RS0405 - 1 ; 2 . ANALYSIS/TEST CORRELATION.

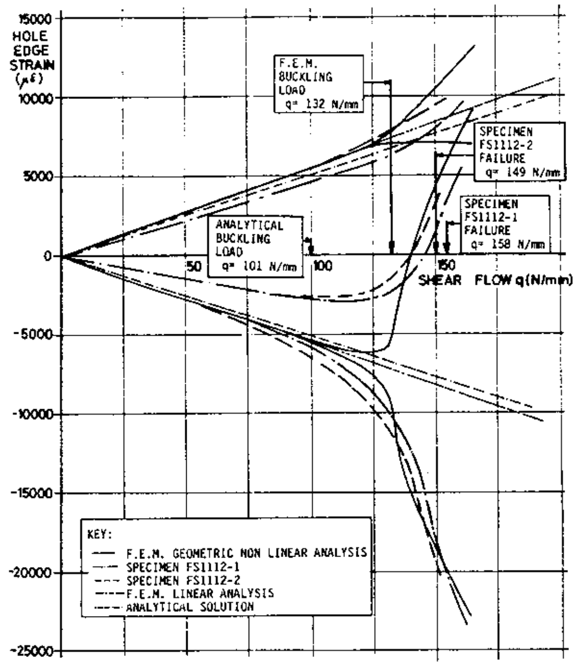


FIGURE 5.3.- FS1112 - 1 ; 2 . ANALYSIS / TEST CORRELATION

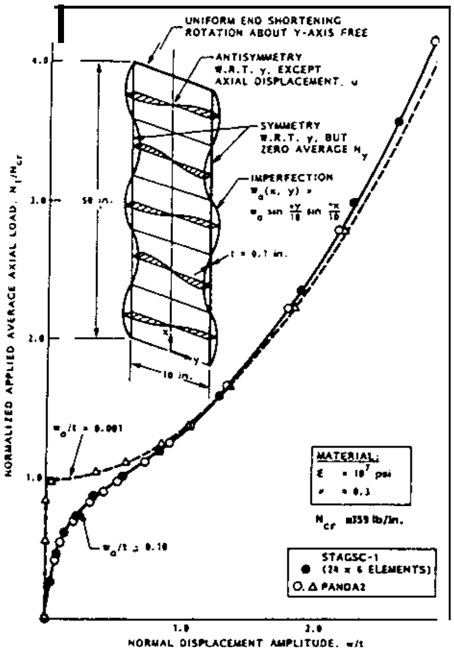


FIGURE *5.4.-* GEOMETRIC IMPERFECTIONS INFLUENCE

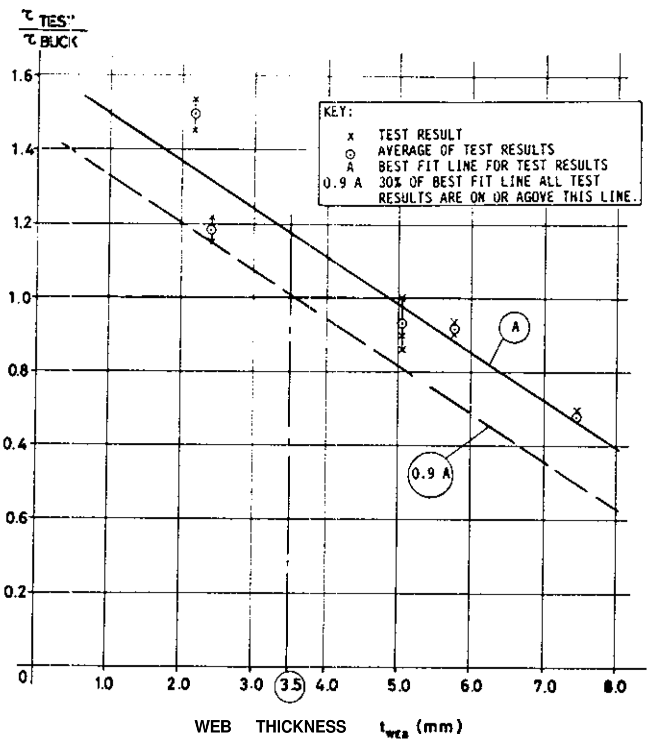


FIGURE 6.1.- τ_{TEST}/τ_{BUCK} vs t_{WEB}

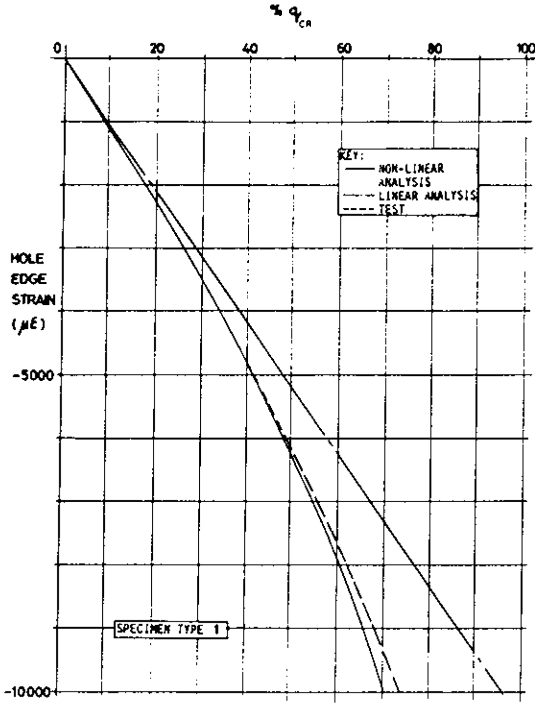


FIGURE 6.2.- RS0203 SPECIMEN HOLE EDGE STRAIN

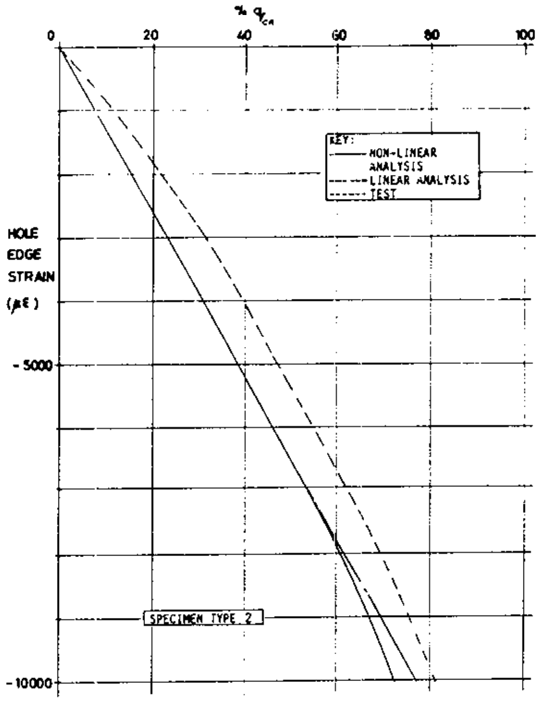


FIGURE 6.3,- RS0405 SPECIMEN HOLE EDGE STRAIN

vs % q_{CR}

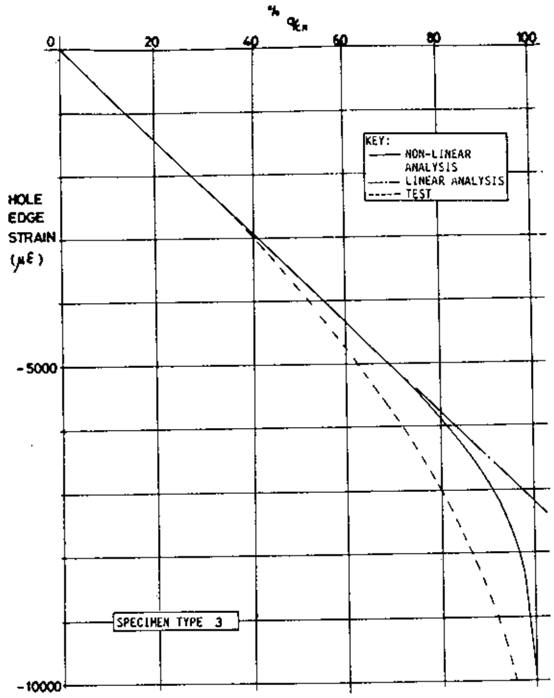


FIGURE 6.4.- FS1112 SPECIMEN HOLE EDGE STRAIN vs % q_{CR}.

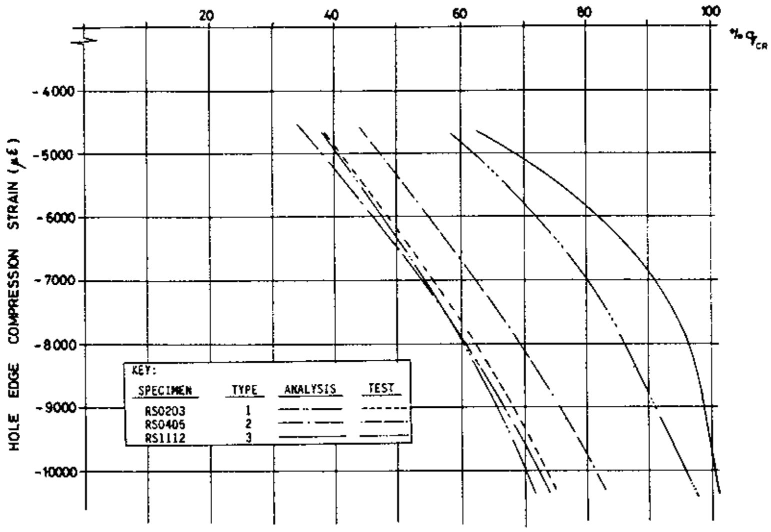


FIGURE 6.5.- SUMMARY OF HOLE EDGE COMPRESSION STRAIN vs % q_{CR}.

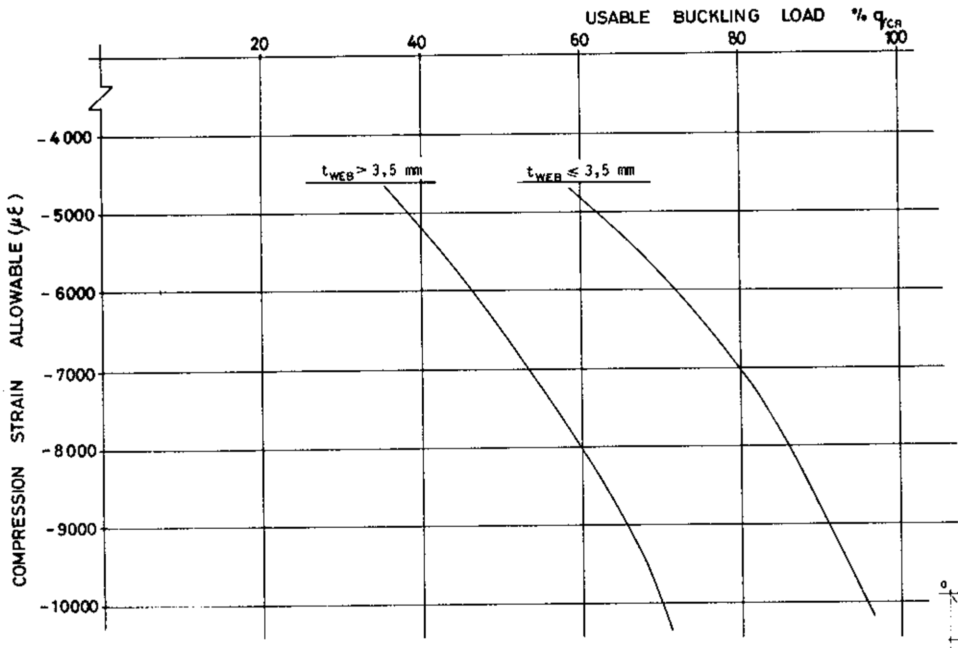


FIGURE 6.6 -

HOLE EDGE STRAIN

LIMITATION CRITERION.

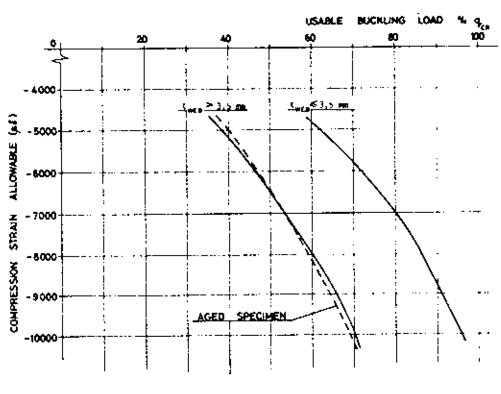


FIGURE 6.8 - HOLE EDGE STRAIN LIMITATION CRITERION SHOWING AGED SPECIMEN RESULT.

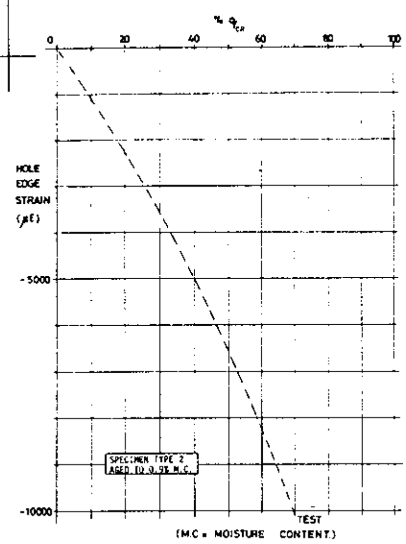


FIGURE 6.7.- RS0203-3 SPECIMEN HOLE EDGE STRAIN vs % q_{CR}.

BEECHCRAFT STARSHIP STRENGTH CERTIFICATION

by

E.H.Hooper
Beech Aircraft Corporation
P.O. Box 85
Wichita, KS 67201-0085
United States

OUTLINE:

- **Basic Aircraft Description**
- **Material Qualification**
- **Environmental Considerations**
- **Certification Basis**
- **Laminate Analysis Validation**
- **Damage Tolerance Evaluation**

Beech Aircraft, in the fall of 1982, launched one of the major aircraft development programs in the recent past. It is called the Beech STARSEIP shown in Figure 1.

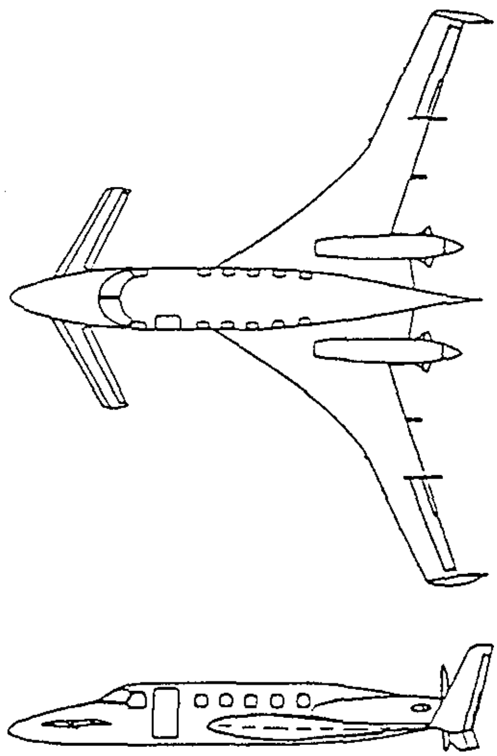


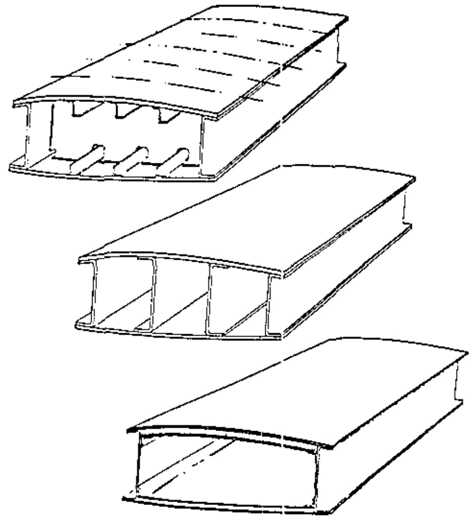
FIGURE 1

This particular airplane resulted from several years of study and research at Beech considering a variety of configurations. These included performance analyses, structural efficiency studies, and wind tunnel tests to verify the parameters.

The configuration selected is rather unusual in the general aviation market. It is a canard configured vehicle. The forward wings or canards pivot in order to trim the airplane depending upon the position of the flaps. When the flaps are extended in order to slow the airplane down and control the deck angle for landing, the forward ving pivots forward automatically to trim the pitching moment. Propulsion is provided by two pusher turbo prop propellers mounted close to the body to minimize the effect of engine out on control characteristics. Directional stability is provided by vertical stabilizers, one mounted at the tip of each ving, which also houses the rudders for directional control. The pitch control on the airplane involves both the elevator on the forward wing and the elevon on the aft ving vorking in conjunction. All fuel for the aircraft is carried in the triangular section at the leading edge root and the basic ving box is free of fuel.

Composite materials are used in the Starship. Significant gains are made in strength, veight, fatigue life, and corrosion resistance.

There are several major concepts for structural configuration as shown in Figure 2.



FIGUBB 2

The approach that is used largely in the Starship is one where there is a small number of spars and shear webs. The skin itself is the honeycomb sandvich that is stiffened and stabilized within itself to carry necessary loads. One of the main advantages of sandwich construction is its stiffness and strength properties shown in Figure 3. The Starship randvich configuration largely utilizes graphite-epoxy prepreg facesheets and the middle core is NOMEX honeycomb (in a variety of densities and shapes). One of the main advantages of the NOMEX compared to aluninum core is the avoidance of corrosion.

	t	2t	4t
Relative Stiffness (D)			3700
Relative Strength	100	700	925
Relative Weight	100	350	106
	100	103	

FIGUBB 3

I would like to share just a fev examples beginning with fuselage. The fuselage is 70 inch dia. and the pressurized section is 35 feet long. It is a monocoque (no frames or stringers) design made of NOMEX core and subsequently bonded together. Figure 4 schematically show hov these two fuselage shells are joined together with a Z type bond line. Adhesive as a secondary operation is cured to join these two halves. As a fail-safety measure, inside the fuselage along the top and bottom centerline, the joints are reinforced by a complete row of mechanical fasteners that would provide limit load carrying capability in the eventuality of a complete bond failure.

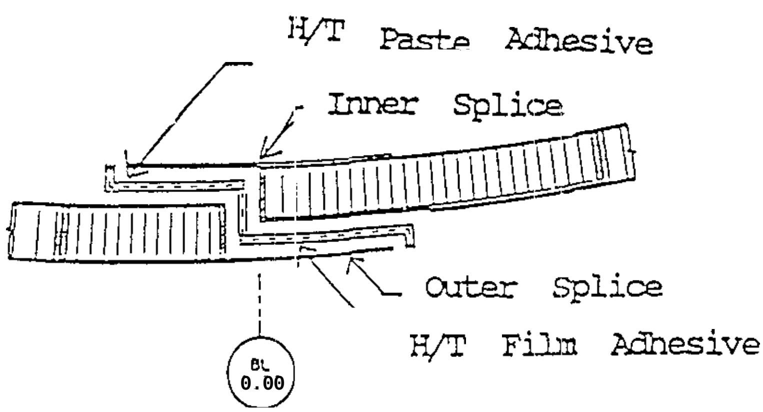


FIGURE 4

The wing is made in a 54 foot long tip to tip continuous skin. It has graphite epoxy facesheets with NOMEX honeycomb core. There are no concentrated spar caps or stringers. The skin is designed vith three shear webs attaching the upper and lover skins to each other. All the bending and torsion loads are carried within the skin.

A key element to this particular design, shown in Figure 5, is the end view of a prevoven structural joint called an "H" joint.

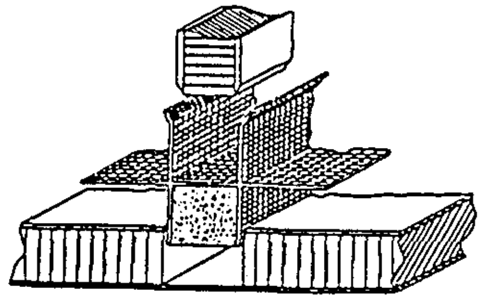


FIGURE 5

This joint has a unique feature in vhich the fibers pass through the intersection to avoid any peeling stresses. The fibers pass through the joint in the vertical direction and in the horizontal direction. The continuous fibers maintain the strength much higher than if laid up.

The Starship vas certified to FAA part 23 regulations including the commuter category option, amendment 4, and special conditions.

STARSHIP 1

Certification Rules: FAR 23

Historically Limited to 12,500 Lb & 9 Passengers

Amendment 34, 1986, Allows up to 19,000 Lb & 19 Passengers

PLUS: SPECIAL CONDITIONS

FIGURE 6

Special conditions vere published by the FAA in the Federal Register, August 8, 1986. These are additional regulations vhich are needed in the viev of thr FAA to certify new technology project vith a level of safety equivalent to that intended by the existing body of rules.

The special conditions, in addition to the company's normal demand for safety assurance, caused the STARSHIP to become one of the most rigorously tested airplane ever certified in the United States. Some of the previously unthought-of test included: flight test of the 85% scale demonstrator vith pressure taps to verify wind tunnel based panel pressure calculations; flight test of the full scale prototype vith simulated bugs plastered on the wings; two lightning strike tests an the same location, before and after repair. In the end, five special conditions applied to various aspects of the airframe and only one specifically addressed certification of composite structure.

The FAA special condition on composite structures specified many tests or analyses which would normally be conducted in the course of a major composites program. These included:

Flaw growth studies instead of fatigue life or fail-safe evaluation (all the time maintaining limit load capability).

Strength after impart damage regarded as an intrinsic property and therefore a material allovable associated vith design ultimate load.

Bnvironmental effects.

But, the requirements included for certification of adhesively bonded composite structures were entirely different than anything envisioned in early coordination meeting vith the FAA engineers. In fact, these abandoned damage tolerance practice developed over the last twenty years on metal structures (including bonded metal structures) and vent beyond the existing FAR part 25 transport aircraft rules.

STARSHIP 1

FAA Special Conditions Published in

Federal Reg. 8-8-86 (Pages 28500 - 25)

***13** Conditions: 5 Airframe,*

Including Composite Structure

FIGURE 7

MATERIALS

The best materials commercially available at the time of the program start were chosen based on fiber strain to failure (for damage tolerance), resin cure temperature (for processing economy and toughness) and resin glass transition temperature (for the required environmental resistance). A vide variety of weaves and areal weights (weight of fiber in grams pe: sq meter) are used to provided maximum design flexibility. Multiple fiber suppliers vere qualified in the certification program and additional prepreg vendors are being qualified for production.

The amazing fiber tensile strength, over half a million pounds per sq in from filaments of only 8 microns in diameter, is a reminder of the theory behind composite materials. In material fundamentals, purity equals strength and a finely dram filament is a way to achieve purity. The resin matrix is necessary to transfer load to the fibers, bind the fibers together, and to enable the material to be molded to shape. Very roughly, resin is added and we are left with 50% of the fiber strength (250,000 psi); plies are laminated to give strength in multiple directions leaving 50% of the unidirectional strength (125,000 psi). This is typical of the Starship wing skin in the spanwise direction; a further factor of approximately 50% is applies in aircraft design to account for practical stability and durability requirement, giving a maximum design stress at ultimate load of about 60,000 psi.

For comparison, aircraft aluminum has a maximum strength of 85,000 psi, and fatigue life considerations restrict the design stress at ultimate load to about 45,000 psi. Finally, the density of graphite/epoxy is only 57% of that of aluminum.

The materials qualification program established the lamina properties for each format certified. Tests are performed in Cold/Dry, RT/Dry, RT/Wet and Hot/Wet environments. It is now based on recent MIL-HDBK-17 committee recommendations but includes an increased number of tests in the first batch tested. The tests are performed according to ASTH D 3039-76 and ASTQ D 695.

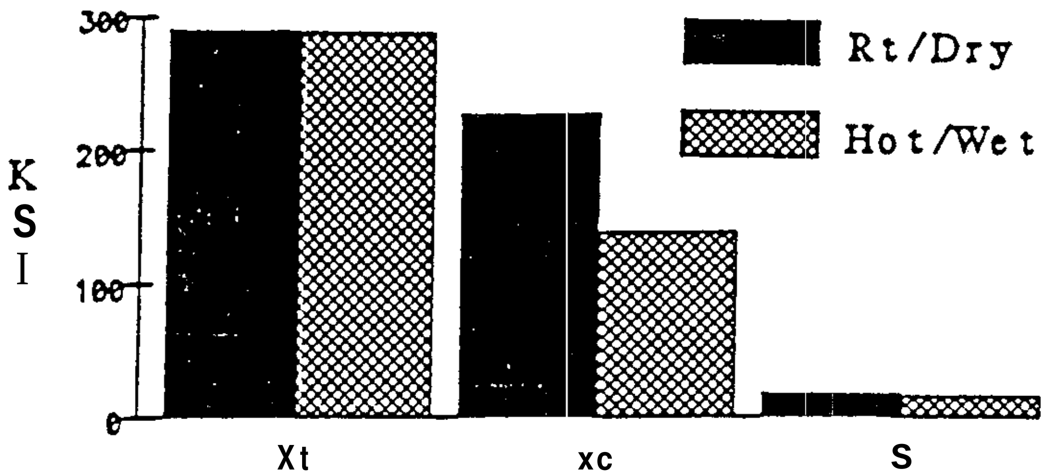


FIGURE 8

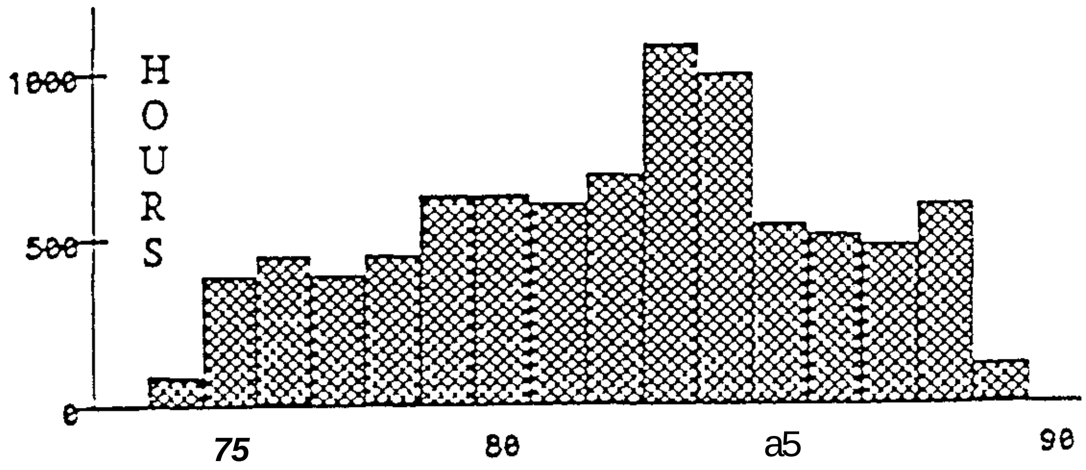


FIGURE 9

Once the definitions of maximum temperature and moisture were fully documented, material qualification testing could be completed.

FAR 23.613 requires that material design values must be taken form MIL-HDBK's 5, 17, 23, etc. Starship qualification program was conducted using the test matrix and statistical methods published by the MIL-HDBK-17 polymer matrix composites committee. The chart shows typical properties obtained at the ply (or lamina) level of material testing. Nine different lamina properties are required for use in laminate analyses for each material qualified; about 8,000 material tests were conducted in the initial material qualification program. The next level of testing up in scale from material coupons is element testing. About 1,200 panels were tested for tension, compression, shear, strength and stiffness in combination with cond/dry, room temperature/dry, room temperature/wet, and hot/wet environmental conditions. Large chambers were custom-built to condition groups of specimens as they were available from the composite fabrication shop.

Test article conditioning was accelerated by increasing the chamber temperature to 160 degrees F. when conditioning coupons and 140 degrees F, when conditioning assemblies. The advantages of basing the conditioning on RH (instead of % weight gain or number of days) are that it works regardless of the materials involved, also, when a large number of parts are declared ready for test, they can be maintained in the chamber until test capacity is avilable. Extra time in the chamber makes only a minute difference in moisture content.

The purpose of the element test program was to validate the Beech-generated Laminate Analysis Software Package (LASP). With a validated analysis method, it would not be necessary to test every possible combination of material plies used in the airplane structure.

The basic building block of all structural analysis is knowledge of the stiffness or modulus of the structural elements. The chart, Figure 10, shows calculated stiffness compared to measured values for different loading cases and laminates typical of those used in wing and fuselage structure.

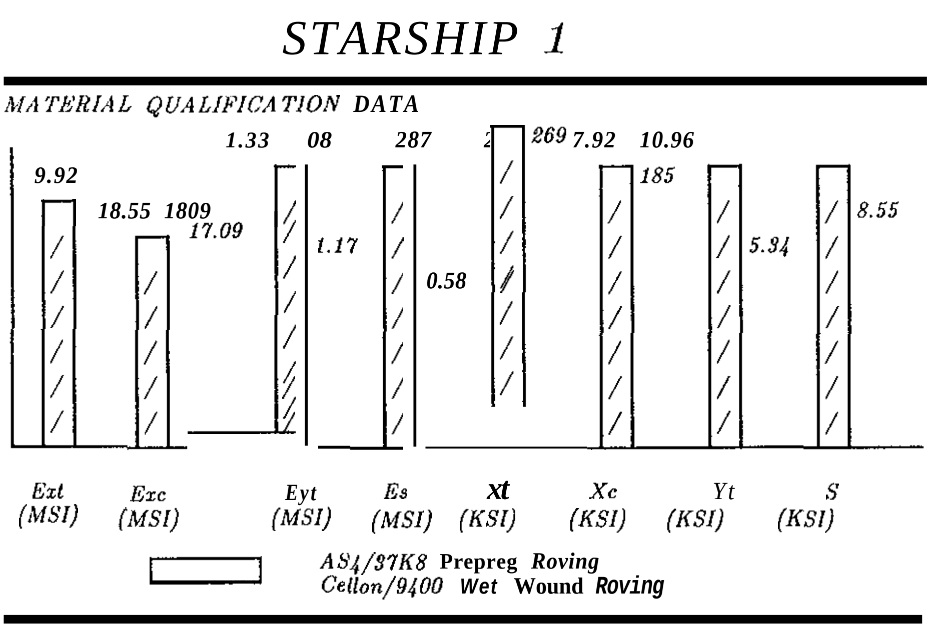


FIGURE 10

In modern airframe design, the failure modes which control the maximum load capability of much of the structure are stability modes; the part does not fail. because of low strength but deflects or bends excessively due to lack of stiffness. and the usual panel failure in compression will be due to column or panel buckling.

If stiffness or modulus of a panel can be calculated accurately, then the failure load in column buckling, also, ought to be predicable.

ENVIRONMENTAL DEFINITION

Because the mechanical properties of composites are sensitive to environmental conditions, a clearly documented definition of the extreme conditions under shich the structure would be interrogated was of prime importance.

THE ISLAND OF GUAM is situated in the Pacific Ocean 13 degrees above the equator. The climate provides steady exposure to combinations of high humidity and fairly high temperature (80" F), both around-the-year and around-the-clock; ideal conditions for the accumulation of moisture in composites. Andersen Air Base, **Guam**, was identified as the worst-case base for moisture exposure in a USAF sponsored survey of 156 bases, world-wide. Actually, many Pacific/tropical locations would provide similar humidity conditions and figure 12 shows typical hours of exposure per year in each relative humidity bracket. The average humidity for the most humid month (85.3% in March) was chosen as a simple single level representation of the year-round exposure. A little margin was added far test chamber control purposes, giving a RH of **87% as** the exposure requirement to generate maximum moisture content far static testing (a slightly lower RH was used in fatigue and flaw growth testing based on a mission/moisture analysis).

DEATH VALLEY, CALIFORNIA, must be another wonderful place to live. The ambient temperature on a typical July afternoon is 116 degrees F. The highest ever recorded there was 134 degrees F. This worst-ever temperature is not considered in combination with flight loads as the airplane systems are not qualified for take,-off with ambients over 120 degrees F. In addition to the very high ambient temperatures the airplane upper surfaces may be heated by solar radiation of 310 btu/sqft/hr. The high ambient temperature and solar radiation combined in solar soak tests to heat the ving upper skin to 150 degrees F when medium gray paint was used.

STRUCTURAL VALIDATION

Over 1200 panels vere tested for strength and stiffness; tension, compression, and shear; and environmental effects of cold/dry, room temperature/dry, room temperature/wet, and hot/wet conditions. Large chambers were custom-built to condition groups of specimens as they were available from the composite fabrication shop.

TEST ARTICLE CONDITIONING was accelerated by increasing: the chamber temperature to 160 degrees P. when conditioning panels and 140 degrees F. when conditioning assemblies. The advantages of basing the conditioning on RH (instead of percent weight gain or number of days) are that it works regardless of the materials involved, and when a large number of parts are declared ready for test, they can be maintained in the chamber until test capacity is available. Extra time in the chamber makes only a minute difference in moisture content.

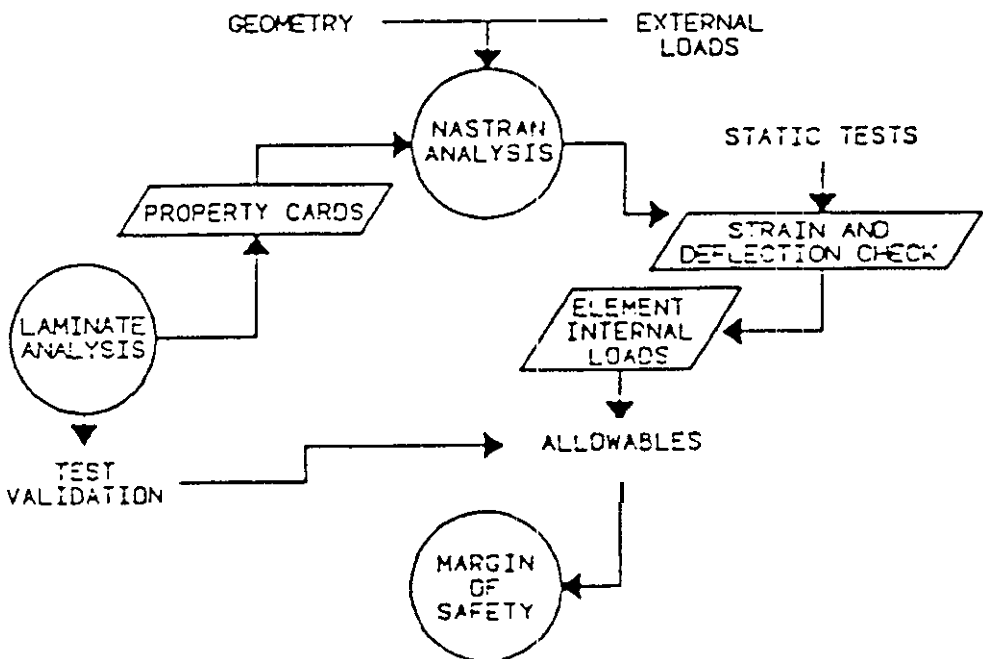


FIGURE 11

VARIOUS TYPES OF DAMAGE were introduced into test panels, including impact damage covering the range from no visible damage to puncture to define the Threshold Of Detectability (TOD) damage level for various laminates.

Also, large, complex-load path subcomponents of the airplane were conditioned to moisture saturation, heated to 160 degrees F, and loaded to appropriate internal pressure and bending loads. Strain and deflection data showed that temperature and moisture conditions alone created a few hundred micro-strain preload in the structure did not change the final accuracy of predictions under environment and load combinations.

FATIGUE AND DAMAGE TOLERANCE

Carbon fiber composites are remarkably resistant to fatigue loading and test samples were cycled at very high percentages of their maximum strength in order to generate flaw growth data.

STRUCTURAL ANALYSIS

Substantiation of composite structure was proposed to be by laminate analysis techniques which had been validated by panel tests, taking into account internal loads on each composite element, the required environmental conditions, and the appropriate lamina properties. The internal loads would be calculated by finite element analysis and the accuracy checked by comparison to full scale static tests run under lab ambient conditions as shown in the flow chart, figure 11.

THE FAA RESPONSE was: "Good idea. State-of-the art program, in fact. But... what about the effect of temperature and moisture on internal strain even before mechanical loading is applied? And, how accurately will finite element analysis predict combined effects of environment and mechanical loading on complex three dimensional structures?" These concerns were successfully addressed by two methods:

One, a commitment was made that analysis predicted strains would match the test results within ten percent. This was achieved by careful modelling of the tested structure with its internal reinforcements, test loading, and the tie-downs and restraints used in the test.

THE LOAD SPECTRUM applied in full scale damage tolerance testing was based on three mission profiles generated from executive usage data in the existing King Air fleet. The cyclic loads applied included gust, maneuver, cabin pressure, landing, engine torque, thrust, and minimum on-ground load.

A SCATTER FACTOR ON LOADS of 1.15 was applied which allowed one service life to be statistically represented by two lifetimes of laboratory testing. All safety of flight components were tested in the full scale program, including: forward wing and nose structure carry-through, main wing and its fuselage attachments, pressure cabin, vertical stabilizer, and control surfaces. The first test lifetime was applied to as-manufactured structures, after which additional damage was mechanically inflicted to each test article.

INFLICTED DAMAGE represented such in-service undesireables as impact damage, lightning strike, loose and missing rivets, disbanded adhesive joints, delaminated and punctured composite parts, and cracked or gouged metal parts. The method of simulating lightning strike damage was to burn the composite with a welding torch until the appearance and size of the burn was similar to that seen from lightning strike tests.

RESIDUAL STRENGTH TESTS concluded the damage tolerance program to demonstrate that the load carrying capability of the structures had been maintained to at least design limit load. Strain and deflection histories were evidence that overall structural stiffness had not changed to an extent that would effect flight flutter margins.

ADHESIVELY BONDED JOINTS were proven to be extremely tolerant of fatigue loading and highly resistant to flaw (disband) growth. However, additional analyses were conducted to show redundant load paths or to identify joints which needed back-up mechanical fasteners to meet the FAA special condition. The option of production proof testing was chosen rather than add rivets to certain extremely weight sensitive and light-gage parts such as control surfaces.

CONCLUSION

The all-composite airframe can be successfully designed and analyzed with today's technology. Simple designs using essentially monocoque techniques facilitate economical fabrication of parts and assemblies. An FAA certification program demands careful planning and coordination, especially concerning regulations, interpretations, test criteria, test plans, and test witnessing.

THE ROLE OF ANALYSIS IN THE DESIGN OF COMPOSITE MATERIALS

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SUMMARY

The current role of analysis in the design of advanced polymer composite materials is presented. The correlation between structural property prediction and measurement is discussed for both material characterization and component testing. It is reported that to predict the stiffness of multidirectional laminates from the unidirectional material properties, laminated plate theory is reasonably accurate and generally accepted. Strength predictions are much more difficult because of the large variety of failure mechanisms, and some of the existing failure criteria are inaccurate. Because of its simplicity, the most widely used is the maximum strain criterion, although it ignores multiple failure mode interactions and is not easily adaptable to model progressive failures. A much better altemative is the quadratic failure criterion, acknowledged as the most sound analytically, internally consistent and easily adaptable to progressive failures. but still not generally accepted by most of industry. Following an assessment of the most advanced and most accurate analytical techniques is a discussion of a few of the most detrimental, although commonly accepted, shortcuts to analysis. It is concluded that: (1) the role of analysis is critical, (2) the best contemporary analysis techniques are underutilized and (3) widely used current practises are compromised by use of outdated rules. It is recommended that analytical tools be built on fundamentals, developing an integrated micro-macro-mechanics design methodology, with the goal of using analysis rationally to ***tailor*** new materials. The full potential of Composite materials can then be realized.

1. INTRODUCTION

Currently, black metal design is predominantly used: replacing a metal part with an identical one made of carbon fiber reinforced polymer (CFRP) and achieving only the design advantage provided by the weight savings of these lower density materials. Industry's inclination is generally not to deviate too much from a quasi-isotropic laminate, although optimal design can only be achieved when laminates are "tailored", orienting the load carrying fibers in the directions that the loading conditions dictate. For any system of fiber and matrix, many different materials in the form of different laminates are possible—too many to characterize empirically however.

The number and ratio of ply orientations in a laminate should be decided by analysis rather then arbitrary rules such as the 10 percent rule, and the rule specifying the use of balanced laminates only. These rules, resulting in an ineffective or even possibly unsafe use of composite materials, were introduced two decades ago due to the lack of analytic understanding and lack of computing power. There are several reasons why simplistic rules are still used for design: (1) analytical tools are unable as yet to provide all the answers; (2) structural optimization utilizing directionality is regarded as secondary to other factors, such as damage tolerance. While simple tools may provide a short term solution, further development of analysis will reduce the number of unknowns, allowing a global perspective that can take into account all the aspects of design.

Laminated plate theory, the acknowledged foundation analysis, is currently only used during the final design stages, although it ought to be introduced during preliminary design. This goal can be achieved today, thanks to the explosive advances of modem personal computers and workstations. To capitalize on the opportunity to "design the material", it is essential that the ability to calculate is exercised, improvements are made as experience is gained, and confidence in analytical techniques is built, so that advantage can he taken of these tools to guide design. Areas of analytical development include technical issues such as the selection of the constituent materials and their processing, and the design criteria for stiffness, failure, fracture, ply orientations, ply drop, stress concentrations and others. Much work remains to be accomplished, particularly in the analytic modeling of progressive failures in three-dimensional composite laminates.

2. COMPARISON OF PREDICTIONS AND MEASUREMENTS

A composite is formed when fibers are combined with a matrix material. Micromechanics is the study whose objective is to analytically relate the material properties of the fiber and matrix constituents to those of the resulting composite material. At this time, there are many micromechanics theories, nearly all of which are limited to stiffness property prediction, being inadequate to relate the constituent strength properties to those of the ply.

Macromeehanics is the study whose objective is to determine the value of output variables, such as laminate strength for a set of conditions, or required laminate thickness and orientation to satisfy a set of loading conditions. Once the material property data have been determined, they are used in conjunction with boundary conditions such as geometry and loading data, and serve as the starling point to analysis. The most fundamental part of macromechanics, laminated plate theory is a two dimensional theory that can consistently predict stiffness measurements of a multidirectional laminate from the ply properties. Its use is becoming generally accepted, displacing the reliance on inaccurate techniques such as netting analysis and carpet plots. Comparison of laminate stiffness measurements and predietion by laminated plate theory generally can be expected to be within a few percent.

A. Fiber and Matrix

Empmcal testing of fibers and of the matrix material is generally for quality control purposes, and not to determine input values for analysis. Fiber tests include filament strength as a function of gage length and strand tests. Although many fibers are orthotropic, we are not aware of a method to empirically determine the transverse and shear properties. In addition to the usual stiffness and strength tests of the matrix, a number of tests such as the gel permeation chromatography (GPC), differential scanning calorimetry (DSC), glass transition temperature, the wetting and bonding at the interface, and compact tension fixture toughness for G_{1C} are not used quantitatively in the design of composite laminates.

B. Unidirectional Laminates

The goal of material property testing is to sufficiently characterize a material so that a rational design process can be exercised for improvements in materials and processing as well as for leading to cost-effective and reliable structures. While isotropic materials, such as metals, have the same material properties in all directions, composites have different properties in different directions, as a function of the fiber orientation. Consequently, more parameters are required to characterize a composite naterial than a metal. For metals, the modulus and Poison's ratio will completely describe the material stiffness, and one strength value is valid not only for all directions but also for both tension and compression. For composites, the following material properties are required for accurate smctural characterization:

Ex	Longitudinal modulus	X	Longitudinal tensile strength
Ey	Transvene modulus	X'	Longitudinal compressive strength
Es	Shear modulus	Y	Transverse tensile strength
nu	Poisson's ratio	Y'	Transverse compressive strength
		S	Sh[illegible]

These unidirectional ply properties are simple to determine and represent the basic building block for analysis. They can be determined by doing a minimum of three repetitions of five uniaxial tests on basic specimens. Guidance for the test procedures can be found from ASTM Standards. The tension test of a 0 degree specimens and 90 degree specimens is described by ASTM D3039; the tension test of a ±45 degree specimen to determine the in-plane shear properties is described by ASTM **3518.** Compression testing of a 0 degree specimen is a more difficult test and there are several test methods in existence, for example ASTM D3410. **The** compression strength of a 90 degree specimen can be determined using ASTM D3410, or alternatively ASTM **D695** which govems nonstructural plastics.

Stiffness correlation is generally accurate within a few percent. Stiffness, which reflects the linear behavior of composites, is easier to measure than strength, which represents the ***point*** of failure. Strength correlation, hopefully varying by less than 10 percent, is more difficult because of several factors: material variability, quality of specimen preparation, stress concentrations due to misalignment of the specimen and load introduction, and test parameters such as loading rate. Confidence in the data is improved by increasing the number of repetitions.

C. Multidirectional Laminates

Similar to the problems associated with determining material strength, the strength prediction of a laminate is much more difficult than the stiffness prediction. A strength prediction capability in the neighborhood of ten percent should be viewed as successful. With composites, many different failure modes are possible. It is impossible to have one criterion that can consider all the mechanisms. While there are several failure criteria in existence, the maximum strain failure criterion is the most widely used because of its apparent simplicity but is not adequate because:

- It ignores failure mode interactions; from which carpet plots can be justified.
- It does not normally include progressive matrix degradation that can bridge between first-ply-failure (FPF) and the ultimate failure of a laminate.
- It is not invariant; **i.e.,** a **0.4** percent strain is not coordinate independent

The **need** to improve the analytic prediction of the behavior of laminates is urgently needed to achieve a high level of confidence in composite materials. The use of quadratic failure criterion is recommended as a first step. This criterion is superior because:

- It is the simplest criterion to include failure mode interactions; from which carpet plots are easily shown to be dangerous and do injustice to composite materials.
- It provides an easy transition between matrix and fiber failures by use of a matrix degradation factor
- It provides a simple determination of FPF and the ultimate strengths.
- It is a scalar criterion **and** therefore invariant. It ***is*** the easiest to manipulate mathematically.
- It can be applied to describe both interlaminar as well as intralaminar failures.

D. Structural Elements

There is an entire book of ASTM Standards devoted to composite materials testing, which outlines about thirty tests. In addition, because of the extensive time and effort involved in establishing an ASTM Standard, government agencies such as NASA and companies in industry have defined **their** own intemal tests. The Standard Test Methods Task Force (SACMA) represents current efforts within the US to unify composites testing. Unfortunately, most tests are not designed with analytical considerations, providing essentially empirical data for qualitative comparison, and are often unable to provide the desired input to the analytic modeling of composite laminates and structures.

Currently, many of the tests in existence are more of a structural component test than material characterization. In many cases, these tests have been defined by materials and process engineers who are not concerned with analysis and design. Tests for quality control should investigate the parameters reflecting the performance of composite materials for their intended use. Tests such as short **beam** shear, open-hole compiession, transverse impact, bearing, nano tester, and others are designed for specific applications; design implications of these tests should be regarded with skepticism.

For example, an empirical data base is being expanded using this open hole specimen. Since the data are complicated to measure and difficult to interpret, evaluation based on this difficult test results in an unfair assessment of composite materials. **Prior** to the measurement of failure strain for an open-hole compression test, which is boundary-condition dependent, basic material properties need to be identified. With these data, an analytical model of an open hole :i e ı can be t, predictive power can be developed and the need for this pirical test will b :liminat‹ ı 1 y, tests must be designed, and at least verified by some analysis for measured deformation and failure modes

3. DETRIMENTAL SHORTCUTS

Instead of developing and improving analytical techniques, there is often a tendency in practice to discount or **dismiss** analysis as being an inadequate basis for design, since it does not yet answer all the "complicated" issues of bolted joints, damage tolerance, etc. Design is accomplished through comparison with empirical databases and "experience". In view of the mechanics work in the Materials Laboratory over the past twenty years, and accepted theories by the academic and research community, design driven by simplified theories and conservative practices is inadequate.

A. Netting Analysis

Netting analysis ignores the smctural contribution of the matrix within a laminate, considering **only** that the fibers **are** carrying the loads. This concept can be treated as a special case of laminated plate theory by letting the matrix degradation factor approach zero. As an example, the difference between netting analysis and more exact thick wall cylinder analysis can be seen in Figures 1a and 1b for the optimum pressure vessel under intemal pressure. The optimum vessel under extemal pressure is shown in Figures 2a and 2b.

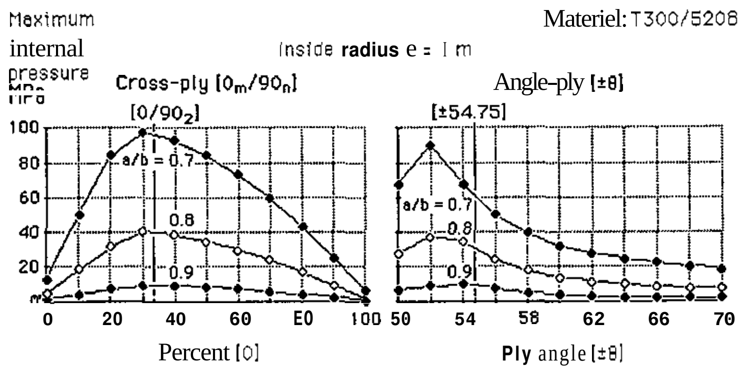


FIGURE 1 a) Maximum intemal pressure capability versus the percentage of [0] in cross ply laminates, for three thicknesses of pressure vessels defined by the ratio of inner to outer radii a/b; b) Maximum intemal pressure capability versus the orientation of angle-ply laminates, for three thicknesses of pressure vessels defined by the ratio of inner to outer radii a/b.

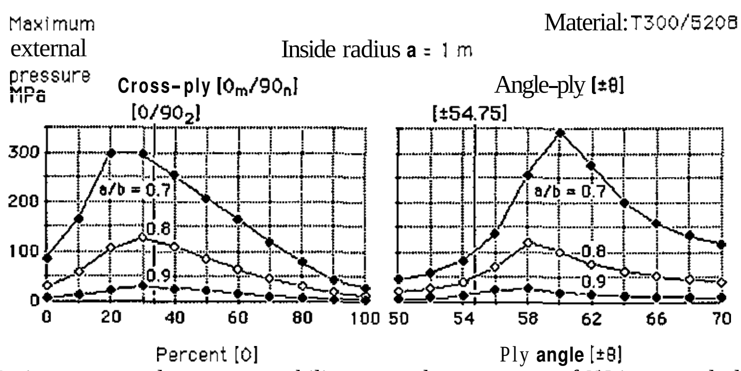


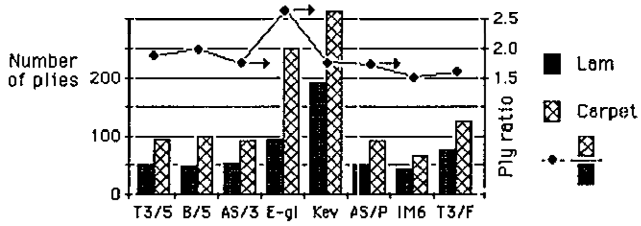
FIGURE 2 a) Maximum extemal pressure capability versus the percentage of [0] in cross ply laminates, for three thicknesses of pressure vessels defined by the ratio of inner to outer radii a/b; b) Maximum extemal pressure capability versus the orientation of angle-ply laminates, for three thicknesses of pressure vessels defined by the ratio of inner to outer radii a/b.

Simple stress analysis of pressure vessels predicts half as much stress in the axial direction relative to the tangential direction. Accordingly, netting analysis translates the 1 to 2 stress ratio. into a 1 to 2 laminate, that is $[0/90_2]$. As seen in Figures 1a and 2a this can be an acceptable criteria for cross-ply laminates, but leads to misleading conclusions for angle-ply laminates, in which load interactions and matrix contributions are more important. Netting analysis incorrectly implies that the $[0/90_2]$ and the [±54.75] laminates **are** equally acceptable. Additionally, netting analysis does not differentiate between extemal and intemal pressure, indiscriminately specifying the same laminate. According to a more sophisticated thick wall cylinder analysis, Figure 1b shows that for intemal pressure **a** [±52] laminate is significantly better than a [±54.75], while Figure 2b shows that for extemal pressure a **[±60]** laminate is the best angle-ply. Note that thick wall cylinder analysis is able to predict a maximum pressure capability that is three times greater under extemal pressure.

It can be concluded that netting analysis ignores several important factors, such as load interaction and matrix degradation, and consequently oversimplifies the analysis and leads to erroneous design.

B. Carpet Plots

Another example of a detrimental shortcut can be seen in the comparison of laminates designed by carpet plots versus those by laminated plate theory. Carpet plots simplify laminate analysis by decomposing a complex stress into three simple stresses, based on the assumed validity of the principle of superposition for strength. Figure 3 compares the two design methods.



Representative composite materials

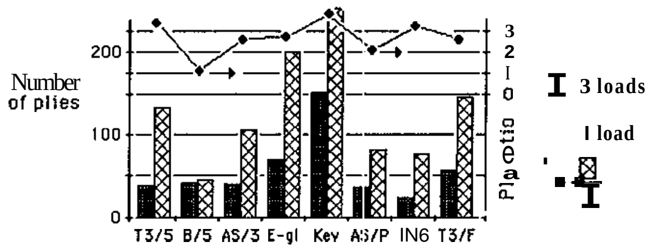
FIGURE 3 Number of plies required as a function of composite material to sustain an applied load {4,-2,1} MN/m, as predicted by laminated plate theory and carpet plots.

For an applied load {4,-2,1} MN/m, the carpet plot desigr. would treat it as multiple loads: {4,0,0}, {0.-2,0}, and {0,0,1}. For this load, carpet plot is over conservative by a factor of two. If a load is complex, it must be treated as such. Superposition does not work for strength.

We axe sorry to report that carpet plots are still in use, including the data shared by the consortium of the current European fighter. We are not challenging a discrepancy of 20 percent that may exist between different approaches. We are talking about several hundred percent. As great as composite materials are, they cannot be competitive if simplified approaches carry with them such penalties.

C. Non-interactive Maximum Strain Criterion

If failure modes are non-interacting, as it is assumed in the maximum stress and maximum strain criteria, multiple loads can be "synthesized as one complex stress. Figure 4 compares the thicknesses of laminates designed. for the simplified one load case and for the actual three load multiple case.



Representative composite materials

FIGURE 4 Number of plies required as a function of composite material to sustain a case of multiple loads {(4,0,0}, {0,-2,0}, and {0,0,1} MN/m, as predicted by laminated plate theory and carpet plots.

For a multiple load case of {4,0,0}, {0,-2,0}, and {0,0,1} which would require the stress analysis be done three times, the design is simplified to the stress analysis of a single, complex load {4,-2,1}. Figure 4 shows that this reduction of the load case leads to thicker laminates than required, calling for more than twice as many plies and making composite materials unwmpetitive. The error however is not consistent: in other load cases, the predicted result calls for less plies than necessary, resulting in an unsafe design. We cannot afford this simplified approach.

D. Progressive Failure Prediction

One example of industry's dismissal of analysis is that they ignore matrix failure, the signal of initial laminate failure. The disagreement is whether matrix failure prim to ultimate failure in a multidirectional laminate should be considered. Industry claims that they are only interested in ultimate failure caused by fiber failure, not recognizing that fiber failure depends on matrix failure and must be accounted for in the analysis.

The difference between a failure criterion with mahix degradation and that without is shown in Figure 5 for an applied complex stress of {3,2,1} MPa. Note that for a design that ignores matrix degradation, unsafe laminates are selected. It is therefore an unconservative methodology.

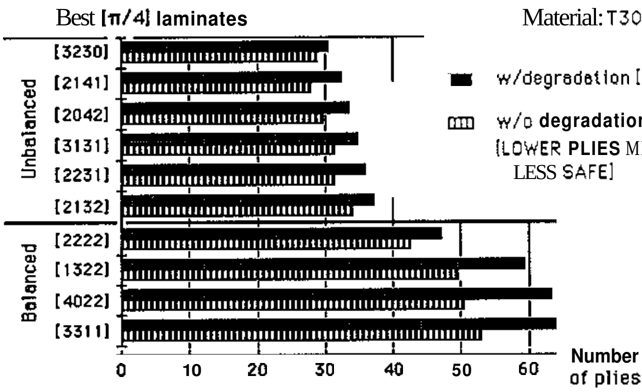


FIGURE **5** Number of plies required to sustain a load of {3,2,1} for different **π/4** laminates with and without mahix degradation theory. The sequence of numbers in brackets on the ordinate axis indicate the ratio of 0, 90, +45, -45 degree plies in the laminate; for example [3230] implies $[0_3, 90_2, 45_3, -45_0]_S$.

Another example of the comparison can be seen in Figure 6 where the strength ratios of two laminates are compared for a variety of loading vectors.

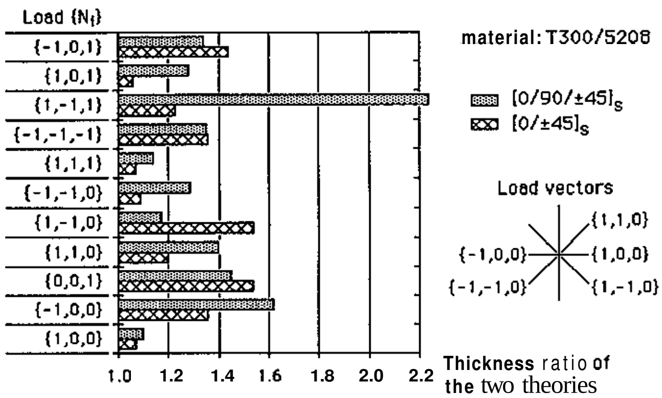


FIGURE 6 Ratio between number of plies required by theories with and without mahix degradation versus different **load** vectors for two laminates. Note that a ratio greater than one implies that a criterion without matrix degradation, like carpet plots, gives an unsafe design prediction.

The ratios on the abscissa axis show that the predicted strength of laminates designed using a criterion that **does** not consider matrix cracking will be higher (translating into a lesser required thickness) than that designed with one that does compensate for mahix cracks. The former over-estimates the strength, is therefore unconservative by a factor over **2.2** for some loadings. To date, we believe that such errors have been hidden by the multitude of safety factors imposed at **so** many steps in the design process.

The problem with ignoring progressive failure is that it severely handicaps the understandmg of composites behavior. Using metals terminology, initial matrix failure can be viewed as limit strength and fiber failure as ultimate strength. While the ratio of ultimate and limit strengths for metals is typically 1.5, this ratio for composites is very laminate dependent. A physical interpretation is that this **ratio** determines whether or not the failure is instantaneous and catastrophic

4. CONCLUSIONS AND RECOMMENDATIONS

At present, lamination is guided by simple rules, and failure predictions are generally made using the maximum strain failure criterion. Inherent in this failure criterion is the concept of strain allowables, which is easy to visualize but does not do justice to composite materials. Redundant safety factors, conservative design, and extensive empirical testing have resulted in many successful composites applications. Nevertheless, composites design hardly utilizes directionality, falling far from structural optimization.

Structural optimization requires the use of analysis. When analysis is used in composites design, it is the accepted laminated plate theory, usually introduced during the last stages of the design process. Laminated plate theory should be introduced earlier, along with a progressive failure criterion. This method provides a reasonable first approximation and better reliability than current design practises. This approach can then be developed into an integrated design methodology **as** outlined in Composites Design.

As **an** illustration of the levels of analysis, Table **1** is intended to summarize the analytical capabilities. Level 1, based on classical laminated plate theory (CPT) and the maximum failure criterion, is the accepted practice by many companies. Level 2, based on CPT and the quadratic failure criterion which includes progressive failure, should be considered the minimum. Level **3** is being developed: and Level **4** should be ready in the not too distant future. These levels are generic formulations of boundary-value problems. Problems of openings, bolted joints, and transverse impact are solvable within the same framework using different boundary conditions.

TABLE 1 PRESENT AND FUTURE LEVELS OF ANALYTICAL CAPABILITIES

Level	Noduli	Failure	Progress failure
1	CPT 2-D	Max strain	Not included
2	CPT 2-D	Quadratic	2-D included
3	Orthotropic 3-D	Intra - quadratic Inter - parabolic	2-D included
4	Nonoclinic 3-D	Intra - quadratic Inter - parabolic	3-D included

Our recommendation is that Level **3** should accommodate the analysis of thick laminates, with progressive failures for both intra- and interlaminar failures. A typical flow diagram may look like that shown in Figure 7.

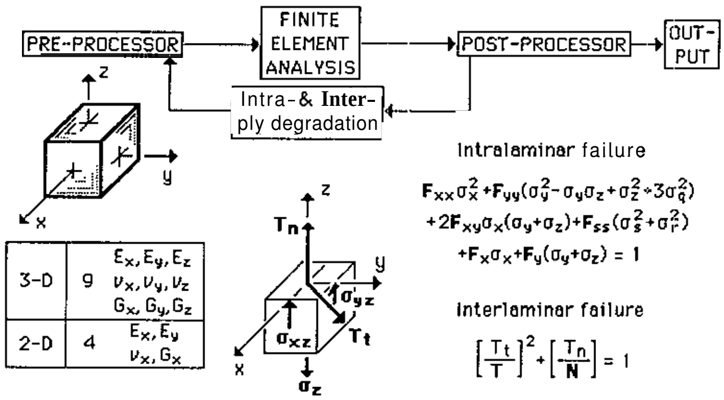


FIGURE 7 Key ideas of the Level **3** (orthotropic three-dimensional) theory scheme

The first generation three-dimensional model is best limited to homogenized, orthotropic materials. With this assumption, all off-axis plies must be grouped together. Hiving only nine independent constants, fast stress analysis is possible. The pmgressive failure analysis is conducted on a ply-by-ply basis. This approach has been applied using laminated plate theory on simple structures, such as a plate with a circular hole subjected to normal stresses, with good results. The effect of the hole size has been shown without resorting to the use of an empirical length for the point-stress or average-stress approximation. A truly three-dimensional approach is being developed.

It is hoped that as improved **stress** analysis and failure criteria are developed, the correlation between analytic prediction of failures will match experimental observation closer than can be achieved today. With improved tools, a factor of two weight reduction over aluminum will be routinely achievable in the structural utilization of composite materials. We expect to reach *this* capability in **less** than **five** years

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EVALUATION OF THE QUAI ATION OF THE STRUCTURE OF A PASSENGER AIRCRAFT BY ANAI S AND FULL-SCALE TESTINC

by

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ABSTRACT

On the basis of the experience with the FOKKER 100 development and full-scale testing, the qualification process leading to a certified aircraft structure is reviewed. The question whether the state of the art is satisfactory or not is discussed, seen from the viewpoint of the F.E.M. specialist, the manager, the structural specialist and the authorities. Special attention is given to the problems with derivative aircraft with respect to the requirements. Computer simulation as a replacement for full-scale testing is discussed and rejected. The practical compromises in full-scale testing and F.E.M. model verification are discussed and some crucial experiences with the testprogram are considered.

1. INTRODUCTION

In the aircraft industry there is a tradition of testing to support the analysis. This tradition is not the same in Europe and the USA and in both regions it is not such that the extent of a test program is easy to establish.
For completely new types the tradition leads more or less directly to a full-scale test program with two or three test articles. When the type is successful a series of derivatives may follow for which tradition and legislation are inadequate in defining the level of testing.

There are formal requirements, issued by J.A.A. and F.A.A.. but the articles are such that additional interpretative material is necessary and even then interpretation depends on several parameters of a subjective nature.
Especially for derivative aircraft, for which a basic design has been tested earlier, there is no definition of what is to be considered as "acceptable extrapolation".
This is even more difficult when a step by step approach to development is followed. Somewhere in this process the discussion will start whether new tests are necessary or not. Different opinions, depending on the different responsibilities, will arise. The F.E.M. specialist, the manager, the authorities and the stressman/designer may disagree. In the case of the FOKKER 100, after a discussion period which lasted for more than a year, a new full-scale test program, probably unprecedented in scope for a derivative aircraft was established.

Comparing different industries it is clear that the aircraft industry is in a favourable position because full-scale testing is a possibility. A full-scale test on a skyscraper for earthquake has to wait for a natural earthquake. Since there are more industries where there is no other way than reliance on other means of compliance than full-scale testing this workshop intends to investigate why the aircraft industry cannot rely mainly on improved analysis techniques.

2. VIEWPOINTS

The hypothetical viewpoints of different specialists will be considered. They are more or less caricaturistic in order to illustrate the situation of conflicting interests

2.1 THE VIEWPOINT OF THE F.E.M. SPECIALIST

Working in the field of FEM codes and testing of finite element behaviour, he is adept at designing FEM models for problems with a known theoretical solution and hopes to *see* his knowledge reflected in production models.
However, it is a horror for him to *see* that production models are composed by stress people who insufficiently understand the mathematics of discretization and who, under time pressure, are satisfied with geometrical compromises.

He will not be surprised by bad correlation with measuremerts. It is beyond his understanding that it is normal practice to predict failure loads with linear elastic analysis.
On the other hand he sees within reach the next generation of computers enabling another order of complexity in analysis with which he claimes true simulation of structural behaviour may become a reality, and analysis may offer many more possibilities than full-scale testing because the mathematical test article will be re-usable.

2.2 THE VIEWPOINT OF THE MANAGER

For this purpose the manager is defined as a specialist in getting things done and who has to take the decision where rationality fails or in the case of conflicting interests. Considering the level of structural integrity of the aircraft flying around the world and observing (in the cas? of the FOKKER 100) that in the static full-scale tests failures only occurred above the load level to be certified, he could be satisfied with the state of the aft in structures.
He could ask: "How can we save costs; can we omit these expensive tests when the only use of it is to show that the design team did a good job ?" and "How can this design team do such a job faster and cheaper ?"
He expects from automation a fast pre- and post processing replacing engineering judgement by a push-button procedure.
up till now the design team has asked for ever better computer power but this did not result in less man hours per weight unit. He is suspicious concerning improving analysis quality and he fears hobbyism. A method which prevents exploding analysis cost is to limit resources. There are a lot of other areas in which the money can be spent with more promising sales potential.

2.3 THE VIEWPOINT OF THE AUTHORITY

Authorities are not equipped to check the analysis in detail. They know that computer results and other paper work is one thing and an aircraft is another. Thert?have been a number of surprises during tests and with the operation of aircraft. At least in Europe they see full-scale tests as means of compliance with the requirements.
However, their position is more jurisprudential than technical. They are also victims of their own legislation as far as interpretation is concerned.
In the interpretative material the following is added to JAR (ACJ 25.3071 and illustrates the nature of the bargain:

"In deciding the need for and the extent of testing including the load levels to be achieved the following factors will be considered by the authority

a) The confidence which can be attached to the constructor',; overall experience in respect to certain types of aeroplanes in designing, building and testing aeroplanes
b) whether the aeroplane in question is a new type or a deve opment of an existing type having the same basic structural design and having been previously tested, and how far the static strength testing can be extrapolated to allow for development of the particular type of aeroplane.
c) The importance and value of detail and/or component testing including representation of parts of structure not being tested, and
d) the degree to which credit can be given for operating experience where it is a matter of importing for the first time an old type of aeroplane which has not been tested."

It requires detailed knowledge of the design parameters and foregoing experience to arrive at a subjective judgement. Based on these ground rules decisions need to be taken in the early stage of a project. Authorities do not like to be put in a position of taking part in decision making which they consider as the manufacturers responsibility. This means that pre-interpretation has to be done by the manufacturer and agreement sought from the authorities much later when sufficient information is available. This conflict of priorities is avoided when the manufacturer decides in an early stage to go for full-scale testing. If testing is considered necessary than JAR requires in principle ultimate loads whereas FAR states: "The administrator may require ultimate load tests in cases where limit load tests may be inadequate".

If it concerns a development of an existing type it can be expected that the manufacturer's interpretation differs from his authority's and even different authorities may have different opinions.

2.4 THE VIEWPOINT OF THE STRESSMAN/DESIGNER

The stressman especially is aware of the iterative nature of the design process which, in his perception, may not converge to the ideal solution. The final F.E.M. results based on final loads will come after drawing release and hopefully in time for certification. The design phase is a hectic period in which things do not always go along the lines he desires. in case of a derivative design there is a great pressure to maintain structural lay-outs as much as possible. It leads to a more efficient structure but there is a risk of going too far.

In the fixed period between go ahead and certification ever more jobs are crammed. Already during this period new variants upon the basic aircraft are started in the design office requiring the attention of the same people who are already overloaded. The task of damage tolerance justification is forced into this period as well.
Implementation of the damage tolerance requirements today is different from the expectation at the issue date of art. 25.571 in 1978. From the side of the legislator it was not intended that the available data for inspections should be better than a safe assumption; it could be updated along with improved knowledge. However, this approach was not to the satisfaction of the operators who require the final results on certification date.
As far as full-scale testing is concerned the stressman should believe in his own analysis and the conservatism of the assumptions. Added to that the knowledge that real structure should show better reserve factors than analysis because of positive geometry tolerances and better material data than the design values he should advocate the uselessness of testing. Nevertheless he is the one who has firstly to answer the question whether such testing is necessary or not. Although this question is confused by politics and company interests the stressman is considered to be able to come up with cheaper alternatives with sufficient conclusive force.

3. PECULIARITIES OF AIRCRAFT STRUCTURES

A very important difference with other industries is that aircraft are built in series production in numbers than can reach to over a thousand of a type.
As a means of public transport the highest level of safety is required; the impact of a failure on public opinion and on the industry is enormous.
The aircraft is the only means of public transport that cannot stop in its own environment and cannot operate at factors of safety appiied to the other forms of transportation. The requirement 25.303 imposes a factor of 1.5 on the external limit loads. These limit loads are not impossible in normal operation.
The structural concepts are such that non-linear elastic behaviour is accepted below limit load and non elastic behaviour below ultimate load.
To determine the allowable load for a structural part is often not a simple task. Simplifying assumptions are used for which it is not always possible to ascertain that they are conservative.
Allowable loads of structural elements have been established in general based on non-elastic behaviour but without feed back, up till now, to the FEM analysis in terms of stiffness reduction. Furthermore there is the influence of detail design on general instability phenomena and fatigue. This influence is generally too complex to take into account and often not revealed before failure on test or in service.
Aircraft structures are composed of many bits and pieces with a great number of attachments and fasteners.
The load transfer and local stresses follow a different pattern at ultimate load condition and under fatigue loading. (yielding versus elastic).
In many cases the influence of yielding is taken into account by very simple assumptions. The elastic stresses for fatigue loads in complex details could in theory be better analyzed but this requires an effort which is not feasible during the development phase.
Up till now the internal loads are calculated with linear elastic methods, also for ultimate external loads.
Considering the uncertainties at and below ultimate load one could have doubts whether the usual analysis procedure can produce sufficient means of compliance with article 25.303 or not.
Given the fact that full-scale testing is a practical possibility and, in relation to the development cost of a new project, also an economic possibility it is understandable that it became a tradition.

4 GENERAL CONSIDERATIONS ABOUT FULL-SCALE TESTING

Although full-scale testing for new projects is reasonably well defined in J.A.R., working groups still discuss whether it could be sufficient to test no further than limit load. In fact this is the basic F.A.A. requirement although occasionally in the USA ultimate load tests have been done. It is the author's opinion that a limit load test requirement makes no sense.
From a mature industry it can be expected that their products are anyway good enough to take limit load.
A strain gauge measurement program makes sense in the linear phase but this is a separate subject and not necessarily connected with limit load testing as a requirement. In this respect it is interesting to quote from (ref. 1).
"The accuracy of durability analysis will be improved by direct stress measurements from the full-scale test article eliminating the need for costly finite element analysis".

Somewhere between limit load and ultimate load uncertainties begin to appear. Often a premature failure is introduced by a local secondary deformation or even a real omission such as missing fasteners. The ultimate load test is obviously the only way to reveal such points or to show that they are not important below the tested ultimate value.
An inconsistency is introduced with the acceptance of a certain growth on the basis of extrapolation; acceptance of extrapolation based on subjective judgement and pragmatic legislation is not consistent with the turmoil which is produced by a test failure at load factor 1.48 instead of 1.5.

Only the failures are corrected which emerged under 1.5 and not the ones just above 1.5 because they remained unknown.

Since every successful type gets larger derivative:: tne truth of the matter is that not many aircraft in operation have been certified according to the test specification.
Should the basic type have been tested only up to limit load this could mean that after a growth period, only 70 % of limit load or less than 50 % of ultimate load is covered by test results.
A last opportunity is a test to failure. It is obvious that doing this only one load case can be tested when the damage is not repairable. Usually wing upward bending is chosen because it is in the manufacturer's interest to establish growth potential. The breakspot is then improved for further growth by extrapolation.

A remarkable precedent has been created by the Dutch RLD. In theory a downward wing bending test is at least as important as an upward bending because generally the lower skin has many more cut-outs.
RLD agreed with the omission of such a test for the FOKKER 100 provided that it could be shown that an additional safetyfactor in the order of 1.5 existed.
In practice this is not too difficult because the lower skin is designed for a fatigue life of 90.000 flights so the compressive stresses are relatively low.
Extending this precedent could mean that the normal safety factor of 1.5 is only valid in case a test is done, otherwise 2.25 is required.

As far as fatigue testing is concerned the European and the US requirements are also different although in the wake of the Aloha incident F.A.A. NPRM's are in the pipeline which should eliminate the differences. The main discussion item is the two life times full-scale test requirement but again derivative aircraft are difficult to compel into this concept and old timers for which it is really important were often not tested adequately.

Requirements is one thing, manufacturer's interest another.
The confidence which is created on the side of customers and authorities by a good test program is of great value. Revealing discrepancies early in the production phase is an important driver, confining costly corrections to fewer aircraft.
The timely solution of one discrepancy can pay back the total investment in testing.

Testing also has its imperfections.
It is unknown on which side of the scatter band a full-scale test result lies.
Despite good test results cracks may appear in service revealing that some details in the test load application were not representative or the typical flight definition does not correspond with later operation practice.

5. ANALYSIS QUALITY

Analysis is not confined to the definition of the F.E.M. idealization. Interpretation of the results and transforming them, if necessary, to enable the comparison with allowables is the real job of the stressman.
Judging the quality of this process as a hole is not easy; one niissed local problem might ruin the confidence in the total effort.
There are only incidental occasions for checking the quality since it can only be done by full-scale testing and even this check is not perfect.
There may be differences between manufacturers because of differences in resources, philosophies and experiences. Some general remarks about limitations of the analysis techniques have also been made in other paragraphs.
I will expand now a little on some peculiarities of the FOKKER 100 F.E.M. analysis. dated 1984, especially those which we intend to improve in the next generation.
Competition for computer capacity meant we had to choose between detailed modelling within components with debatable boundary conditions or optimizing component boundary conditions at the expense of some details in modelling.
We decided to opt for an almost complete aircraft model built up from substructures in order to get the best compromise in global and local stress distribution. The price of this was a lumped stringer concept (72 real fuselage stringers:=32 model stringers, 17 wing upper panel stiffeners:=9 model stiffeners, 28 lower wing panel stiffeners:=9 model stiffeners). Cabin windows were treated as panels with reduced stiffness. Cut outs for doors and hatches were rectangular with sharp corners. (fig. 1).
Fuselage frames were eccentric beam elements, coupled to the skin grid by shear cleats with low radial stiffness.
For cabin windows and door cut outs the F.E.M. analysis gives only general results which have to be interpreted by engineeringjudgement. Improvement of such details for fatigue analysis requires very fine modelling (plate bending effects should also be taken into account). A trial calculation with rounded off cut-out corners which could be compared with strain gauge readings from the fatigue test article (fig. 1d) was made after the static

failure (ref. par. 6.1.1. a4). The correlation was promising at the fatigue load level but for the ultimate load case the stresses were so far above yield stress values that no relation with the failure load could be established. A thorough investigation is still necessary to explain this.

The centre wing upper panel is subjected to a complicated loading system. Wing bending together with loads from crushing, cabin pressure and inertia loads supported by elastic ribs.
The F.E.M. analysis only produces the wing bending stresses to be used further in a compression bending calculation.

Research analysis has been performed by the National Aerospace Laboratory (N.L.R.) on the critical loaded stiffener with the computer program STAGS (ref.2). The collapse behaviour was investigated taking into account different combinations of the variables. Although the results are really interesting this analysis is far from the wish to simulate the complete upper panel with its curvature and varying load distribution in chordwise direction.
The collapse load comes remarkably close to that of the formal certification analysis in which the area where yielding takes place was assumed on another location.
Since this type of structure, as a consequence of weight growth, will be used up to its limits it is a strong desire to know the collapse load very accurately.
The NLR report closes with the remark that "In principle such complete simulation of the panel with all its details is coming within the range of possibilities. The only barrier at this moment is the tremendous computer costs connected with such undertaking".
Such a remark is outdated very soon; a new generation of hardware and software may change the world.

Although the analysis is done according to the requirements, combining ultimate wing bending with 1.5 x cabin pressure (max. relief valve setting). we used the escape clause that for testing a lower level can be agreed with the authority. We consider stability problems induced by cabin pressures above max. relief valve setting unrealistic and sufficiently covered by ultimate wing bending with a factor of 1.2 on the max. relief valve setting in a full-scale test.

The F.E.M. model, designed for static loads has also been used for the fatigue loads. The results, in terms of element loads, are in general too crude to be used directly in a damage tolerance analysis. They have to be translated by engineering judgement into detailed stress distributions in small areas of real structural elements, taking into account local eccentricities and secondary effects. The required accuracy is very high since there is an exponential relation between stresses and fatigue life.
Added to this is the incidence of substantial scatter in real hardware and the influence of parameters such as fretting, surface condition, residual stresses and manufacturing variations.
Only a large margin between required and calculated life may be considered a reliable guarantee of problem free life. But often a factor of two on the results means the difference between a serious problem and no problem at all. People working in this field are sometimes at their wits' end and not prepared to take responsibility for the conclusions.
There is a real gap between this world and the suggestion which is provoked by the requirement 25.571 that it should be possible at date of certification to deliver the ultimate proof.
A more accurate F.E.M. analysis can make this process more reliable but it remains only a part of the game. At least one positive thing can be mentioned about this imperfect process: It requires additional attention for the design details with respect to damage tolerance which means a quality improvement; gaining experience is the only way to improve methods.
Similar conclusions can be found in professor J. Schijve's paper (ref. 3).
quote:
1) Predictions for specific design situations can be considered as estimates only, which generally will have low and unknown accuracy. If they are still used for design purposes substantial safety factors should be introduced with the risk of undue weigh penalties.
2) In view of the limited accuracy of predictions. experimental verification is very much necessary if weight penalties should be avoided. From the previous discussion on full-scale testing it will be clear that such experiments should be carried out on realistic specimens with realistic flight-by-flight load sequences.

6. TESTING OF THE FOKKER 100

Considering this project as a derivative aircraft the discussion about the necessity of testing was difficult. It would become the first derivative with a full-scale test program which was a disadvantage against the competition in program cost. In the beginning the status of the project was to extend the F28 production line for a number of years. Nobody saw at that moment the sales success which developed. During the design phase the project status altered to a state of the art concept for the future.
Together with a number of geometry and weight increases this led to the conclusion that extrapolation from F28 testing had no credibility and a new test program was established.

This program consists of two full-scale test articles for wing and fuselage (static as well as fatigue), a test set up for the horizontal stabilizer with upper fin part and test set ups for engine support structure, moving surfaces and main undercarriage.
In this lecture attention is mainly confined to the two test articles for wing and fuselage (fig. 2).
In order to fit these tests into the building facilities one test rig was designed to perform both tests consecutively. This led to considerable complication in scheduling the program.

The total elapsed time, including the decision phase. is in the order of ten years.

1 year decision phase
3 years preparation phase
6 months static testing and test article change
2 years crack initiation
2 years crack growth and residual strength testing
1 year somewhere in the program for test article change and failure test

Risks in performing this scenario are numerous. Ultimate load testing should cover the design envelope but every load case has a potential failure risk jeopardizing further testing.
The endurance test will produce cracks. In order to gather knowledge about crack growth those cracks are allowed to grow much further than in operation but then repair may be much more complicated and not representative of the standard repair. When a redesign is necessary, either a retro mod or a series production mod, this should be fed back into the test but then life time until repair is missed or the test time should be extended.
The requirement to test two life times at a conservative spectrum is really a challenge; during the second life time damages might be expected which are beyond economic repair if the design is just good enough.

For the static test program 12 symmetric load cases were selected representing maximum bending and shear in wing and fuselage and a few cases with local importance. This was blown up to 45 test cases because first a series of limit load tests was done, then the ultimate loads and after that some cases at increased load levels.

The fatigue test is based on a conservative spectrum covering also the operation in heavy gust areas. This spectrum has been converted to a test spectrum by truncation on the high end and omission at the low end. The flight types and the sequence of loads in each flight have been fully randomized. The heaviest load is repeated once per 5000 flights. Details of this test spectrum were presented in a lecture at the ICAF conference 1985 (ref. 4). The test spectrum loads have been increased to 110 % in order to speed up the test experience. An economic repair life of 90.000 flights will be covered. With a scatter factor of two on the results this is 180.000 equivalent test flights. The test duration and test experiences will be corrected for the 110 % load level.

6.1 SOME INTERESTING TEST EXPERIENCES

6.1.1 Static tests

Four failures have been encountered.

a1) A stability failure of stiffeners in a beam supporting the aft pressure bulkhead, just below the required 2 X cabin pressure.
No modification was required because the specified load could be maintained.

a2) A stability failure in the tailcone 3% above the required load (stabilizer down load) (fig. 3).

a3) A tension type failure in the main undercarriage back up structure under lateral loading 6% above the required load.

a4) A tension type failure in the fuselage side panel just above the wing, starting from the aft emergency hatch cut-out, under high shear load from a landing case (fig. 4).
Although the load was above the specified load this case was superseded in the mean time as a result of main shock absorber drop test data.
Certification could only be obtained by modifying the shock absorber characteristics and new drop tests.

Testing only up to ultimate load would not have revealed these weak spots and extrapolation would have been accepted.

Although in general such failures are part of the game they are still a surprise. The exact failure phenomenon was not predicted and afterwards it is not always easy to explain it. In the case of a2) it is almost

impossible to distinguish primary and secondary damage. Case a4) taught us that the quality of the F.E.M. model from the design phase was inadequate and that the present state of the art (detailed cut-out corner idealization) gives a much better indication of local stresses.
Are we sure that this was the last lesson we need ?
There are still supporters of the vision that speed during the design phase is more important than sophistication !

6.1.2 Endurance test

Although scatter factors are applied to the test results in order to arrive at safe inspection instructions, the nominal test results may stili be considered as representative of what might happen in operation. Considering that a heavy test spectrum has been chosen, it might well be that in a real aircraft such cracks will develop much later if the particular aircraft encounters a lighter spectrum.

b l) Cracks developed in the corners of the emergency hatch cut-outs above the wing at 20.000 test flights and more (see fig. 5a and 5b).
This problem has a relation with the static failure a4).
A weak spot has been improved in series production by extending an existing doubler to postpone cracking and to create additional static strength. Repair patches on the test article must show the quality of the repair in the event of cracking.

b2) A fuseiage circumferential joint strip at the wing spar Stations developed cracks after 30.000 test flights (see fig. 6a and 6b).
This was a serious deficiency which could have resulted in the loss of aircraft if it had remained unknown.
A prompt modification action was started and no aircraft are flying with the original design. With a detailed F.E.M. model for the stringer coupling the stress at the location of the highest loaded rivet could be established and the phenomenon explained afterwards.
However, the stress department has to take the blame for not having payed enough attention to this joint in the design phase. The fatigue test has paid off itself !

b3) Rear spar web cracks developed from the attachments to the lower boom at 52.000 test flights and up (fig.7).
Corrected to 100 % load ievei this means weii over 70.000 unfactored flights and so only aircraft operating under a similar spectrum might expect such cracks before the 90.000 flights. However the inspection requirement and the eventual repair are a burden for the operator so a modification was considered necessary.
A more thorough fatigue analysis during the design phase could have revealed this short life. However the comparison with F28 types together with the service experience until then gave no reason for suspicion. The analysis presented for certification did not reveal this problem. Afterwards the results were adapted also to earlier models of the F28.

b4) A serious crack in the wing lower skin initiated from a circular cut-out for a fuel probe at 60.000 testflights (fig.8). The cause was a design omission (the increased panel strength had not been introduced in the same ratio in the cut-out compensation). A redesign was considered necessary.

b5) In the connecting elements of the forging and the built-up frame structure of the rear spar fuselage frame cracks developed at 45.000 test flights (fig. 9). A straight forward repair in case of cracks was not an acceptable approach because of the inspection burden for the operator. A terminating action has been prescribed by means of an improved connection. Corrected to the 100 % load level 45.000 test flights means 63000 unfactored flights. The predicted unfactored life based on measured stresses was 106.000 flights and 435.000 flights based on F.E.M. stresses.

b6) At 40.000 and 50.000 flights the first cracks were discovered in the piano hinges of the cargo doors.
The cracks initiate from a sharp machined edge which can be considered as an improper detail design.
(The test article is of the basic configuration with the small belly doors).

7. F.E.M. VERIFICATION

One of the formal objectives of a full-scale test is the verification of the analysis. Part of this is the F.E.M. analysis which, to a certain extent, can be verified by strain gauge measurements.
The question is whether the model maker created a good simulation of real structural behaviour.
Verification is confined to the linear elastic behaviour; in consequence only strain gauges on locations where this is the case can be used. Generally this excludes the verification of stresses in thin panels of the fuselage because of early buckling.

F.E.M. models have been designed to the state of the art of the available computer power and man power, so very often they do not adequately represent local stress gradients.exactly the subject of interest to the stressman who defines the straingauge locations. The number of strain gauges always has a limit below the desired number so only a relatively small number is suited for verification.

Also for the 'ideal' locations there are causes for discrepancies,

1) A special F.E.M. model for the test situation is in fact needed because the boundary conditions and the load applications are different from the real aircraft. This extra effort is not always done and then the comparison is made with the model which is used for aircraft certification.

2) Every F.E.M. model is still a discretization of reality.
Every new generation shows much more perfection an3 as long as this has an influence on the results the final model has still to come.

3) Secondary load paths which have not been incorporate3 in the model may contribute to the stress distribution.

In the FOKKER 100 case cause 1) is applicable and we had to wait for more than two years after certification for an opportunity to correct this.

The criterion for "good correlation" is often not investigated and may differ per location. Compared to the theoretical stress (real stress in a theoretical aircraft with all geometries of nominal values and the nominal Young's modulus), a measured stress which is 20% lower might have a reasonable probability and even more for machined forgings. In clean stiffened panel structure, like a wing upper panel. a better correlation must be expected but anything within 5% is at hazard.
Since the analysis has its own unknown deviations, for which 10% can not be considered as bad in a complicated structure, a measured stress which is 30% below the calculated value is not unconditionally a bad performance.
Higher stresses than caicuiated at important locations are anyway considered as baci correlation. The explanation for a deviation is difficult to find because, mostly, only an incomp ete picture is available of the stress pattern.
To bring a F.E.M. verification, if ever possible, to the scientific level of a physical experiment goes beyond the objectives of a commercial activity.
Nevertheless some useful lessons are learned from measurements which deviate from analysis. It directs us to the locations where we may achieve improvements in the F.E.M. models.

In the formai sense verification has only to show that the analysis is conservative and we are fortunate that is most cases this is true.

As a matter of fact the occasions for a verification by full-scale test are very rare and it is a one shot happening performed by a team without an earlier comparab e experience. The existence of a fatigue test article provides the opportunity to have second thoughts and to do local measurements during a longer time span.
Only authorities are in a position to be able to collect and compare knowledge from different sources in a relatively short time span but even on a global scale this experience is liinited.

8. F.E.M. ANALYSIS IN FUTURE PROJECTS

In the last 30 years rapidly increasing computer bower stimulated a considerable improvement in analysis quality.
There are still practical limitations in the way these techniques are thrown into gear. Also the nature of a design process is at odds with perfection in analysis.
The first improvement within reach is a better accuracy in the linear field so that sorne fatigue prone areas can be analyzed better and earlier. This requires more perfection in modelling and more coinputer power. This will not be the ultimate answer in fatigue analysis (ref. quotation in par. 41 but if pre- and post processing automation and computer power improve, it will be cashed.
Standard rules for sufficient model perfection are not available. The author is not aware of any case study in which the model for a representative aircraft structure has been refined until no further improvement of the results could be found. Automation can provide us the possibility to gain more knowledge in this respect.

Going beyond linear elastics is several steps ahead. In principle the methods are available but practical application is very limited. It requires another order of computer power and another approach in problem definition. Every load case has its own critical areas for which t i e model has to be designed. The reliability is doubtful. especially because secondary effects are difficult to incorporate or, more probably, are overlooked. But even forgoing secondary effects, the authorities require that the reliability of the analysis can be shown so the need for testing remains.

It is not easy to see where the profits lie of such an extension in F.E.M. efforts requiring high combined skills in structures and methods.
One could think of weight saving but if this is substantial then some inherent safety from the traditional approach is removed and new failure phenomena could emerge.
There is a tendency to *see* no limits to the application of computers. Especially the aerodynamicists are looking forward to the second next generation of super computers.
What is different in aerodynamics compared to the structures ?
Aerodynamicists have to reach there final answers before go ahead because they are essential for the geometrical definition. The last sophication can have a pay-off in aerodynamic efficiency. They are not hampered by structural details or changing loads but they have the free air available for element definition.
In structural analysis it is the other way around. Depending heavily on the solution of structural details and the last issue of loads the final analysis has to wait for the finalization of these input sources. The importance of the analysis is more formal; it has to show that despite the irregularities during the design process. in the end the structure is still strong enough for the latest set of loads. During the design process the F.E.M. analysis is improved as an interaction between design and analysis.

Do we need complex non linear methods for that or does this mean replacing one kind of engineering judgement for another ?
There seems to be an imbalance between the urge to sophistication in strength analysis and the level of accuracy of external loads.
Should we have to adopt a philosophy which sets limits to computerization where those limits no longer are dictated by hardware and software ?

Improved analysis quality removes to a certain extent one of the drivers for ultimate load testing. However the subject of testing is not so strongly related to analysis quality but more of a philosophical nature. One could argue that the added safety from tests is debatable. especially because we do not know where in the scatter band the result, good or bad, has to be situated.
In general we are working in the margings of the accepted probability of a failure.
Since it is impossible to produce a legislation for derivative aircraft which is consistent with the principles for a new design we should find a way that eliminates the controversal situation.
If the safetyfactor of *1.5* has its legitimation based on the tradition of testing then the consequence is testing or *a* higher safetyfactor for every situation which is not tested.

9. CONCLUSIONS

Authorities in Europe see the ultimate load test as an essential part of certification. Their organization is not able to verify the analysis in detail and the tradition of low safetyfactors is an important factor in not relying on analysis only.

The acceptance of a certain amount of extrapolation from earlier test experience must be possible for pragmatic reasons but is in principle an inconsistency. The floating interpretation may lead to varying conclusions among manufacturers and authorities.

Although test experiences sometimes seem to show that the state of the art in designing for a certain ultimate load has matured to great reliability, inside knowledge may reveal that the failure modes and loads were not predicted.
The analyst is still looking forward to the next opportunity. Then he can incorporate the lessons learned in the last round and take advantage of the latest improvements in software and hardware.

Although in F.E.M. analysis great achievements have been reached in the last decades, modelling technique is still in its experimental phase and mainly based on secluded experience.

The analysis quality in itself cannot guarantee the capabilities of a structure. Errors in detail design, in drawings and production instructions, in procedures and on the shop floor add their own deficiencies. Full-scale testing is the only possible way to cover the end product.

These conclusions are valid. for different reasons, for static loads as well as for fatigue loads.

For static strength substantiation the requirement for ultimate load tests could be traded against an increased safety factor. For damage tolerance substantiation it is already common practice to use different scatter factors for analysis and for test but reliance on analysis only is still not considered acceptable. The aging aircraft issue drives towards a fullscale test requirement also in the U.S.A.

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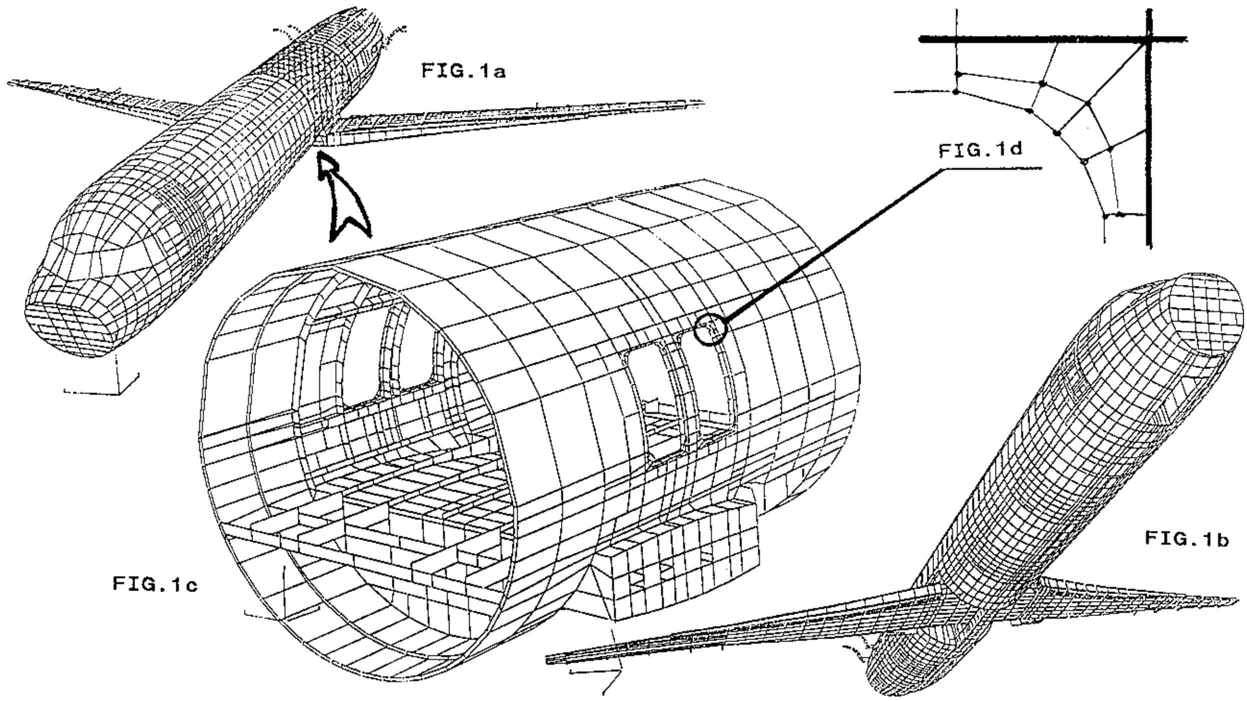
FIG.1a
FIG.1d
FIG.1b
FIG.1c

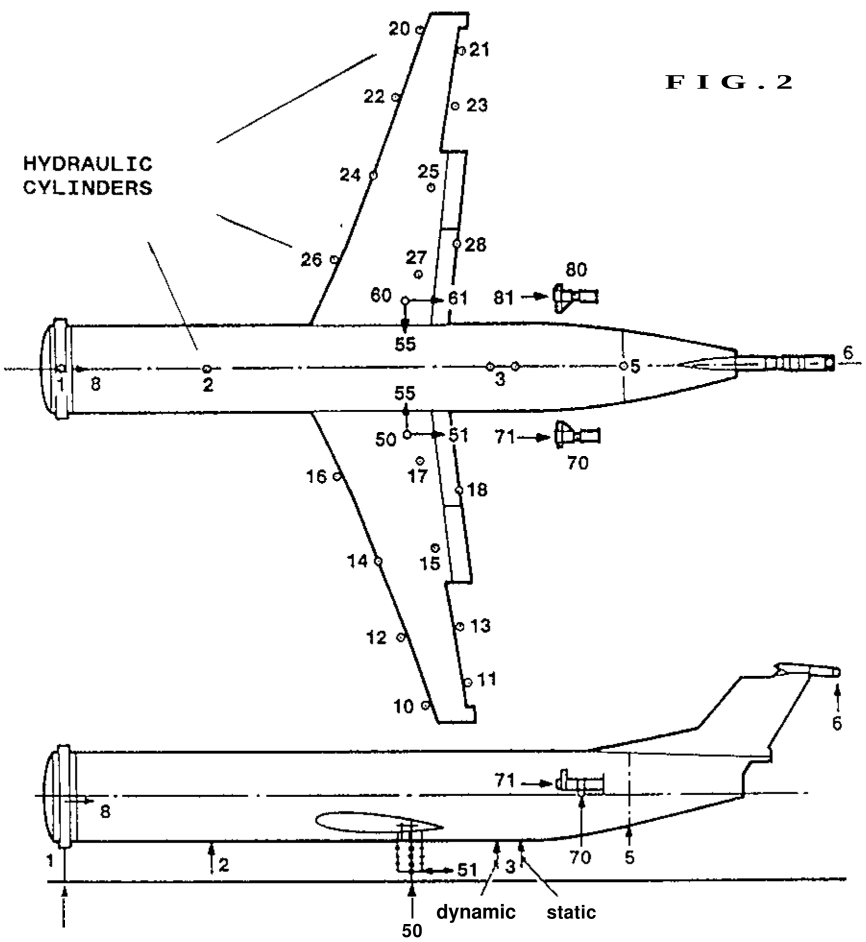
FIG.2
HYDRAULIC
CYLINDERS
dynamic
static

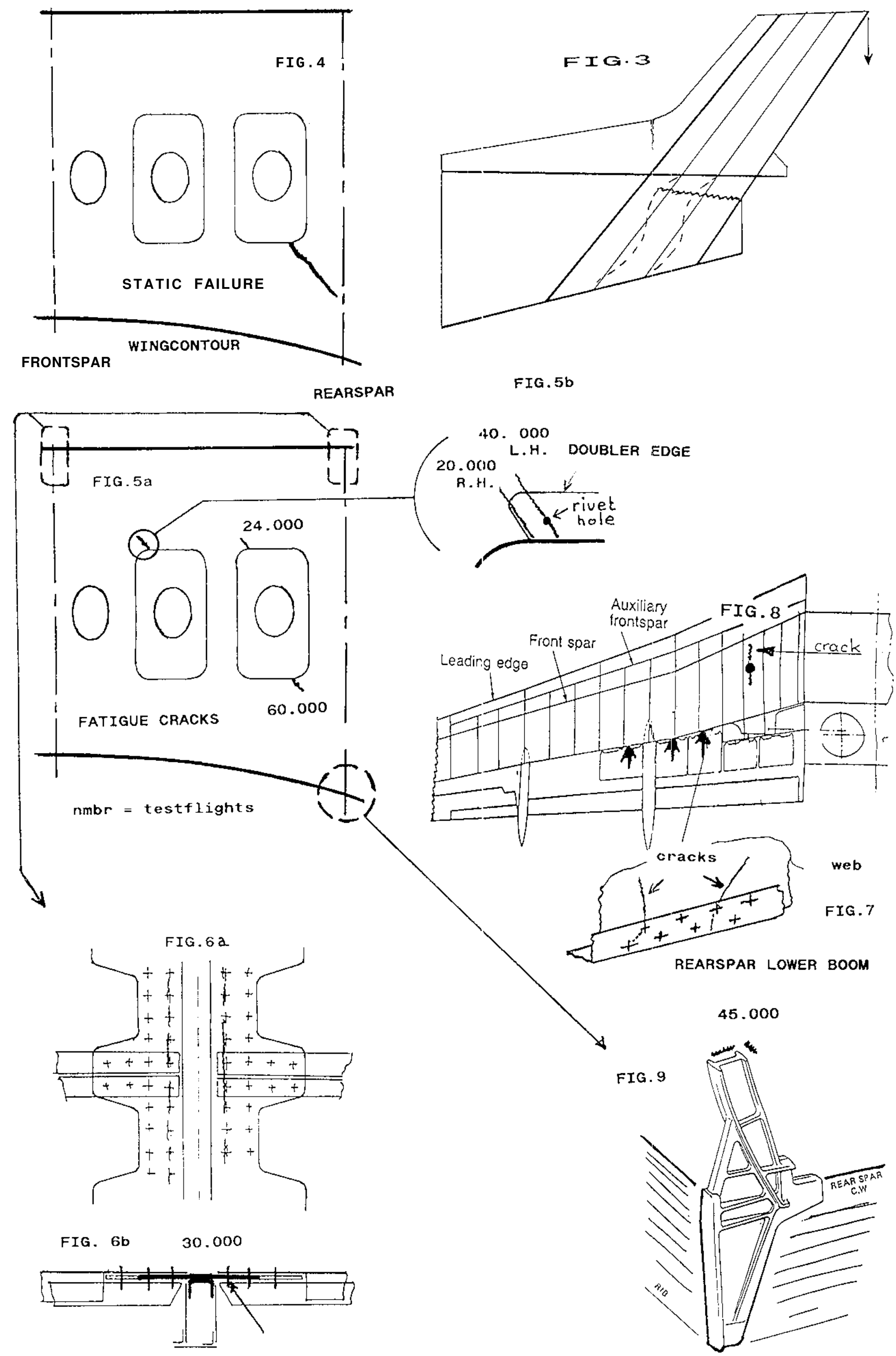
FIG.4
STATIC FAILURE
WINGCONTOUR
FRONTSPAR
REARSPAR
FIG.3
FIG.5b
40. 000
L.H. DOUBLER EDGE
20.000
R.H.
rivet
hole
FIG.5a
24.000
60.000
FATIGUE CRACKS
nmbr = testflights
Auxiliary
frontspar
FIG.8
Front spar
Leading edge
crack
cracks
web
FIG.7
REARSPAR LOWER BOOM
FIG.6a
45.000
FIG.9
REAR SPAR
C.W
FIG. 6b
30.000

DESIGN-ANALYSIS-TEST

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SUMMARY

The role of structural analysis in the Design-Analysis-Test cycle is addressed, and particularly the role of Finite Element Analysis. It is suggested that the dominance of finite element code application in structural analysis is moving from simple stress and deformation evaluation to the prediction of failure, particularly in areas where multiple testing would otherwise be necessary. The use of analysis to predict failure, instead of experimental tests, raises many problems, and these will be exacerbated if finite element codes become common tools of non-specialists. A way forward is suggested.

1 INTRODUCTION

In 1983 the National Agency for Finite Element Methods and Standards [Ref 1] was set up in the UK to address standards with a very wide brief in the use of finite element codes for structural and other analyses. The needs sparking such an organisation (and others are appearing [Ref 2]) are discussed later. In 1984 in a newsletter an article was published [Ref 3] which speculated on the prospects for finite element analysis replacing structural testing in the certification of structures in Aerospace and other safety-critical areas It seemed to strike a chord at an AGARD meeting at that time and two successive Structures and Materials panel meetings reviewed aspects of "qualification of structures by finite element methods". It is therefore timely to see if anything has changed or become clearer in the past seven years and whether the practical design-analysis-test cycle for Aerospace will change, or should change.

The orginal concerns of 1983 were somewhat naive and simplistic, namely that the CAD-CAM cycle was becoming so automated that the finite element method was becoming an embedded feature, and likely to be used by designers or engineers who knew nothing of the many sources of error present in the finite element modelling and analysis process. There are still available CAD-CAM packages with finite element systems and the CAD aspects have become much more sophisticated and user-friendly. fully able to take advantage of the massive resources available on modem graphics workstations. CAD packages market integrated design/analysis/automatic shape optimisation/parametric design all as a consequence of the commercial success and pressures in the CAD market. We should remember that CAD is more than an order of magnitude more important commercially than FEM, and an extra like "finite elements for dumb users" is another sales gimmick.

It is natural that the CAD-FEM interface should be broken down. No one wants to generate two separate models, and everyone wants instant design changes to be assessed immediately for safety, production, and costs. Yet what can be integrated into a CAD system is claimed to be a shadow of what finite element analysts actually wants [Ref 6]. There is much activity at the international level designed to produce agreed standard (neutral format) interfaces but the impasse has not yet been removed. Meanwhile commercial systems like CATIA-ELFINI claim to have removed it. Yet we are still some way from rapid design and redesign with structural integrity updated, or even better an optimum design incorporated. What is true is that the systems are much more interactive, friendlier, and rapid, and this trend is inexorable as the 10 Mflop workstations appear in 1990, to become 20 Mflops in 1992, etc etc. Thus one aspect of evolution cannot be ignored. Design and analysis is getting cheaper, and testing is getting more expensive. Even allowing for the fact that designs are becoming more ambitious in a competitive world, the incentive to replace test by analysis has not disappeared in the past seven years.

Another naive assumption is to suppose that CAD is purely a mechanical exercise to fit form to function and then to follow it with a CAE-CAM exercise to produce the article with minimal costs. At least in Aerospace the iterative design cycle is much more complex and involves the interdependence of many constraints: the extemal aerodynamic shape and controls, the intemal structure, aeroelastic coupling, the whole systems-driven targets and mission requirements, and of course the manufacturing constraints. Add to this an identical scenario for the engine and its system. and the chances of an automated design cycle may seem far away. However optimum design is now in common usage separately in configurational and aerodynamic design, in structural design, and in filling mission requirements. Thus although an embedded finite element package, untouched by human hand, is not yet a threat we should be prepared. However let us turn to the topic of whether finite element analysis will replace test in the airworthiness certification process.

The role of structural analysis in certification is covered at length in a separate article for this meeting by P. Bartholomew and only some of the conclusions will be repeated here. Finite element analysis has replaced traditional analysis even at the project stage for early identification of load paths, for the selection of worst cases, and the likely envelope for final selected tests - hopefully just one static, one dynamic, and one fatigue, supplemented by as many component tests as may be deemed necessary. Full scale testing is still mandatory for a new prototype but may be waived for future modifications, certainly-for civil aircraft if not for military aircraft where the ultimate design load is a much more frequent occurrence. Powerful computing and friendly software tools make it more likely that a refined FE model will be used ab initio and that condensation - the opposite of refinement - then follows for optimum design exercises or aeroelastic evaluation. Thus the role of analysis can be seen at one extreme merely as a way of defining a comprehensive test programme with a safe envelope, and of course in saving the manufacturer time and money in rapidly achieving an optimum design. No one doubts that the design and simulation cycle has been dramatically improved in both aerospace and engine design, but is the test always the final proof of airworthiness, and has finite element analysis made any difference?

2 THE IMPACT OF FEA

Classical analysis involved "exact solutions" to an approximate structure, constructed in the imagination of the skilled engineer and stressman. The FEM is an approximate solution to the nearest model to reality that the analysist can construct and the computer accept. Generations of stressmen became used to the idea of an analysis providing a reasonable prediction of "load paths" then local loads would be applied to a component such as a panel, shell, bulkhead, frame, beam, joint and so on. Data sheets would be (and still are) used to assess the capability of the component in withstanding the local loads. It is common to find modem finite element packages criticised for delivering detailed and complex stress fields which make the familiar data sheets impossible to use. This criticism is old fashioned of course, but is voiced even more in the fields of Civil, Mechanical, and Chemical Engineering where codes of practice have been evolved by trial and error to deal with local loadings. bending moments, and intemal forces. The main driver for rendering this attitude obsolete is of course the advenit of new materials and innovative configurations. Past conventions and practice will let us down in the use of novel configurations in metal, and even more for composite and ceramic structures. To produce even more data sheets, based on empiricism, theory and test, for the new generation of materials can be a very long and expensive process.

We assume therefore that, with enough computing power and time to construct models, the finite element method will deliver intemal stresses in a structure much better than traditional methods. The next step is to use the FEM, not just to predict stress levels, but to predict failure also. In aircraft structures "failure" usually means buckling in compression (followed later by material failure) and fracture, unstable crack growth, or fatigue in tension. Pure material failure other than fracture can occur in tension in local areas of stress concentration, particularly joints. All these failure processes have one thing in common - they are all non linear.

Non linear analysis has been thought of as an expensive business, and subject to algorithmic problems whether large deformations or elastopastic material behaviour. Computing resources are now cheap and some codes extremely robust, yet it is not common to hear of a non linear analysis used to predict failure. The author has not heard of a global incremental analysis of a complete aircraft, with non linear capabilities, so that a local failure anywhere is followed by load shedding and further failure elsewhere. Post buckling analysis of individual stiffened compression panels, and collapse mechanisms for mechanical joints are two local exceptions, and yet data sheets are still used (and manipulated) to find ultimate compression panel loads and joint strengths. Local analysis of cracks still tends to be linear elastic fracture mechanics with empirical or data sheet strategies to account for plasticity. The use of finite element analysis alone to predict the ultimate design load of a new structure is going to depend on its success in predicting failure. Unless confidence and credibility can be gained in predicting this in its many guises, then FE analysis will never replace testing. The route to successful and persistent success in failure prediction is a complex amalgam of many factors which will now be reviewed by example. Unfortunately success and failure in using finite element analysis in this field is often a closely guarded commercial asset, so many of the quoted examples have to come from the author's own experience.

3 STATIC AND DYNAMIC FAILURE PREDICTION

- Buckling and Post-buckling in Metal Panels

  As has been mentioned the use of data sheets for predicting buckling and post-buckling strength is well established, so it is suffcient to simply ask the FEM to deliver local panel loading. However it is often necessary to make assumptions about the buckling mode of stiffened panels: is it an overall mode or a local mode? Computer codes tends to be used as stand-alone aids rather than a re-run of the main code. Once such code VIPASA [Ref 7] and its later NASA development [Ref 8] is well known and it makes the minimum of assumptions about the buckling modes; e.g. plate stiffner node lines are not assumed to remain straight VIPASA is a two-dimensional code (a finite strip method) which cannot predict the coupling which may occur when an overall mode in an eccentrically loaded plate causes local stresses which precipitate local buckling. Some modelling expertise is also called for - as in most analysis. For example, how does one represent the restraints offered by mechanically joined stiffners in fig.1?

  Dedicated codes, particularly now that they are easily mounted on workstations with graphics I/O and interrogation are a powerful diagnostic. VIPASA has an early success in explaining unexpected buckling modes in the space shuttle in the mid-70s and its decendents are in continual use today. A simplified code has the advantage that it can be incorporated in an optimum design cycle with no great overheads when a large number of iterations are performed. Computing times may be between 100 or 10,000 times faster than a general purpose fmite element program [Ref 18]. However these codes are at risk when used to predict failure as mentioned and discussed again in a later section.

- Fatigue and crack growth

  One of the strongest arguments against omitting a full scale test is the fatigue problem. Because fatigue sources are stress concentrations there is a fear that the finite element modeller may miss the crucial source - even though the analysis is perfectly capable of delivering three-dimensional stress concentration factors and stress intensity factors. Surely this can no longer be true? The problem remains of course of simulating fatigue damage before cracks, or very small cracks. The civil route of defining inspectable crack lengths, and therefore restricting the analysis to prediction of stress intensity factor and unstable growth, may not be an option for military aircraft. The prediction for crack growth under a complex loading history is less straight forward. Some civil aircraft, or military transports, do have very odd loading histories involving many peak loads of either sign, and hence the likelihood of retardation (or acceleration) of normal cyclic crack growth under constant amplitudes. It is unlikely that all feasible spectra for all possible mission profiles can be tested for all possible crack sources. A theoretical prediction, particularly the effect of crack stoppers is desirable. In particular different material properties should be capable of being injected into the model.

An attempt to construct the finite element solution of cyclic crack growth, at amplitudes much less than the static unstable crack growth level, was made at Imperial College [Ref 10]. The entire elasto-plastic material behaviour in the vicinity of the crack was simulated, including both kinematic and isotropic: hardening. Crack growth was tested using an energy release (G) device for the chosen FE mesh and taking critical values G_c to include the crack tip plastic wake left behind Initially it appeared successful since crack sharpening under a high overload was successfully predicted (see Fig 2) but the consequent increase in G was not sufficient to explain below-threshold growth. It was realised that the usual strategy of allowing a virtual crack extension to evaluate G was not a true physical simulation of cyclic crack growth. It is necessary to allow the plastic slip lines to create minute fresh surfaces without crack opening, one for loading and the other for un-loading. This needs a completely new numerical analysis with finite strain slip lines, so a common ruse was then deployed, using a mixture of FEA coupled with empiricism The fmite element model is used to predict the crack closure - or opening - after high overload or underload. This closure is then used to update the usual Paris law which relies of course on two experimentally determined coefficients. However the predictions with retardations are good as Fig.3 shows. A mixed strategy of finite element simulation plus empiricism is an acceptable compromise, and is still much cheaper than test. It can be used with crack stoppers such as changes in plate thickness.

o Impact in metals - low and high velocity

Low velocity impact in metal structures is local arid not really a problem. Whole aircraft crash worthiness is an issue, particularly for civil aircraft or military helicopters. Simulation of this type of low lelocity impact is relatively straight forward using either lumped beam models (KRASH), [Ref 12] or genuine FE models (DYNA 3-D, DYCAST, etc). Possibly the only limitation favouring the KRASH approach is the computing demands when modelling a complete aircraft or helicopter structure with more than 20,000 unknowns. A simplified elasto-plastic shell element ICIS [Ref. 13] has been shown to be adequate and far less demanding on computing resource. High speed (ballistic) impact is also amenable to simulation and DYNA 3-D is widely used for bird innpact or armour penetration. Ballistic impact penetration in aircraft is not usually a structural issue since the projectile enters and leaves the structure cleanly, unless it is explosive as well. The same is not m e of fuel-filled wing boxes where the kinetic energy is converted to pressure - Hydraulic Shock or Ram. This mode of destruction is also simulatable but the marriage of fluid Euler codes and a structural finite element code can be expensive. A finite element fluid code (HYDRO, [Ref. 14]) has been constructed for hydraulic shock-structure interaction. The fluid finite elements are simply acoustic with no flow, but cavitation can be modelled and agreement with test is good. (See Fig. 4). However many modelling mistakes were made on the way, including the omission of high pressure water flow back through the entry face, the moving cavitation profile through fixed Lagrangian elements, and the sensititivity of peak pressures to projectile entry details. The big problem here is the carbon composite wing box which is more susceptible to impact and whose materials' strain rate effects are still improperly understood. [Ref 15].

o Dynamics

All finite element codes have a dynamic capability, both steady state and transient. Recent benchmark studies by NAFEMS [Ref 16] reveal that most codes are good in predicting natural modes and frequencies - only a few missed some modes. However these benchmarks essentially test the code and not the user. A parallel bench marking exercise [Ref.19] recently conducted chose a fairly simple fabricated structure, consisting of a cylinder mounted on a beam, and asked 12 code users and developers to model it. The results in the following table show the worst predictions for nine modes were more than 50% different from the experimental values, either way. It was no consolation that there was every bit as much scatter in the frequency response from experimental tests conducted on the same structure by a variety of research and commercial organisations! These large discrepancies were due usually to poor joint modelling - the bete noire of dynamicists. Joints are non linear and have unknown damping characteristics. The only solution to this problem does seem to be a database of joint properties derived by a large number of organisations in the private and public sectors - like AGARD for example.

	Rigid Body Modes			Beam Bending			Cylinder Bending		
	R1↑	R2→	R3↻	B1	B2	B3	C1	C2	C3
Minimum	64	43	57	87	85	78	94	77	98
Maximum	100	94	159	151	150	129	126	114	103

F.E. versus **Test**

The ability to predict dynamic response is most crucial for helicopters of course, and the separation needed by rotor forcing frequencies is embarrassingly small for typical helicopter structures, often of order only 4 cycles per second. Nevertheless one helicopter (the WG30) has been cleared without test. purely by FE prediction. This machine differs considerably from the LYNX military helicopter from which it is derived.

4 COMPOSITE STRUCTURES - AND FAILURE

All of the uncertainties of analysis, material properties, modes of failure., and modelling, are compounded when we switch from metals to composites. Such is the degree of uncertainty of carbon fibre composite modes of failure of full scale components, that severe weight penalties have often been built in to some designs by the use of "knock down" factors to allow for material variability or environmental degredation. Yet such factors of order two must be removed, for carbon composites to show their true weight saving and increased performance. The threat of degredation in compression, due to mamx moisture absorption, is still with us. Its absorption can be analytically predicted and the property degregation estimated reasonably well. This is fortunately since airworthiness requirements effectively prohibit more than 15% degredation in properties, and do not even recommend a major test with a degraded structure. Other modelling problems remain.

- Through-thickness stresses in composite plates

  Some years ago we looked at the post buckling strength of simply supported CFC plates at a time when industry was too unsure to let this be even a design case [Ref. 20]. At the time the only way to predict post buckling failure threshold due to bending strains was to use a mythical failure value of 5000 microstrain. A careful and controlled experimental test programme revealed that failure was actually due to interlaminar shear stresses at an edge where a detailed finite element analysis revealed the horrendous picture shown in Fig. 5. Nevertheless the FE prediction was good and results agreed to within +/- 16% of test failure compared to +/- 60% when using the wrong fibre strain criteria. This programme demonstrated a new need for FEA, that is to confirm the failure.sources in the interior of a laminate when the only validation the experimental test will provide is based on surface strains, ultrasonic scanning (which does not identify the mode) or fractographic post mortems after failure has occurred. Smart fibre technology for monitoring the health of inter-laminar zones is still a long way off.

Delamination failure modes are now recognised as the prime source of panel failure at edges or at the intersections of plate stiffners. Three dimensional FE studies are undertaken using brick elements per laminar, or, if the model is too intricate or expensive, peel tests may be conducted to predict stiffner debonding at typical interfaces [Ref. 21]. Future progress depends on commercial composite codes deploying in a laminated plate element the facility to deduce inter laminar shear and pealing stresses, by (say) integrating the equilibrium equations through the thickness if possible. However stress based criteria is only a half measure and any FE analysis is likely to deduce infinite stresses at free edges or component interfaces if the idealisation is very refined. A fracture approach should really be adopted. Some progress has been made in introducing energy release strategies from defined flaws at composite plate edges or component interfaces. This must be the way forward, but the problem is firstly finding reliable data on the criiical values for G_I, G_{II}, G_{III}, particularly the last-named. The: second problem is the fact most debonding or delamination is a mixed mode, and the correct interaction formula is not yet agreed. Even the crude strategy of using total energy release is still better than using permissible stress or strain criteria A study of simple coupon tensile failure reported experimentally by many workers, revealed that simulated fracture from an edge was able to predict failure loads for 8 separate sets of laminate tests to within 15% at worst compared with errors greater than 60% for stress based criteria [Ref 22].

The whole topic of failure and material degredation laminar by laminar cannot be said yet to be a robust feature of any code. 3-dimensional brick idealisation per laminar cannot be an economically feasible model in the long term Partial and gradual degredation of the plates' stiffness matrix must be a favoured route. There is not even a favoured treatment of simple stress or strain based failure criteria. Do we use an equivalent measure (like Tsai-Wu) or allow component failure like in-plane tension, compression, inter laminar stress, matrix or fibre, and so identify the mode explicitly. The latter must be the aim surely with such disparate strength modes [Ref 18].

A new pressure to model failure rather test, is coming from the explosive growth in novel composite material combinations of carbon, glass, aramid, metalic, ceramic, fibres and an even more bewildering choice of matrix materials. The combinations of fibre/particles and supporting matrix has become so very large we would like to deduce material failure from the micro-mechanics of its components [Ref 27]. Testing all prospective combinations does not Seem feasible.

- Damage tolerant frameworks

Another idiosyncratic example of poor matrix modelling occurred when we investigated the performance of a damaged tolerance prototype for helicopter fuselages and tail booms. The idea was to construct an (old fashioned?) framework covered with a non-structural cladding which would blow off easily if penetrated by an explosive shell. The framework was constructed by laying carbon fibre tape along the orthogonal geodesics and overlapping at the joints as shown in Fig 6. There was some concern that this structure would behave like unstiffened thin shells which are notoriously sensitive to imperfection - and the offset joints guaranteed that imperfection. A routine NASTRAN analysis gave very poor agreement with experimental tests, so it was improved by designing a new beam element which had both continuously varying curvature and twist [Refs. 23 and 24]. The errors were reduced from 100% to 30%, but after many blind-alleys the gap was finally closed when it was realised that the epoxy matrix in the joints was allowing a finite rotation between beams. Any finite element programme of course does not permit this. On introducing therefore a "swivel-stiffness" the code predictions then worked and Fig.6 shows the rather extraordinary performance whereby a structural modification does not change the linear loaddeflection characteristic at all, but as initial buckling is approached the new divergence occurs much earlier. The curves also demonstrate that the "shell" is only gently imperfection sensitive. Failure is pure instability and hardly needs stress criteria to predict it - although this has also been done.

- Impact simulation

The success in simulating both high and low velocity impact in elasto-plastic metal structures has already been mentioned. The need if anything is greater in composite structures which have relatively brittle fibres and matrix, cannot rely on gross plasticity to absorb impacting kinetic energy, and therefore may be exceptionally sensitive to how much elastic strain energy plays a role in surviving the impact process. At high velocities inertia forces inhibit the shuctural response; fracture damage and perforation tends to be the dominant mode and this can be predicted well using conventional empirical ballistic methods and appropriate fracture energies. If numerical simulation is deemed unrealistic then ballistic firing (excluding hydraulic shock) at representative plate samples is not an expensive process. The same is not hue of low velocity impact due to handling during manufacture, maintenance, or service, military or civil. Because impact forces cause peak shears in the interior, the threat of barely visible impact damage, and a hidden loss of residual strength, is always there. The use of coupon tests here is not so convincing. The kinetic energy is not the sole criteria in a full scale structure where the dynamic response would not be the same as a laboratory coupon. It will depend on impact velocity, ratio of impactor and structural inertias, local stiffness features and geometry-dependant load paths. Because there are so many low velocity impact scenarios in a complete aircraft, surely numerical simulation has to be used if only to identify trouble spots for testing or assessing damage tolerance potential before prototype and materials are firmed. A current EUROMART project (now BRITE-EURAM) addresses this whole problem and two of the eight partners aim to simulate both the 3-dimensional impact induced stress field (using DYNA 3-D) and the post damage failure process in either compression or tension. As a compromise our contribution is to propose a finite element inpact code to deliver a history of the force and the bending strains, and then use a single coupon test as a calibrator - for a given laminate - to relate damage thresholds to these two quantities. The code predictions for this approach look promising as Fig. 7 indicates. The entire programe is timed for completion in mid 1992.

- FEA credibility

Having identified FE modelling of failure as a crucial pivot in a strategy to reduce dependancy on test, we should question the areas of weakness in this analysis process.

- QA of the analysis procedure

  This topic is vital, particularly with several sites or companies involved in national or international collaboration. The parallel paper by Barlow [Ref 25] addresses this aspect succinctly, and it is hoped by NAFEMS that QA procedures will be agreed and monitored by organisations involved in the design of safety critical structures. This need in future may be written into contracts by customers or governments.

- FE codes

  Testing machines are expected to meet standards of acceptability, and therefore analytical tools should also. No code will every be bug-free, but competition to be near perfect and robust as well is a real driver. The activities of NAFEMS [Ref 1] in promoting standards, designing benchmarks, and providing educational aids has been recognosed internationally by both users and code developers, who have been extremely supportive.

- Analysts

  If the testing machine has to meet standards, so must the tester. The Aerospace industry pioneered the fmite element method and clearly has always had the experts and the on-site codes. This is changing and most companies are now using commercial finite element codes which are designed to be increasingly used by designers not necessarily analysts. The danger of not appreciating shortcomings in the method and the idealisations must be there. One safeguard is to build in diagnostics into the code, these can reveal poorly described models or badly conditioned solution procedures. However an exact solution to a poor model is no comfort. Expert system aids to modelling have been iminent for some time (!) but have as yet made no significant commercial impact A recent review Of expert systems in finite element analysis [Ref 26] identifies the areas where an intelligent assistant is most needed, but it is also suggested that for the very long term real expert systems will be used to create models From a very tight configurational specification. At present probably the best route to competent modelling is to rely on the design modelling and analytical expertise of the design organisation. For such organisations an accreditation scheme and implementation is outlined by Barlow [Ref.25] in a parallel paper. ECE harmonization after 1992 will accelerate the need for accreditation and certificates of competence:.

If codes, analysts, and QA management procedures meet acceptable standards then the parallel improvements in the computing power of workstations, plus more ambitious pre and post processing aids, must combine to make the finite element analysis procedure a more reliable tool. The biggest challenge, and the source of most errors, will continue to be in the modelling. Structural tests can be guaranteed to reveal the (unexpected) sources of failure in a full scale structure. Finite element analysis will only reveal what the modeller has put in. Perhaps the growing computer hardware and graphics software capabilities will also reduce modelling errors since since many arise because of attempts to control the degrees of freedom. If engineers feel free to use very refined meshes and 3-dimensional elements rather than to attempt approximations, error sources will be reduced.

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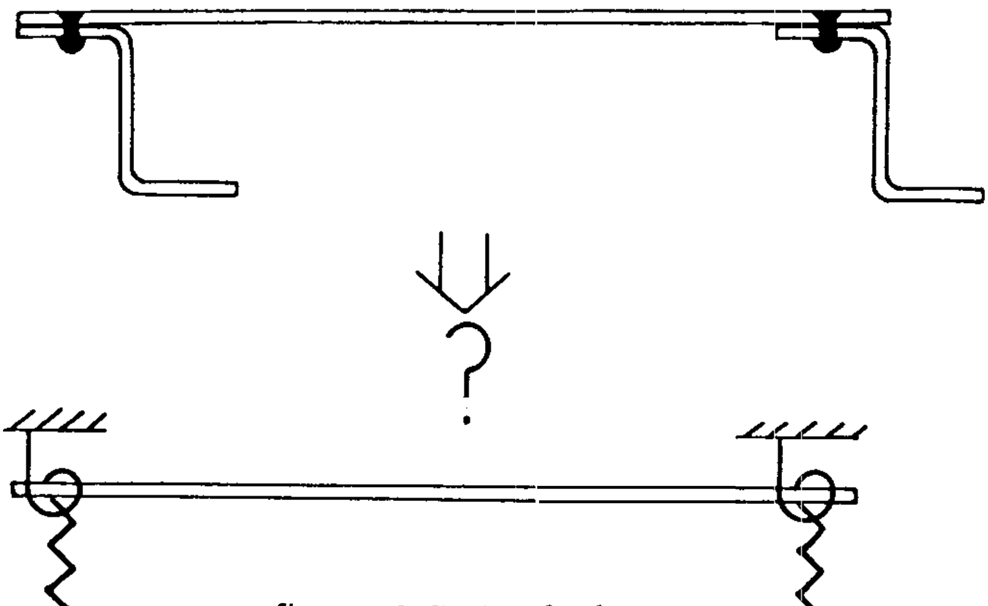


fig. 1. Idealisation of real structure

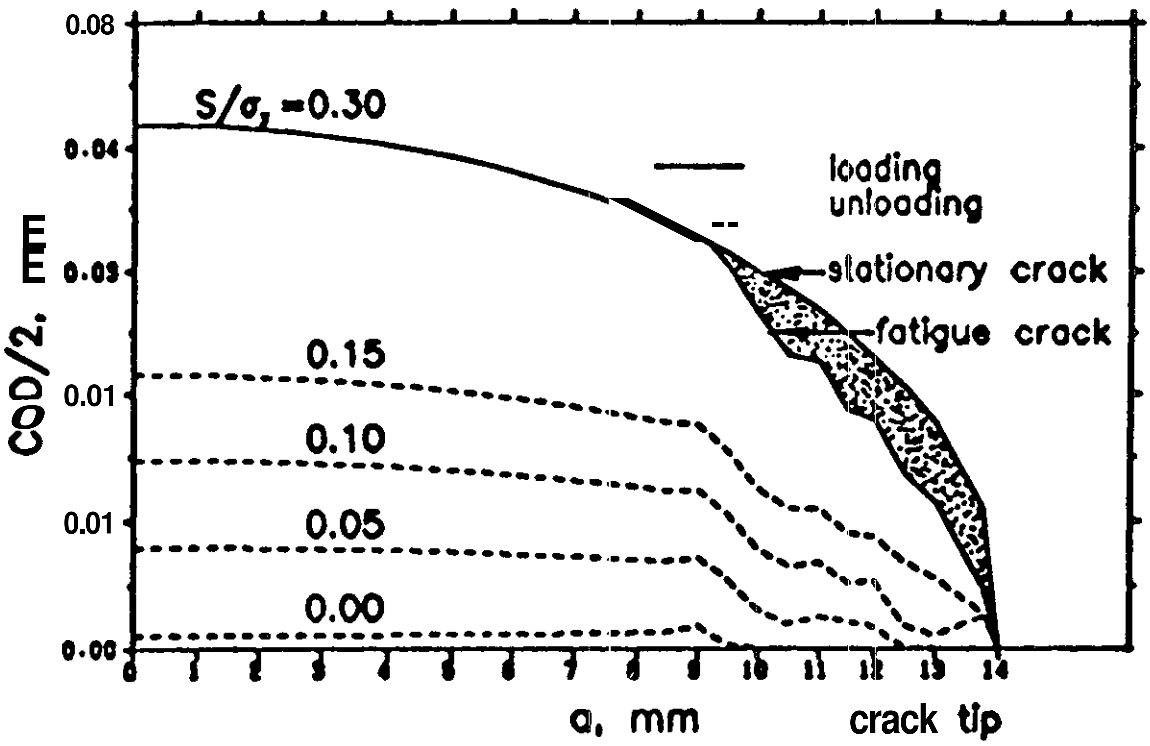


fig. 2. Crack-opening displacement profiles with and without cycling

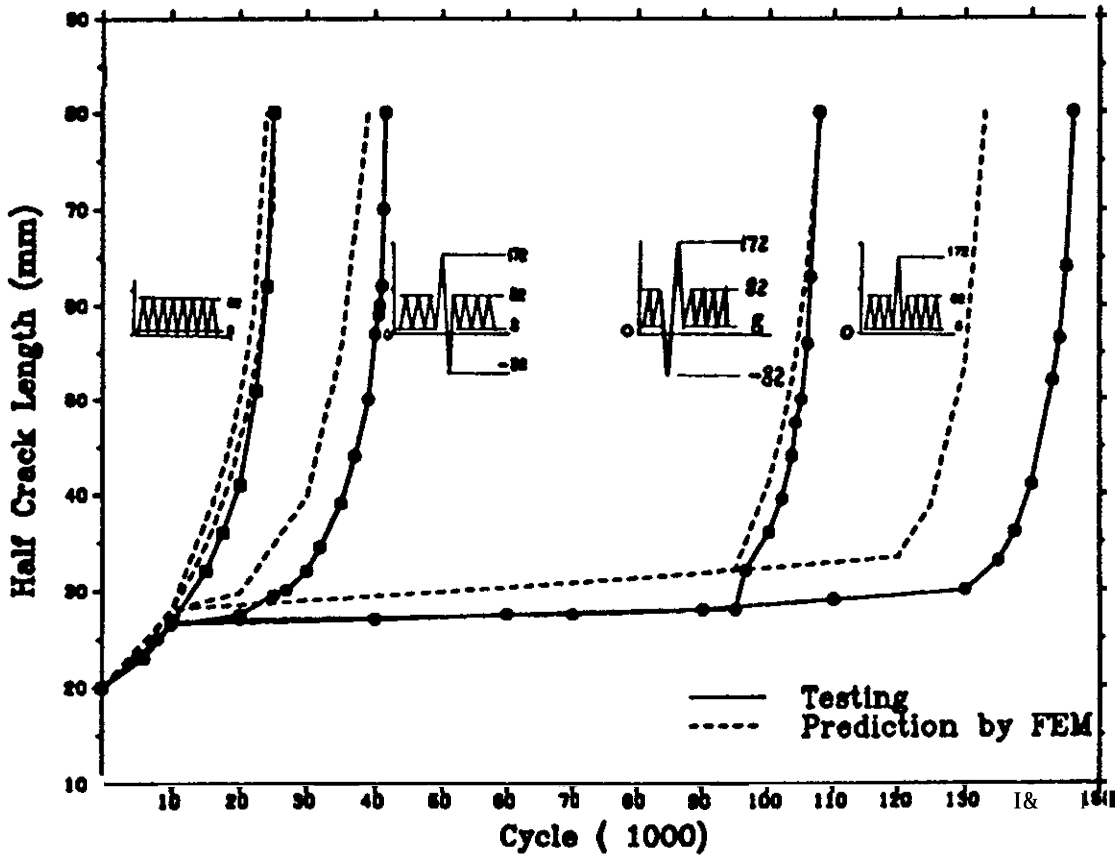


fig. 3. predicted and test crack propogation histories with overload and underload

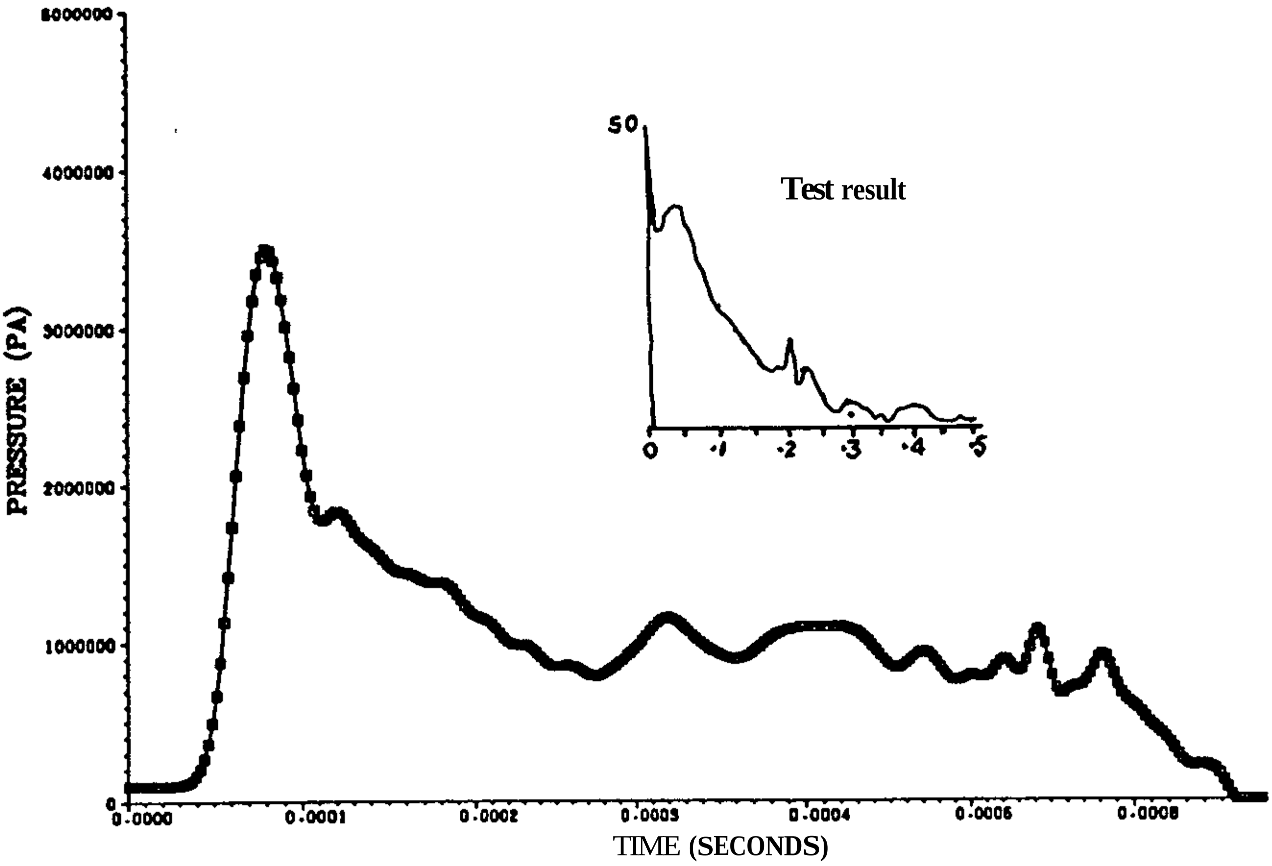


fig. 4a. Hydraulic Shock: pressure prediction using finite element d e l

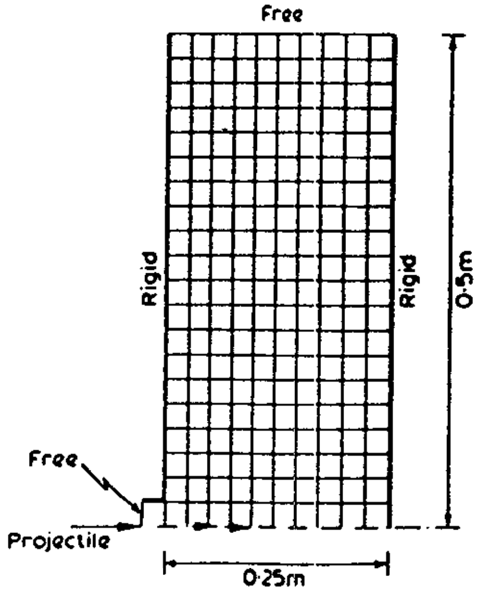
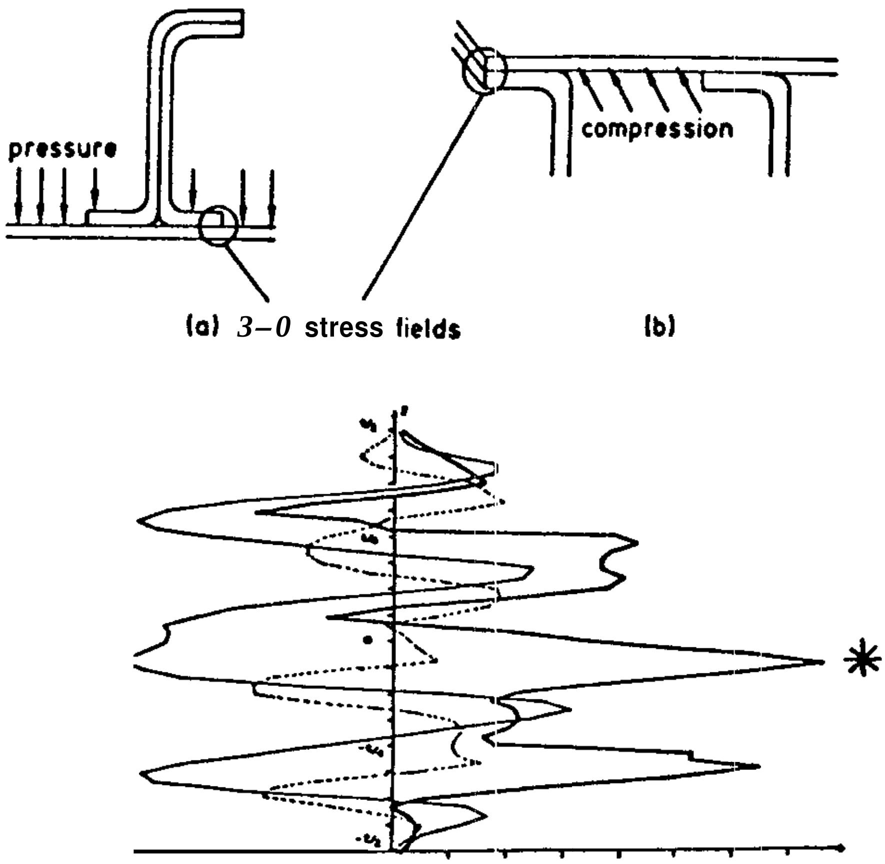


fig. 4b. Coarse fmite element mesh for fluid



✳ **Test confirmed delamination here**

fig. 5. **Through-thickness variation of all 3 stress components near laminate edge**

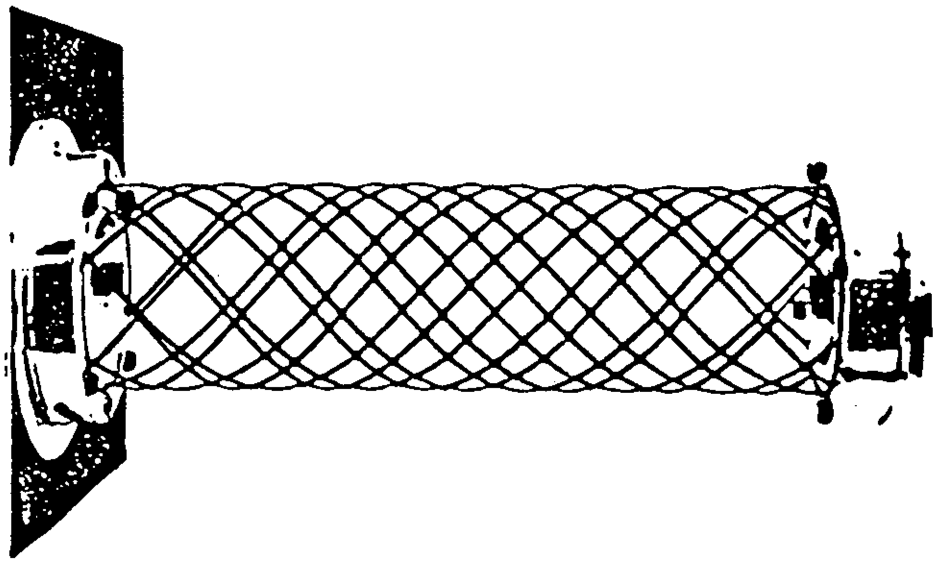


fig. 6a. C.F.C. Geodetic Framework

fig. 6b. Detail of joint construction

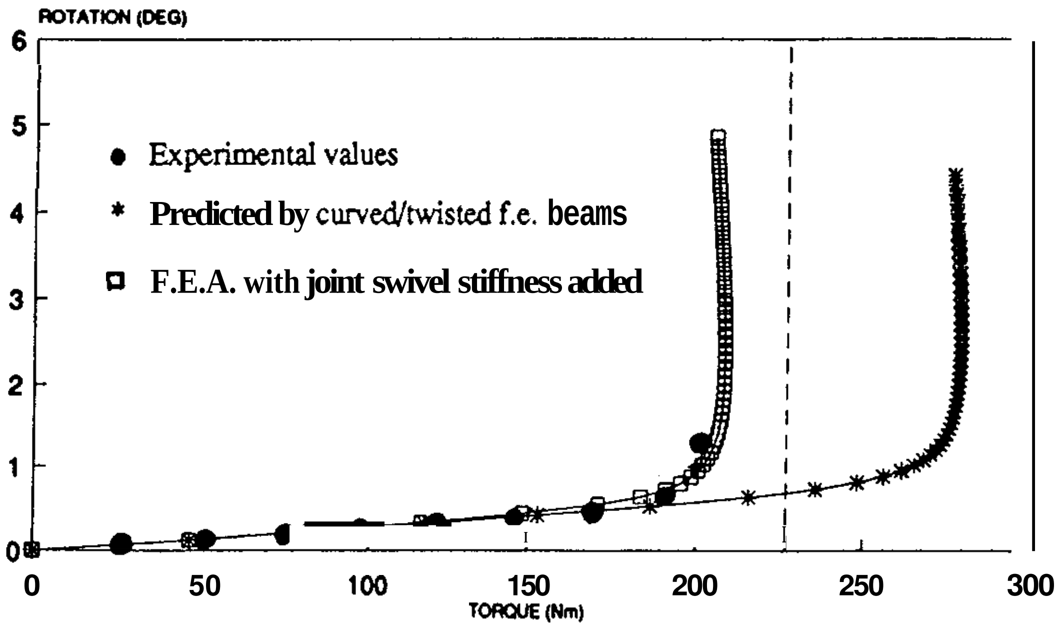


fig. 6c. Buckling under torsion

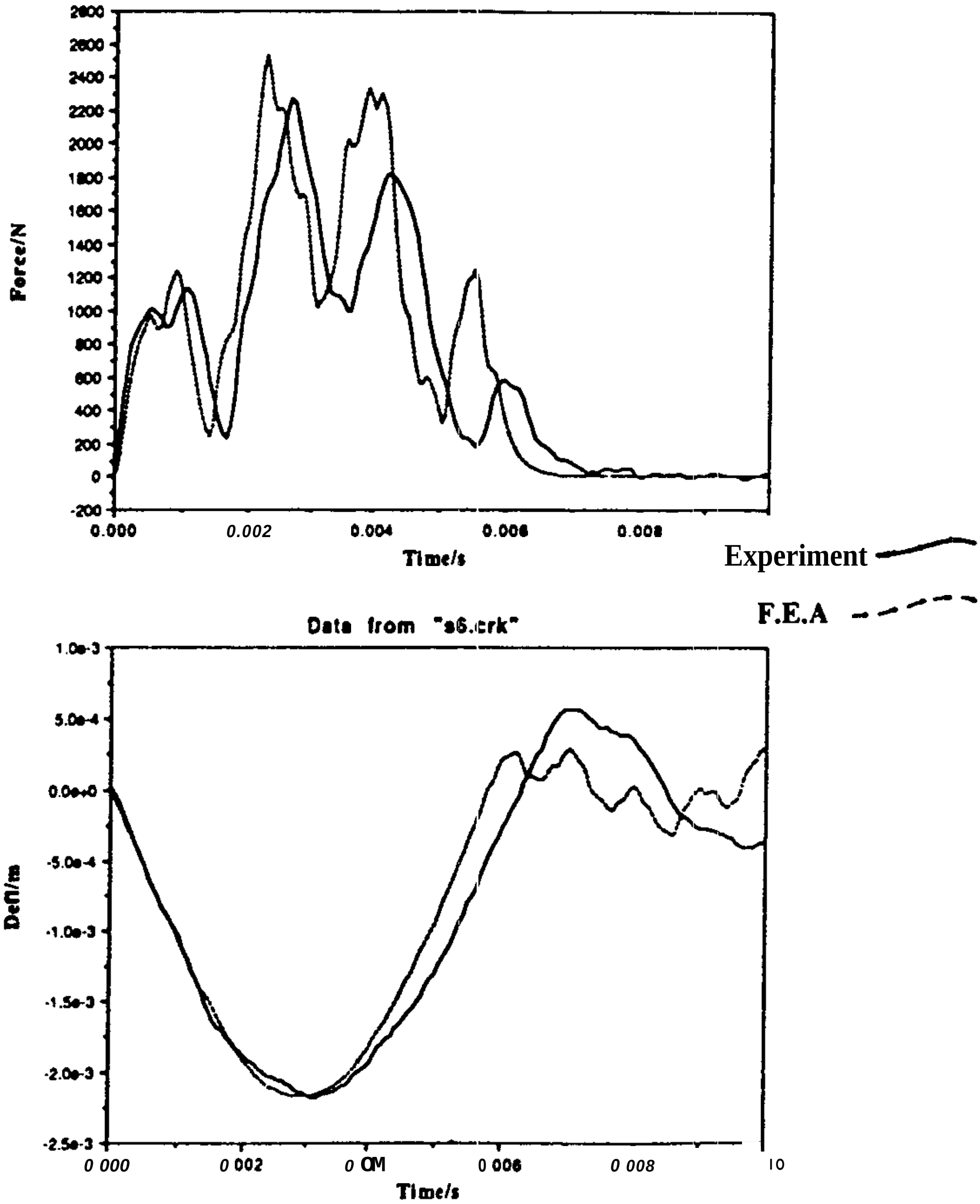


fig. 7. Force and displacement histones during low velocity impact of thin plate

THE ROLE OF STRUCTURAL ANALYSIS IN AIRWORTHINESS CERTIFICATION

by

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SUMMARY

Those uses of structural analysis which have a bearing on airworthiness are reviewed. In particular the extent to which finite element analysis is already implicitly relied on in the Context of clearance by test is considered, and factors which may be expected to lead to an increased reliance are discussed. One such factor is the increased use of the computer aided engineering (CAE) approach which changes the design process itself. An assessment is made of actions required to **ensure** that results of analysis provide a consistent and reliable basis for airworthiness judgement.

The report is the product of discussions which took place at **a** series of meetings of the RAE/SBAC Committee on Analysis and Optimisation. The committee members represent the Royal Aerospace Establishment, British Aerospace, Westland Helicopters, Short Brothers and Universities.

1. INTRODUCTION

The **use** of numerical structural analysis techniques is increasing rapidly in all branches of structural engineering. Many comprehensive analysis suites are available which offer the **user a** wide range of facilities enabling the response and life expectancy of structures subjected to Static and dynamic loading to be predicted. Of the numerical methods available, by far the most popular is the finite element (FE) method. This method is favoured generally because of its flexibility but it is particularly well suited to the analysis of the highly discontinuous thin walled structures which characterise aircraft construction.

The structural analysis suites are themselves increasingly regarded as an integral part of computer-aided engineering software which is encouraging a change in design philosophy from the traditional build-and-test approach to **one** in which alternative concepts are evaluated and designs are developed using computer models.. In the early phases of design, the object is to obtain an Overall understanding of the load paths through **a** structure and to develop a balanced configuration For comparison with other concepts. These tasks can be achieved with relatively coarse analyses but enable large savings to be made in development time and cost.

A much more demanding application for analytical techniques is their use to verify the integrity of a structure., Many industries which employ structural analysis, including the civil engineering, nuclear and ship-building industries, are concerned with such large-scale structures that it is impractical for them to perform full-scale tests. It has become standard practice in some Sections of these industries to submit finite element analyses, together with demonstration of conformity with relevant codes of practice, as proof of the safety of bridges, offshore structures, dams etc. The necessity to rely on analytical techniques has driven these industries to develop sophisticated analysis procedures and the models Used for validation simulate the response of the structure in detail and represent the opposite end of the spectrum to the models used in initial design.

In the aerospace industry, it has been traditional practice to perform full-scale tests to confirm failure modes and demonstrate safety margins. The present philosophy for determining the integrity of a structure is to use numerical predictions for the many necessary load cases but always to validate with a test. Several full-scale tests **are** used for static loading, fatigue and dynamic response. However, these tests are expensive and time consuming and present such technical difficulties that the aircraft industry may need to modify this traditional approach and employ analyeis. validated by component tests, to reduce major Static testing. This possibility is causing concern to both the manufacturers and the airworthiness authorities and has extensive implications which must be addressed at an early date.

2. AIRWORTHINESS ISSUES

2.1 Overview

Airworthiness engineers aim to ensure that certification procedures keep pace with advances in knowledge and technology whilst neither presenting design penalties nor allowing unexpected weaknesses to escape detection.

In order for an airframe to be structurally certified, the manufacturers must satisfy the airworthiness authorities that, under the most adverse combinations of loading and environmental conditions anticipated, sufficient strength and stiffness is present to ensure that there is a low probability that the Structure will either fail or become unairworthy in some other way. Particular attention is paid to cumulative effects such as fatigue and allowance is made for uncertainties in material properties, component dimensions, loading approximation, modelling etc.

The integrity of the structure under static load is determined by a combination of physical testing and theoretical analyses in which the loads applied are factored appropriately over the maximum loads expected in service to allow for the uncertainties mentioned above. Airworthiness engineers look to structural analysts both in the fields of certification and trouble shooting (ie correcting design faults revealed during tests or in service) for information on loading distributions. deformations and local stresses and strains.

Major Static analyses are used to interpret the Strains measured during strain gauge surveys, which are necessarily performed for selected design cases, in order that the results may be extrapolated to other design cases. The analysis determines the major load paths through the Structure and hence assists the identification of critical features. In addition, the loads to be applied for detail Stressing and component testing are determined from the major analysis. The long term integrity of an airframe depends upon maintenance: in this context, structural analyses are required to identify the critical features upon which to concentrate maintenance and also to provide data for the calculation of crack growth rates and critical crack lengths in order to determine inspection intervals. Major static analyses are invariably performed using the finite element method and recently this method has bee:? used extensively as an alternative to data-sheet information for detail stressing.

Traditionally, analytical predictions of critical structural details, failure loads and failure modes are substantiated by major static tests in which the structure is subjected to at least design ultimate load (DUL). These tests provide a last line of defence intended to identify previously unrecognised critical details and failure modes. However, because of the substantial cost of major Static tests, the time taken to perform them and the difficulty of simulating the aircraft's operational environment accurately, there is an increasing requirement for the approval of structures to be based on analyses and component tests, with reduced reliance on major static tests.

The problems of timescale and simulation of the operational environment are particularly acute for composite Structures. As manufacturing and non-destructive testing techniques for composite materials have improved, the variation in quality of nominally identical composite specimens has decreased to become similar to that achieved with metals. A major consideration remaining is their susceptibility to environmental degradation. This is specific to current composite materials but must be accounted for when testing. The time required and the technical difficulties involved in performing a full-scale test of the degraded structure are likely to prove prohibitive. Whilst this problem affects mainly military aircraft at present it is also relevant to civil aircraft.

2.2 Certification of Civil Aircraft

Instances of an analytical approach to certification have been more evident in the case of civil aircraft than military aircraft. Large civil aircraft are usually developed considerably through their lifetime. At the moment, the practice is to test the initial basic type of any planned family of aircraft knowing that in the future it will be stretched or shortened, or have its flight envelope changed. Also, experience has shown that in-service operations may lead to a revision of the load spectrum. This can arise as a result of unforeseen loadings occurring in the aircraft's original role or as a result of a complete change of role, as in the extreme case of the British Airways Tristar fleet converted for tanker/freighter operations with the RAF. However all these things are considered as modifications and the major test is not necessarily repeated. Having once accepted that the analysis provides an adequate model of reality, the authorities will accept future revisions for the basic load paths through the structure on the basis of analysis. Nevertheless, if modifications to the structure do take place then component testing is required.

There are also instances where the role of major static tests in the certification of the basic type has been reduced. A number of civil aircraft, including the latest variants of the Trident and HS125, have been certified on the basis of finite element analysis and conventional detail stressing methods without a major test to DUL. In addition, the Westland WG30, which is a civil helicopter with substantially the same mechanical systems as the Lynx but with an entirely new fuselage, has been certified with no major static tests.

Pressure, on both British manufactuers and the CAA, to reduce major static testing can arise because of the necessity to remain competitive with US manufacturers. The Federal Aviation Authority (FAA) is influenced, when setting airworthiness requirements. by the interests of the large civil aircraft manufacturers in the US. Naturally, considerable importance is attached to the economic viability as well as the safety aspect of the requirements and the reduction in time and cost offered by the replacement of tests with analyses is attractive to the manufacturers. The policy of the CAA is to retain consistency with FAA practice if at all possible and this has been the policy in the present case. British manufacturers are therefore able, and are forced by economic considerations, to follow the US trend.

One major reason why civil aircraft manufacturers are able to use an approach to certification which relies more heavily on analysis is because of the low probability that during a flight the primary structure of a civil aircraft will be subjected to a significant proportion of the limit load. This is because the limit load case is usually a gust case (being specified as remote combinations of gust velocity and air-speed which vary with altitude).

2.3. Certification of Military Aircraft

Because combat aircraft frequently operate at load levels of up to 80% of the limit it is crucial that the full design strength is present, and combat aircraft are invariably subjected to at least DUL in a number of load cases during major Static tests and to failure under one load case.

Whilst there is no move towards reducing the role of major static tests in the certification of military aircraft constructed from conventional materials. the introduction of composite structures has necessitated a significant reappraisal of military airworthiness requirements. In the static certification of composite Structures it is necessary to account for the most adverse environmental degradation anticipated. This occurs as a result of exposure to elevated temperature after a prolonged period of high humidity and may cause a substantial reduction in both the strength (especially the compressive strength) and the stiffness of the material. However, major static tests under such conditions are both extremely expensive and difficult to perform. It is likely that in the UK and Europe there would be strong pressure for a major static test with the structure in a degraded condition if the degradation was greater than 15% of the undegraded mean. This level of degradation is unlikely to occur widely in present composite structures, although in the fully degraded test on the Tornado taileron the strength reduction measured at 123°C was 30%-40%. The USAF, which is the biggest customer for fixed wing combat aircraft in the US and hence with whose requirements the US manufactuers must necessarily comply. have said that they will not use structures which degrade more than 10%-15% and will not under any circumstances require a major static test under degraded conditions.

With no major static test of the degraded structure available, a proposed certification procedure is as follows: a major Static test would be performed on the undegraded structure in which, by loading the Structure progressively, the limit of Structural airworthiness would be determined. The structure would then be reloaded to the DUL in the more adverse design case. Failure might be expected to occur at or near DUL due to the lower mean strength under these conditions of structural details not susceptible to degradation. A structural analysis would then be required to interpret measured strains in selected design cases in order that the Strains at DUL under degraded conditions could be estimated for all design cases. The extent of the modification to the analysis which would be necessary to model the degraded structure would depend upon the degree to which the stiffness reductions cause load redistribution within the structure.

The analysis would be used to estimate the limit of structural airworthiness with the structure in a degraded condition and also to ensure that the (usually higher) allowable strains under such conditions were not exceeded at DUL. Data would be retained in order that the analysis was reproducible for trouble-shooting. The analysis of the degraded structure also determines the loads to be applied for detail stressing and component testing. Element tests and component tests would be performed under degraded conditions to determine the mean failure stress and the failure mode. Standard data-sheet information would be used to determine the variability associated with the failure mode and hence the allowable for substantiation purposes. These tests would be used directly to validate detail stressing of degraded components and indirectly to achieve some confidence in the method used to account for degradation in the major analysis.

A different procedure was adopted for the certification of the Harrier GR5(AV8B). In that case, strength reduction was accounted for in the major static test of the undegraded structure by introducing an additional factor over and above the ultimate factor. This factor was intended to allow approximately 20% over the ultimate factor for strength degradation with an additional margin for variability. Failure in the actual test occurred at 1.2 x design ultimate load and the variability margin was not achieved. No allowance was made in the major analysis for load redistribution due to stiffness degradation, nor for the possibility of a resulting change of failure mode. However, degraded tests were performed on large structural components and McDonnell-Douglas assured the authorities that they had been careful to avoid any failure modes dominated by matrix properties.

Another aspect related to the introduction of composite structures arises from tneir relative sensitivity to poor detail design, and the likelihood of failure under static load due to through-thickness stress concentrations in particular. In fact, the arguments for performing two major fatigue tests, could logically apply to static tests for composites, the second test being used to confirm the adequacy of structural modifications resulting from shortcomings revealed during the first, early test.

Whilst the USAF is the largest Customer for fixed-wing combat aircraft in the US. the US Army is the largest Customer for helicopters. Unlike the USAF, the US Army tend8 not to dictate its requirements to the manufacturers but is more willing to follow the manufactuers lead. This, together with the fact that fatigue rather than static strength tends to be the dominant consideration in helicopter structural integrity, makes it likely that, in the US, it is in the certification of helicopters that any move away from major static testing will be initiated.

3. THE USE OF FINITE ELEMENT ANALYSES FOR CERTIFICATION

3.1 Introduction

In the long term, a trend away from major static testing is almost inevitable, possibly culminating in its replacement by analysis validated by component testing. There are already Sufficient instances of substantial reliance being placed on calculation to necessitate a major reappraisal of the role of analysis in certification and of the modelling and analysis procedures used in both major analyses and detail stressing. The dominant role of the finite element method in major analyses and its increasingly large role in detail stressing make it potentially the area in which most problems will arise. In order to employ procedures heavily reliant on FE analysis to certify an aircraft, airworthiness authorities must have confidence in:

the procedures used to control the analysis function;

the modelling practice used for analysis;

the integrity and reliability of the computer codes used;

the capabilities of the personnel responsible for analysis;

their own ability to assess analysis-based procedures.

These issues are discussed further in the following sub-sections

3.2 The procedures used to control the analysis function.

A major requirement of analysis for airworthiness purposes is that it should be consistent in quality. It is assumed that a quality management system to AQAP 1 or ISO 9001 is adhered to in order to provide the infrastructure necessary for the implementation of AQ standards Specific to finite element analysis. It is important that the analysis procedures and embodied modelling practices, which are established as being adequate for the purpose Of demonstrating structural integrity, are recorded in suitable form and that they are adopted uniformly. There should exist auditable documentary evidence of analysis work plans, specifications, analysis procedures and reports.

In order to justify the modelling practices employed within a company, it is likely that a library of validated reference antilyses will need to be created, which may subsequently be used to demonstrate that the modelling practice adopted for a particular analysis has been proved adequate for the class of component or detail under consideration. The level Of correlation required between the subject and reference structures will depend on the category of importance or class of the structure.

3.3 Modelling Practice used for Analysis

This section is devoted to a discussion of the modelling practice and usage currently common within the airframe industry, and includes some speculative comments as to where change may be anticipated.

Finite element analysis has been slower to make an impact on design at the project stage in the aircraft industry than in many other industries. The emphasis at the project stage is on obtaining load path information and traditionally in the aircraft industry this has been obtained using a combination of techniques including classical analysis, beam modelling, empericism etc with heavy reliance being placed on experience with previous designs. At present, design techniques are in transition with finite element analysis becoming used much more extensively. This use of the finite element method is becoming more convenient because the general trend towards the integration Of design software has meant geometry created for other purposes is readily available to analysts.

The major load paths through a structure can usually be determined, to the accuracy required for design purposes, by conducting a small-displacement, linear-elastic analysis based on a highly idealised FE model of the structure in which there is considerable amalgamation of structural members. Constraints on turn-around time and computer resources, and ignorance of the detailed construction in the early design stages make these simplications both necessary and legitimate.

In the aircraft industry, the greatest impact made by finite element analysis has been on analytical validation of designs or 'check-stressing'. Check-stressing involves a major analysis of the airframe combined with detail stressing of components. At present, major analyses are invariably performed using the finite element method. The major analysis is used to confirm the load path information gained during the design process and to select the critical design load cases or load envelopes which will be applied in the major static test. In addition, the major analysis is used to identify the critical structural details and to determine the loads which will be applied to these features for detail stressing and for component testing.

The major analysis results may be validated by comparison with the results Of a strain gauge survey. To accomplish this, however, requires considerable post-processing of the analysis results, because the FE models used at present are not sufficiently detailed to permit a direct comparison to be made. The strain gauge survey is usually taken as having validated the analysis if the stress levels in an area of the structure correspond to within 10%-20% but it is unusual to obtain correspondence in the detail stress distribution. The prediction of critical details from analysis results has been made in the past with the knowledge that, at least under the most significant load cases, the major static test would identify any details which were overlooked in the analysis. This 'safety-net' provided by the test has been an enabling factor in the past allowing analysts to simplify both the modelling and the type of analysis used.

Detail stressing is at present performed using a variety of techniques involving classical and numerical methods, and many of these techniqeus are embodied in dedicated computer programs. However, the use of finite element analysis in detail stressing is becoming widespread. The traditional approach to the use of results from finite element analysis to predict reserve factors is to combine them with 'data sheet' information based on a combination of theory, tests and empericism from which failure or proof deformation is estimated. These reserve factors form the basis of the type-record of the aircraft which is submitted to the airworthiness authorities as part of the certification procedure and is referred to when clearance of modifications to the original aircraft is sought. However, it is now becoming Common, especially in the case of novel compoennts or components with dimensions outside the range of the dedicated programs, to establish failure directly from finite element models. Advances in pre-processors and increased computer power is making such analyses cost effective compared with test.

In order to obtain load path information from the major analysis and ensure all the critical details are identified and then to predict failure mechanisms and failure loads from detail models with sufficient accuracy and reliability, rigorous and systematic modelling procedures must be adopted and used routinely. In general. modelling will need to be at a more detailed and realistic level. If a Structure is efficiently designed, the major loads will be transmitted by membrane stresses. However significant bending stresses can be introduced at load application points and geometric discontinuities and, whilst membrane elements may be adequate for many design purposes, they are less appropriate to validation. Care will be required to ensure that the modelling is improved in a Consistent way and the increased complexity of the model is fully exploited, for example the direct replacement of membrane elements by bending elements will not exploit the bending information fully. A systematic approach to the modelling will help to avoid dependence on the stress engineer's intuition, which can lead to the model being refined Sufficiently only at features which are expected to be critical. This should enable the identification of importent stresses which might otherwise be overlooked. The approach will be assisted by the tendency when using automatic mesh generators to use more elements and reduce the amount of lumping.

It is important that methods of representing the stiffnesses of subscale details are developed so that major analysis can be performed without either having to model the features in detail or accepting the error due to ignoring them. In some cases this may only be satisfactorily achieved by the formation of an iterative loop between the major analysis and the detail stressing such that condensed stiffness matrices derived from the detail stressing models are used to update the major analysis model. A related area in which modelling needs to be improved, especially for the assessment of the structural airworthiness limit, is in the way in which account is taken of joint/actuator stiffness. Work at BAe Warton has also identified the values of the stiffness used for panel joints as important parameters in determining the accuracy achieved in analyses, particularly when displacement information is sought.

Non-linear effects due to deformations and material properties will need to be accounted for in an analysis which is required to predict the response of the structure close to failure. Again, the modelling will need to be at a more detailed and realietic level in order to give estimates for the additional bending stresses Present in the post-buckled state. Routinely calculating the geometric stiffness matrix, even if it Were not used for a full non-linear analysis, would be a useful method by which to examine how well a geometrically linear approximation was modelling the response of the structure.

The analysis of structures containing composite materials makes severe demands on modelling and analysis procedures. These arise primarily because of the susceptibility of the material to degradation, the absence of sufficient plasticity to alleviate local stress concentrations and the tendency for inter-laminar processes to be important in failure. Degradation tends to cause changes in local failure modes, in particular emphasising the importance of failures due to minor loading in the third (through thickness) direction. The degree of plasticity, which allowed analysts to disregard local modelling inaccuracies when modelling metal structures, is not present in composite structures. High local strains, which tend to reveal. themselves as fatigue problems in aluminium alloy Structures, can instead cause severe static problems in composite structures, though fatigue remains a problem in some cases. Since both of these features tend to give rise to local problems, extremely fine meshes are required in order to characterise them. This is even more true of edge effects which have been found to cause premature failure in Composite panels.

For certification purposes, these problems require a response from analysts in two areas. The first is to investigate the use of major analysis models which minimise the possibility of failing to identify a critical structural detail. Such analyses must account for degradation of strength and stiffness, possess a sufficient degree of refinement over the entire structure and use bending elements more extensively. The second is the involvement of computer-based analysis groups in detail Stressing where it will be necessary to assess the accuracy with which failure may be predicted using very detailed models. This second task especially represents an important research activity because, at present, analysts have very little experience with very detailed models containing elements of sub-laminate size. Hybrid metal/composite structures present particular difficulties because yielding of the metal components may cause substantial load redistribution close to failure. This material non-linearity must be accounted for where it arises both in major analyses and detail stressing.

3.4 The Integrity and Reliability of Computer Codes

An important part of ensuring the quality of analyses used for airworthiness purposes is the thorough development and testing of finite element analysis suites. A small number of suites have been Shown to be sufficiently comprehensive and reliable to be used as standard for design purposes within the aircraft industry, although many of the facilities within these suites which will be required for certification purposes are used infrequently at present within the aircraft industry and consequently their performance remains largely untested. At the most basic level it is essential that the elements within analysis suites are subjected to benchmark tests, performed by bodies independent of the vendors, in order 'that their performance and suitability to particular applications is known.

In this respect organisations like NAFEMS, which is seeking to provide quality assurance standards for finite element use and to establish standards by which finite element suites may be evaluated, are capable of providing a valuable service. An important feature of this organisation is that it is able to draw on experience from a number of branches of structural engineering. At present, neither the CAA nor the UK Military Airworthiness Authorities are staffed to assess the analysis suites used by aircraft manufacturers and aircraft component manufacturers, nor would they particularly wish to become involved with making such assessments or with the qualification of programs. Nevertheless, demonstration of the integrity Of bought-in software is a necessary and continuing task, and is probably best done with:." industry, either at individual sites or under collective arrangement. Evidence of adequate quality control procedures and of software verification tests should be sought from the software developer. The authorities would only be involved indirectly, by confirming that such assessments had been carried out, as part of the procedure for issuing a certificate of design approval.

An increasing role for calculation in the airworthiness clearance will mean analysts incur much greater responsibility. It is essential that they have confidence in the capabilities of the analysis suites which they *use.* Similar considerations also apply to pre- and post-processors and any internally developed software adjuncts to the finite element code.

3.5 The Capabilities of Personnel Responsible for Analysis

The quality and credibility of any analysis is also critically dependent on the experience and ability of staff responsible for analysis. Expertise is required in several areas. Clearly, familiarity with the type of analysis being conducted is necessary and. even more important, previous experience of the design requirements for the class of structure under consideration is essential. In addition, software expertise is called for to ensure that code is correctly installed, that data is adequately protected against corruption and is transferred between codes without loss of integrity. It is unlikely that a sufficient breadth of experience will be present in a single individual, but it may be expected that all the necessary skills will be present within the analysis team.

Outside the aircraft industry, the *use* of analysis suites by unqualified people is a matter for wide concern. Inexperience of finite element usage is a particular problem in small firms whose staff are less likely to possess a theoretical knowledge of the method. Finite element analysis is beginning to be regarded as 'push-button' and the *use* of results without regard to inherent assumptions and limitations is potentially a major problem. This attitude towards finite element analysis is being encouraged by the production of CAE packages containing embedded FE analysis systems. the details of which are invisible to the user. In the extreme these packages enable draughtsmen to perform a finite element analysis without any knowledge of the method.

It is becoming more common for manufacturers of aircraft components to perform finite element analyses and it is also common practice for small bureaux to perform finite element analyses under sub-contract to the large aircraft manufacturers. Experience has on occasion shown that the standard of analyses produced by sub-contractors. using both finite element and traditional methods, **can** be very poor'. and it has been essential to monitor the quality of analyses by maintaining constant and close contact with the sub-contractors. It is important that independent expertise is available to Small firms and here again organisations like NAFEMS will be essential.

Even within the major aircraft companies the use of FE systems has long since ceased to be the preserve of the Small specialist groups who established the methods and programs. Thus. here too, there is the possibility of inexperienced applications analysts being called upon to work with inadequate specialist supervision. It is unrealistic to look for any more than an appreciation of the finite element method in a new graduate employed as a general stressman, since demands on engineering syllabi in undergraduate courses are such that universities are unable to provide sufficient practical experience. It is thus the responsibility of the employer to provide guidance and instruction and to enable stressmen to gain experience of the method. It should not, however, be assumed that more Senior staff, who may be involved principally with traditional methods, necessarily have a better understanding of the method: specific attention needs to be paid to the provision of suitable specialist supervision.

Since the credibility of any analysis is dependent on the experience and ability of the people conducting it, it would be a useful and sensible measure to establish a standard level of qualification and categories of experience for finite element users and supervisors, defining the level of knowledge and experience acceptable for members of the analysis team. This is particularly so where the analysis is used to provide **a** definitive assessment of the integrity Of a structure for which failure would incur unacceptable financial penalty or loss of life. Such an approach would be analogous to the requirement for a major test to be supervised by an engineer with chartered status. Ideally the role of enforcing such standards should be adopted by an authoritative body independent of the manufactuers.

3.6 Ability of Airworthiness Authorities to **assess** Analysis-based Procedures

The primary duty of airworthiness engineers is to **ensure** the safe operation of aircraft and, if manufacturers present complex analyses for certification, airworthiness authorities must be capable of making an assessment of accuracy and reliability of such analyses. This assessment must be made in the context of: the track record and experience of the particular analysis team; evidence offered to show the reliability of the modelling practice employed in predicting internal load distribution and strength of similar Structural elements; quality **assurance** and validation procedures used to establish the credibility of the capabilities of the finite element code employed, together with any user-supplied adjoints to the FE code; and finally the adequacy of procedures established for specification and recording of analysis.

For the more technical aspects of the assessment, airworthiness engineers will require support from independent Structural analysts who have an expert knowledge both of modelling and analysis procedures, and of the analysis suites used. The depth of support available at present to the authorities in the UK is limited **even** for current demands. If this situation is not recognised, the Option to reduce the present degree of reliance on testing procedures will not be available, which will increase project costs and significantly influence the competitive position of manufacturers.

4 CONCLUSIONS

It is apparent that UK manufacturers and airworthiness authorities recognise the longer term trend away from reliance on major static tests. It is important that the implications of this trend are realised and a reappraisal of the role of structural analysis in certification is made, on a national and international basis. It is evident that **a** considerable improvement in experience and development of analysis techniques is required before the aircraft industry will be in **a** position to submit an analysis of Sufficient demonstrable quality to be accepted by the authorities **as a** major element of proof of the integrity of an aircraft structure.

It would be useful for the aircraft industry to remain aware of practice within the civil engineering, nuclear and ship-building industries which face problems of similar importance with the use of analytical techniques for verification of Structural integrity. Experience in these other industries suggests that certification procedures which rely heavily on analysis **are** a realistic aim, although the special problems of lightweight structures using advanced material and subject to complex and continuous loading actions which arise within the aircraft industry are acknowledged.

There is certainly a requirement for substantial improvement in the accuracy and effectiveness of the finite element technique, particularly in specialised areas such as fatigue and fracture mechanics, non-linear analysis, and the modelling of through-thickness effects in Composites. Nevertheless, an increased role for structural analysis in certification is Currently possible given fresh commitment within the aircraft industry to the exploitation of existing analysis capability since such use is not Solely limited by the intrinsic deficiencies or lack of capability in the methods.

The necessary improvements of analysis techniques should be accelerated by a dedicated and coherent programme of research with certification by analytical techniques as the target. Some of the areas which this research would need to cover have been outlined above and, in addition, fresh effort should be devoted to establishing the credibility of current analysis capability. Industry should be in a position to demonstrate that its codes are adequately validated, that its modelling practices are reliable and that it has staffed all safety-critical analyses appropriately. Given an increasing acceptance of the view that 'product assurance in the wider sense encompasses quality assurance of the processes used to define the product' some of these aspects are likely to be covered by standards arising outside the aircraft industry, which may then be adapted to our needs.

A substantial capital investment in computer hardware and software is also required at an early stage if the necessary expertise is to be gained and economic benefits reaped in the future. Further resources will need to be committed to training to ensure that personnel engaged in analysis with the aim of demonstrating structural integrity are adequately qualified and have Sufficient experience for the task. There is a requirement for educational programmes providing an extensive theoretical grounding and the opportunity for hands-on experience of analysis suites. These must be complemented by good quality industrial training, under expert supervision, to ensure that consistent, Proven modelling practices are employed. Failure to supply investment in these areas will prejudice the competitive position of firms within the aircraft industry.

Even within a clearance procedure based primarily on the use of major static tests, airworthiness authorities recognise the heavy reliance which is already placed on structural analysis in certification and the increasing role which finite element analysis will perform in detail stressing. This reliance on structural analysis should be reflected in clearance procedures. Current procedures place the major emphasis on test results but should also specify requirements for the analysis, where it is used for identifying and determining the loads to be applied in the test cases, extrapolating from the tested cases, and forming the type-record for the aircraft which is heavily relied upon to justify modifications to the original aircraft.

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APPENDIX

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The Role of Analysis in the Design and Qualification of composite Aircraft Structures

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summary

This paper is concerned with the use of analysis in the qualification of aircraft structures made with composite materials. Pour areas of work are discussed: statistics for material allowables, damage tolerance modelling, hygrothermal modelling, and buckling of stiffened compression panels.

List of Symbols

a delamination length
A_i allowable for population i
b_i deviation of batch i mean from the overall mean
D_x transverse diffusivity
f_i distribution for population i
G $(n - M_i)/(M_m - M_i)$, fraction of maximum possible weight gain
G_m strain energy release rate
G_i initial linear portion of G versus $\sqrt{t}$ curve
h thickness
I total number of observations per batch
J total number of batches
k allowable factor for normal population without batch to batch variation
M weight percentage moisture content
M_i initial M
M_m saturation M
M_x x-direction plate bending moment per unit width
Q_x x-direction plate transverse shear force per unit width
r ratio between two population distributions
R σ_b^2/σ_w^2
s sample overall standard deviation
s^2 sample overall variance
s_b^2 variance of sample batch means
s_w^2 variance of sample observations within batches
t time
T allowable factor for normal population with batch to batch variation
w_{ij} deviation of observation j in batch i from the batch mean
$\bar{x}$ sample overall mean
x_{ij} the j^{th} observation from batch i
x_p X value of a given population percentile
μ population overall mean
σ_b^2 variance of b_i
σ_w^2 variance of w_{ij}
σ_x population overall standard deviation

Introduction

This company's experience with composite flight structures has shown that extensive testing has always been required to support design, development, and certification. This is so because the physics of the problems involved are difficult to model fully and accurately. Testing has been used to tell us things about the materials' behavior that we could not predict quantitatively using analysis or past experience.

This reliance on testing has been accepted in the past because of the advanced nature of the technology. As composites move into the mainstream there is pressure to reduce the amount of time and money needed to develop and certify new designs.

one way to make the design of composite structures more timely and less costly is to replace testing with analysis. Analysis is almost always a much quicker and less expensive way of determining a structure's behavior than is testing. Higher performance designs will also result because more alternatives can be examined.

At Canadair, we are developing methods that are incrementally reducing the amount of testing we do. Although testing in no area can be eliminated altogether, it is gradually being reduced as we gain experience with new analyses. The following sections outline how we are exploring analysis in four different areas of design: statistics, hygrothermal behavior, damage tolerance, and buckling.

Statistics for Material Allowables

Material strength is a stochastic property, exhibiting some degree of random variation. BY using experimentation and the tools of mathematical statistics, it may be possible to derive certain lower limits on the values that strength can take on. These lower limits are sometimes called allowables.

An allowable is a single number which is defined in probabilistic terms. It is derived from a random population sample. Two commonly used allowables are the A-basis and B-basis allowables, respectively defined as the lower 99th and 90th population percentiles stated with a confidence of 95%.

Allowables are required for the most critical service conditions: as defined by temperature and exposure to moisture and chemicals. one way of addressing this need is to test large numbers of specimens under all the conditions of interest. This approach is expensive, and has spurred interest in pairing, a way of deriving allowables under some conditions without extensive testing.

Batch Variability

The statistical procedures needed to derive design allowables for composite materials are determined in part by the phenomenon of batch to batch variability. In addition to the random differences in the strength of material taken from one batch, material taken from different batches can show systematic differences. Figure 1 illustrates this with mean tensile strength data extracted from acceptance tests for unidirectional graphite/epoxy.

ANOVA

A method for estimating population percentiles in the presence of batch to batch variability has been developed by Lemon [1] and Hee and Owen [2]. This method, with modifications, has been incorporated in Military Handbook 17B [3].

The appendix outlines this method, and the manner in which it has been extended to compute A-basis as well as B-basis allowables.

The procedure has been coded and used to analyze material qualification data for several composite material systems. It was quickly found that the results are very sensitive to the number of batches used in the testing.

To investigate this effect, a Monte-Carlo simulation was carried out. Artificial samples from normal populations with fixed values of μ, σ_b^2, and σ_w^2 were generated and analyzed many times. The resulting large samples of allowables were examined to determine the distribution of allowables generated by the method.

Figure 2 shows results for a population with $\mu = 100$, and $\sigma_b = \sigma_w = \sqrt{50}$. These parameters give an overall population coefficient of variation of 10%. Each error bar is the result of 15,000 simulations of a sample of 100 observations. The error intervals have a 90% probability of containing the computed A-basis allowables. The upper ends of the error bars are correctly positioned at the 99th percentile of the population since there is a 5% chance that the allowable will be above the interval. This follows from the definition of the allowable.

The figure shows that if too few batches are used, the allowable is likely to be extremely conservative. As the number of batches increases **for a** constant total number of tests, the allowable estimate steadily improves up to the limit where there are only two observations per batch.

These results can be compared to the case where there is no batch to batch variability. For a normal population, the allowables may be estimated using:

$$x_p = \bar{x} - ks \tag{1}$$

where k is a pre-determined factor. Monte-Carlo simulation was again used to construct the 90% confidence interval of the computed A-basis allowable. In figure 3 this interval is compared to the best result with batch variability when the maximum number of batches are used (Eq. (A2)). The allowable computed with Eq. (1) has less spread than the one computed with Eq. (A2). To compute an allowable with Eq. (A2) with the same qnality as one computed with Eq. (1), a larger number of specimens, from a sufficient number of batches, would have to be tested.

Examination of results for B-basis allowables shows similar results. Less conservatism in the estimates of the allowables is found.

Pairing

During a material qualification programme, allowables must be computed for several properties under multiple conditions of temperature, moisture and chemical exposure.

Testing large multi-batch samples for each combination of property and environment would be very costly. For this reason, a pairing procedure was developed.

Pairing has been used to calculate design allowables for metallic materials, and is described in Military Standardization Handbook 5D [4]. This method has been adapted for use in the presence of batch variability.

The appendix describes the methods that have been developed to allow pairing with batch variation.

Additional simulation was carried out to study the distributions of allowables computed using this pairing technique. In the simulation, the properties of population 1 (for which allowables were calculated directly) were $\mu_1 = 100$, and $\sigma_{b1} = \sigma_{w1} = \sqrt{50}$, and for population 2 $\mu_2 = 50$, and $\sigma_{b2} = \sigma_{w2} = \sqrt{(50)}/2$. These populations both had a coefficient of variation of 10%, and the exact value of r was 1/2. The number of batches for both samples was 10. Allowables for population 1 were calculated directly as before, then allowables for population 2 were calculated by pairing.

The results for the A-basis allowables shown in figure 4, showed that the estimates were conservative, as expected. There was little improvement in the paired allowables as the number of data in the small samples from population 2 were increased.

Comparison of figure 4 with figure 2 shows that the allowable estimates for J = 10 from figure 2 have essentially been scaled down by a conservative estimate of r to give the error intervals in figure 4. This implies that the spread of the r estimate is not very large compared to the spread of the allowable calculated for population 1. Improving the allowables for population 2 would require an improvement in the allowahles for population 1. As explained in the preceding section, that would require an increase in the number of batches or specimens or both.

Damage Tolerance

The development of analysis methods for damage tolerance is difficult because there are several aspects to the problem, and each aspect is a complicated problem in itself.

The first thing to consider is the source of damage and the resulting damage state. The damage state may include delamination, transverse cracking and crazing, and fiber breakage, splitting, and debonding. The spatial distribution and density of the different damage types must be quantified. Damage sources may include impact, mishandling, manufacturing defects, and usage effects like abrasion, fretting, and wear. The relationship between damage source and damage state must be quantified.

The second aspect of the problem is the response of damage states to driving forces. Damage states may evolve under the action of monotonic and cyclic loading and thermal cycling. Damage progression is affected by stress level and stress state, layup, and the properties of the material system such as matrix modulus and toughness, system critical energy release rates, the fiher longitudinal and transverse strength and toughness, the interfacial strength and friction properties, voids, and fiber distribution and volume fraction.

The third aspect of the problem concerns the failure criterion. The criterion must be defined in terms of the damage state parameters. Failure may be defined as a degradation in stiffness or in the ability to carry load. or it may be defined by the physical dimensions of a particular damage feature.

The overall problem is extremely difficult to analyze quantitatively, and few generally applicable methods exist. The tendency has been to develop design data empirically. Analysis, if used, is based on simplified models that contain at least some parameters that need to be determined by test.

Delamination Modelling

Work is under way to model the behavior of single-level delaminations. Such delaminations may arise from foreign material, such as films or backing paper, being left in the laminate during manufacture. Impact usually gives rise to multi-level delamination, plus other effects. Future work will attempt to show that the parametric behavior of single-level delaminations is similar to that of impact delaminations.

Plate-type finite-element models have been made of delaminations. Both square and round cases have been studied (figure 5).

The square delamination was examined because a supporting analytical buckling analysis could be more easily performed. Using a NASTRAN geometric nonlinear solution (solution 66), the delamination was modelled in the post-buckled condition. The load-deflection curve of the delaminated plate was derived with some difficulty due to numerical instability of the solution algorithms. The analysis was then repeated for a slightly larger delamination. The shape of the delamination-front advancement was made proportional to the crack-opening bending moments in the delamination uncovered during the nonlinear analysis.

Analysis of the enlarged delamination gave a second load-deflection curve (figure 6). Integrating the two curves, taking the difference, and dividing by the area of the delamination extension gave the strain energy release rate (figure 7). This release rate was an overall value, not separated into 'the component modes, but it served to show the feasibility of the approach.

Work is Currently proceeding on the more realistic round delamination. Recent improvements in NASTRAN's geometric nonlinear capabilities are easing the task of obtaining solutions, but numerical difficulties remain the most time consuming aspect of this work. When solutions are obtained, they are sometimes unexpected. Figure 8 shows the round delamination buckled into an s-shape instead of the expected outward-hulging configuration, caused by overall buckling unexpectedly preceding the delamination buckling.

Edge Delamination

Available analysis consists of 3-dimensional analysis of edge stresses in the absence of geometric discontinuities. such analysis seems able to indicate the propensity of a laminate to delaminate at a straight free edge, although critical stress levels have not been obtained. Figure 9 shows the through-thickness normal stress as determined by the finite element method [5].

Tension dominated spectrum fatigue testing of this laminate resulted in delamination of the test specimen, while compression dominated fatigue did not [6]. Delamination occurred at the 90° plies, where through-thickness tensile stress was maximum.

Hygrothermal Effects

Analysis procedures for hygrothermal effects in. composites are generally based on the hypothesis that the behavior is Pickian, as illustrated in figure 10. This is occasionally not true. one material system has been observed to exhibit Fickian behavior at lower temperatures and non-Fickian behavior at higher temperatures (figure 11)[7].

In cases where non-Fickian behavior is observed.,a Fickian approximation may he made if deviations in the moisture uptake profile are not too great.

Property Determination

Results from [8] for absorption into an infinite plate under constant conditions show that the entire uptake curve is well approximated by:

$$G = 1 - \exp\left[-7.3\left(\frac{D_x t}{h^2}\right)^{0.75}\right] \qquad (2)$$

$$\text{where } G = \frac{M - M_i}{M_m - M_i} \qquad (3)$$

while the linear initial portion is given by:

$$G_i = \frac{4\sqrt{t}\sqrt{D_x}}{h\sqrt{\pi}} \qquad (4)$$

Empirical estimates of D_x are made from the initial part of the curve using Eq. (4), and M_m is estimated by allowing test material to saturate. One wishes to minimize the time required for test material to reach saturation. This can be accomplished for any given D_x by making the test coupons thin. This has the undesirable effect of shortening the initial linear portion of the curve, perhaps making it difficult to obtain enough readings to get an accurate estimate of the slope.

By writing an expression for the error between Eq. (2) and Eq. (4), setting this error to 5%. and setting t to 2 days, one finds that $D_x/h^2 = 8.5\text{x}10^{-7}s^{-1}$. With this, one can use a rough preliminary estimate of a and solve for h. This thickness of material will result in the moisture uptake curve not deviating more than 5% from linearity for the first two days of exposure. This allows sufficient time to take several readings in the linear portion of the curve, while keeping the specimen thickness to a minimum.

Conditioning scheduling

Once the Fickian diffusion parameters have been determined as a function of temperature and humidity, they can he employed in the analysis of test structures. For this purpose, a public domain computer program has been used to solve diffusion problems with variable boundary conditions [9].

such an analysis may be employed to plan a moisture conditioning program to give desired moisture levels and through-thickness moisture profiles. The advantage of this approach can be seen by comparing it to the alternative: soaking in a worst-case environment until saturation is reached.

Figure **12** shows the moisture uptake curve for an actual component exposed at it's maximum service temperature and *95%* RH. **A** qualitative look at this data suggests that the component is saturated, and is ready for testing. **A** simple calculation, however, shows that the highest stressed, thickest, and most critical sections of the assembly have only reached **40%** of saturation. **Due** to the non-uniform nature of the transient moisture profile through the thickness, the moisture content at the center of the thickest sections, where there happens to be a bond line, is even lower. Conditioning must evidently continue until calculation shows that the critical areas have reached an acceptable moisture level.

If the material can tolerate temperatures higher than the maximum service temperature, conditioning could be accelerated by taking advantage of the increase in diffusivity with temperature. If the target moisture level is less than saturation, a period of drying may be employed after conditioning to achieve a more uniform through-thickness moisture profile.

service simulation

A conservative moisture level can be achieved in test parts by soaking them at the maximum service temperature and a high humidity, for example, **85% RH** (Ref. **[3]** sect. **2.2.1).** This has been shown to be a good rule, not only for epoxies, but also for a bismaleimide-modified epoxy in the instance of a transport aircraft environmental spectrum (Ref. **[6]** sect. **6.4).**

A fighter spectrum, with the effects of hot dry excursions, could give an appreciably lower equilibrium moisture content. In **this** case, detailed environmental simulation using a realistic service spectrum is justified.

stiffened Panel Buckling

Experimental results on stiffened compression panels have shown that skin/stiffener separation can be the critical failure mode (Ref. **[6]** sect. **7.1.1.2.1).** Premature failure at about *50%* of the expected ultimate load has been observed due to this phenomenon.

This failure mode involved two factors: skins which were designed to operate in the post-buckled state, and the stiffener/skin co-cured bond which was not designed for peel strength.

TO help understand the failure, the full sized 2-bay panel (figure 13) was modelled by a **1/2** bay finite element assembly with symmetric boundary conditions along the lateral edges (figure **14).** The NASTRAN geometric nonlinear solution *64* was used to derive results in the post-buckled state.

These results were used in two ways. First, a 2-dimensional fracture model of the stiffener/skin bond was analyzed using as input the element forces derived from the non-linear analysis (figure 15). By detaching nodes along the bondline, it was possible to derive a strain energy release curve (figure 16).

The G_c versus a (crack length) curve shows unstable/stable growth, as indicated by the change from positive to negative slope. This was observed during the test. The absolute level of cracking could not be predicted because information on initial crack length and critical strain energy release rate (G_c) was lacking. Further work would have to be done to separate G_c into the component modes I, 11, and 111. For these reasons, this approach was deemed impractical for design and certification support.

An alternative approach, using a strength of materials philosophy, was then tried. The peeling forces and moments in the panel at the base of the stiffener were extracted from the model. Examination of the values showed that the maximum peeling force and moment were about *8.5* N/mm and 110 N-mm/mm at the load corresponding to the first detected cracking in the test (figure **17).** The maximum values were located at the central buckling antinode.

A strength test was then performed on the bond by applying a direct peel load to the stiffener. The skin was supported in such a way as to induce realistic bending at the bondline. Taking the average of three tests, the failure peeling loads and moments were **7.7** N/mm and **96** N-mm/mm at the bondline. This is encouraging agreement with the modelling results.

The strength of materials approach is more practical than the fracture mechanics approach for use in design and certification. Analysis can provide reasonably accurate loads on the critical interface. Small-scale testing can then be carried out to choose optimum geometry, material, and details to give a sufficiently strong joint.

Conclusions

This paper has addressed four areas of analysis:

1. Statistics. It has been shown that standard methods for deriving allowables in the presence of batch to batch variability can be extended to give A-basis allowables and to allow pairing. The method is sensitive to the number of batches used in testing.

2. Hygrothermal Analysis. A quick way of sizing test coupons to minimize the time needed to obtain the material properties has been derived. standard analysis methods based on Fickian diffusion are used routinely.

3. Damage Tolerance. Modelling of single-level delaminations has shown that the geometric non-linear finite element method is a feasible way of obtaining energy release rates. The method has not yet been shown to be practical for design.

4. Stiffened Panel Buckling. Geometric non-linear finite elements have been shown to provide accurate loads on the bond between the stiffeners and the skin in the post buckled state. simple tests can be used to asses the integrity of the bond and to select optimum configurations.

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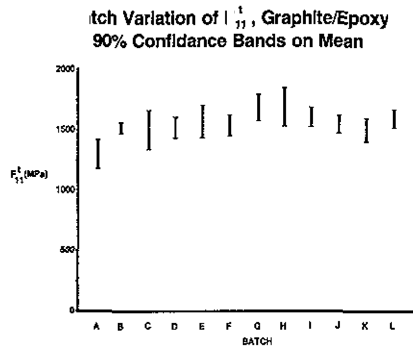

Figure 1. variation of mean tensile strength with batch

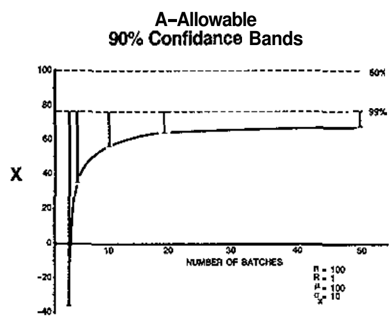

Figure 2. variation of allowable with number of hatches tested.

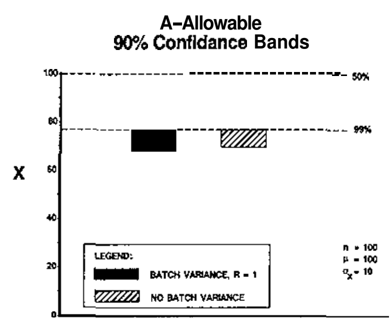

Figure 3. comparison of allowable with and without batch variability

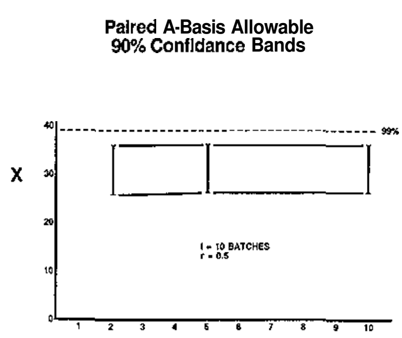

Figure 4. Paired allowable with hatch variability.

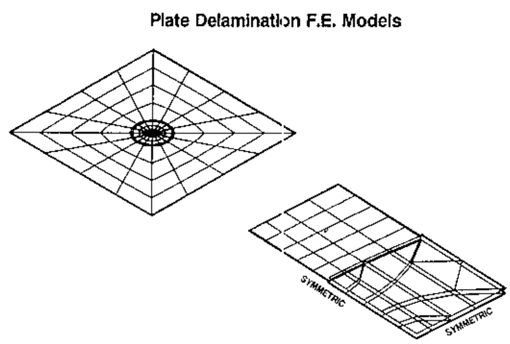

Figure 5. Finite element delamination models.

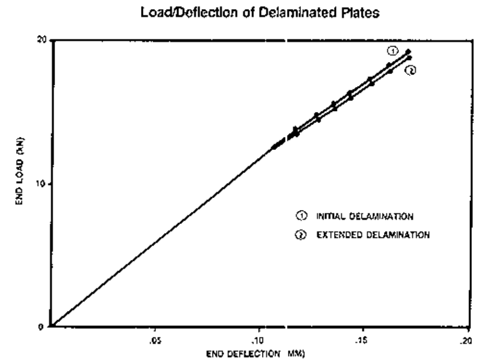

Figure 6. Load-deflection curves for delaminated plates.

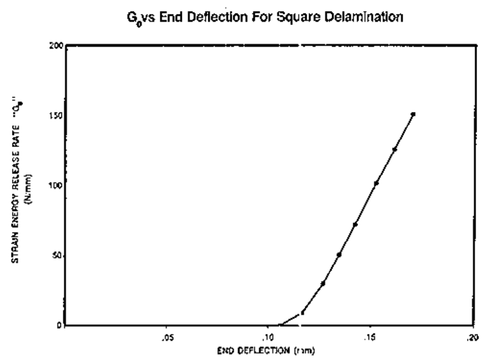

Figure 7. Energy release rate for square delamination.

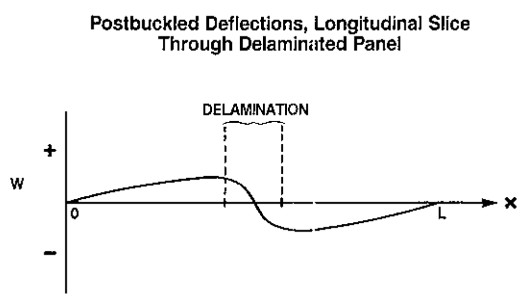

Figure 8. Overall buckling of plate with round delamination.

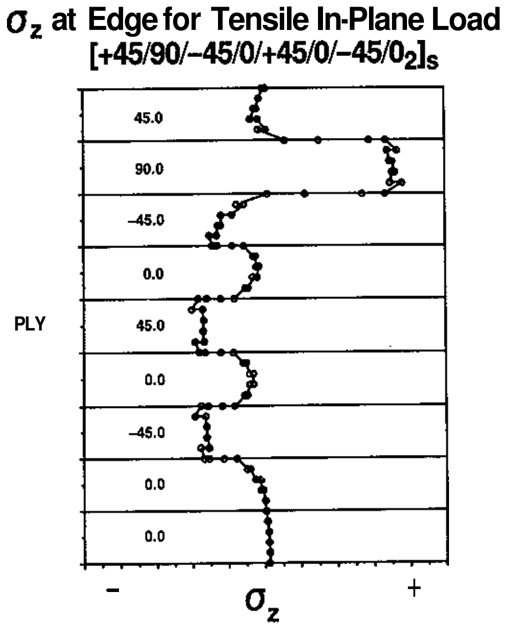

Figure 9. Through thickness normal stress at edge.

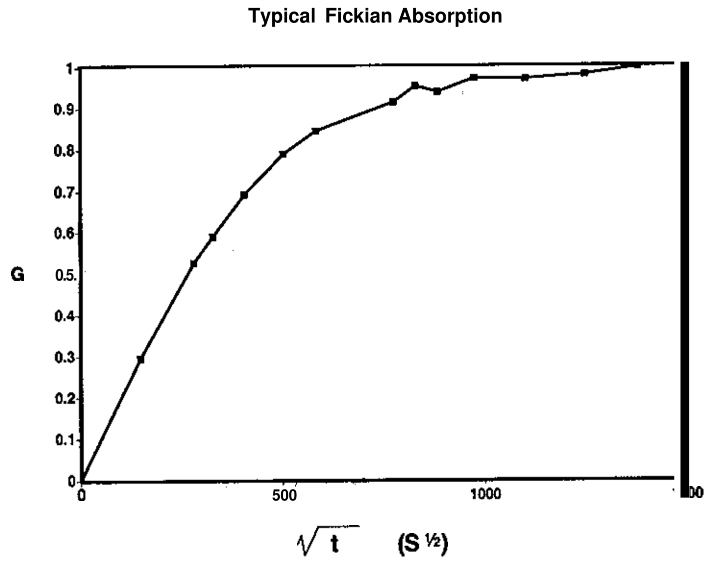

Figure 10. Example of Fickian absorption.

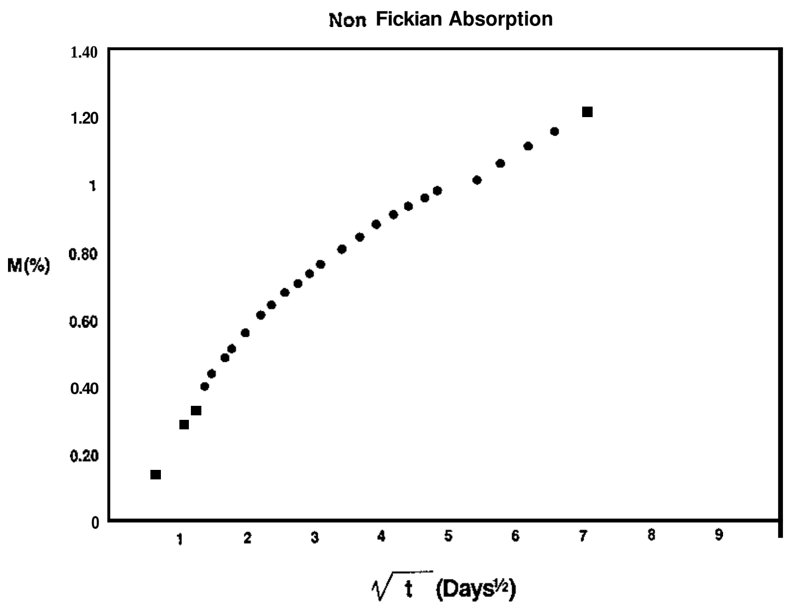

Figure 11. Example of non-Fickian absorption.

Test Component Moisture Absorption

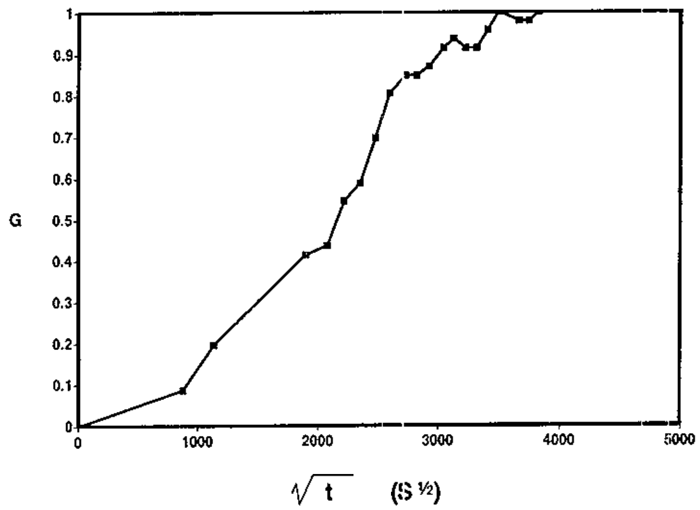

Figure 12. Moisture uptake of actual component.

Stiffened Compression Test Panel

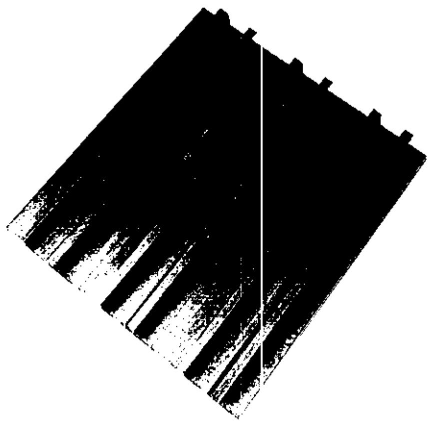

Figure 13. stiffened compression panel..

Geometric Non-Linear Model

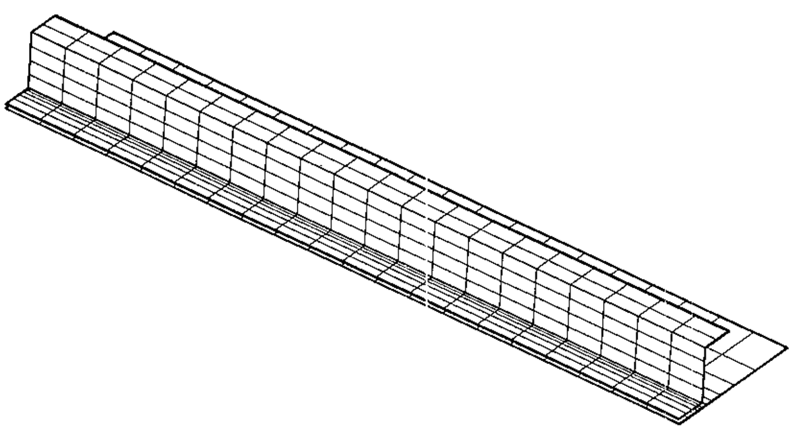

Figure 14. Finite element model of compression panel.

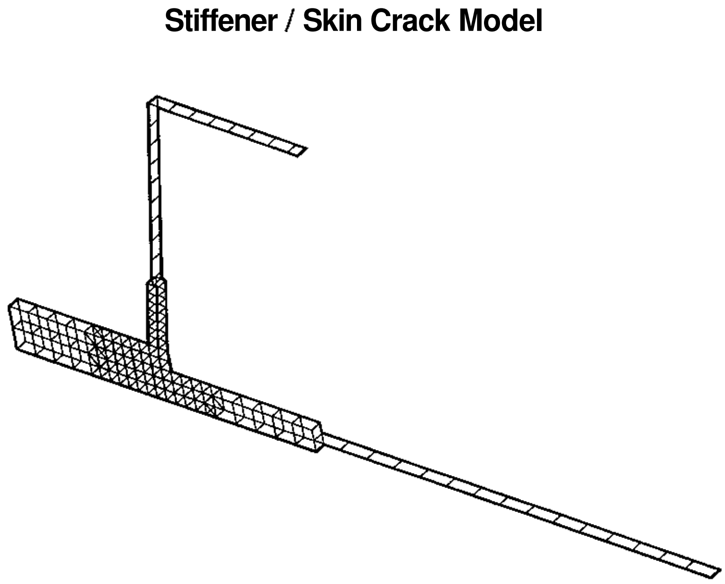

Figure 15. 2-dimensional fracture model of skin/stiffener interface.

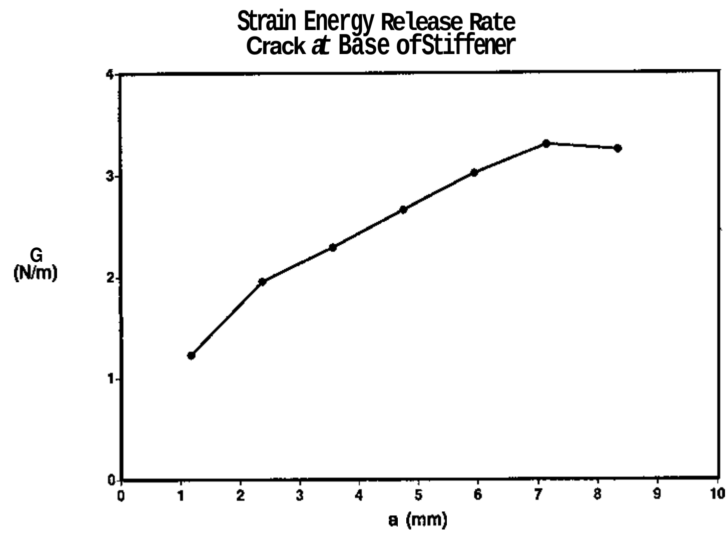

Figure 16. Strain energy release rate for skin/stiffener crack.

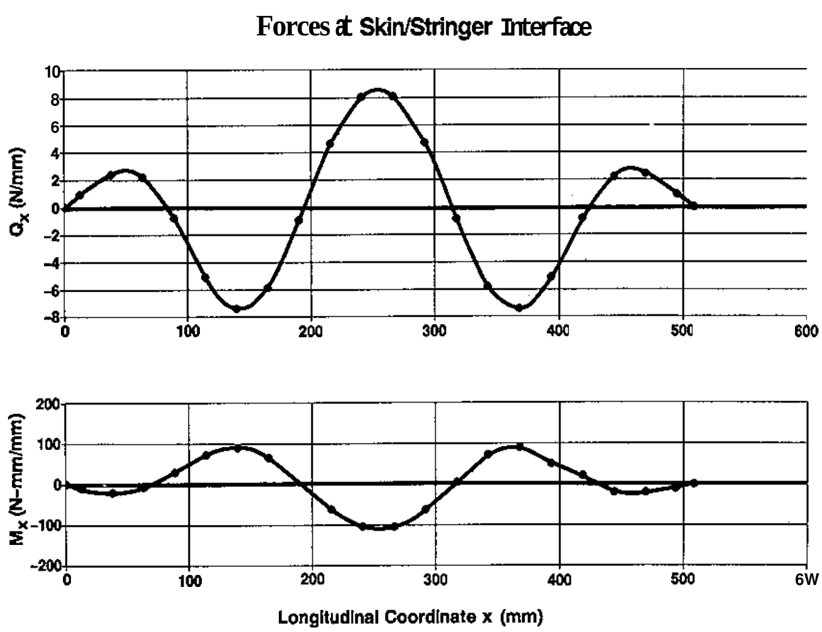

Figure 17. overall peeling forces and moments at skin/stiffener interface

Appendix

Deriving Allowables With Batch Variability

The method is based on the analysis of variance (ANOVA). It is assumed that the population can be described by a model of the form:

$$X_{ij} = \mu + b_i + w_{ij} \qquad i = 1 \text{ to } I,\ j = 1 \text{ to } J \tag{A1}$$

b_i and w_{ij} are assumed to be normally distributed with variances σ_b^2 and σ_w^2 respectively. The population percentiles are estimated from:

$$x_p = \overline{X} - T(s_b^2 + s_w^2)^{1/2} \tag{A2}$$

where $\overline{X}$, s_b^2 and s_w^2 are estimates of μ, σ_b^2, and σ_w^2 respectively. Reference [3] gives a procedure for calculating T such that x_p corresponds to a B-basis allowable. To find T such that x_p is an A-hasis allowable, the factor 1.282 in equation 8.5.4(h) of [3] should be changed to 2.326 as required by equation 2.3 of [2].

Pairing

The method assumes that two populations of material properties are related by a ratio:

$$f_2 = r \times f_1 \tag{A3}$$

consequently, if an allowable is known for population 1, then the allowable for population 2 is given by:

$$A_2 = r \times A_1 \tag{A4}$$

If the large-sample method descrihed above is used to derive the allowable A_1 then an estimate of r is all that is needed to find A_2.

A sample of r can be constructed by dividing individual test results from population 2 by individual results from population 1. The number of batches and specimens used in testing population 2 should be equal to or less than the number used for population 1. Individual batches from population 2 are then paired with individual batches from population 1. Individual specimens from batches in population 2 are then divided by randomly chosen specimens from the paired batch in Population 1.

This procedure will result in a sample of r values divided into batches. The number of batches and specimens is equal to the number used for the sample of population 2. This sample of r is then analyzed using the method of [3] to obtain a lower 95% confidence estimate of the median value. This is accomplished by setting 6 = 0 in equation 8.5.4(h) of [3] as required by equation 2.3 of [2]. Taking the lower 95% confidence estimate of the median of r should result in conservative estimates of the allowables for population 2.

INFLUENCE DES PERFECTIONNEMENTS DU CALCUL DES STRUCTURES SUR LA PROCEDURE DE QUALIFICATION DES AVIONS

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RESUME

Nous examinons ici l'influence que peuvent avoir les progrès accomplis dans les techniques de calcul sur la façon de qualifier les structures d'avion.

Nous passons en revue les potentialites, les défauts et leurs corrections possibles, pour les méthodes pratiquées dans les trois grandes branches du calcul des structures d'avion :

- Calcul des contraintes
- Determination des critères de rupture en statique et en fatigue
- Calcul des charges, de l'aéroélasticité et du Flutter.

I1 apparait qu'au depart les risques d'erreurs sont omniprésents, aussi la procedure de qualification doit-elle inclure obligatoirement un processus fiable de detection et de correction de ces erreurs.

Nous presentons l'organisation de conception et qualification des structures mise en place par Dassault. L'analyse sur ordinateur y prend certes une part tres large, mais elle reste contrôlée par un nombre minimum d'essais pertinents.

Finalement, il apparait que les progrès du calcul ont d'abord permis de mieux optimiser la conception des structures, ils ont permis aussi de diminuer considérablement le nombre et le volume des essais partiels et ils ont réduit le risque de rupture imprévue lors des grands essais de qualification. Cependant, du fait principalement des risques d'erreurs humaines dans la manipulation des calculs, la garantie apportée par ceux-ci reste insuffisante pour se permettre d'éviter les grands essais de qualification.

ABSTRACT

We examine the effects of improvements of analytical methods on the process of airframe qualification.

We review potentialities, weaknesses and corresponding corrections for the three main branches of structural analysis of aircraft :

- Calculation of stress fields
- Determination of failure criteria in static and in fatigue
- Calculation of loads, aeroelasticity and flutter.

Risks of errors are omnipresent, so the structure qualification must include a reliable process of detection and correction of errors.

We present the resulting organization of design & qualification reached by Dassault. Computer analysis take a large place, while remaining controled via minimum number of relevant tests.

Progresses of analytical methods have first been profitable to design optimimization, it have reduced the number of required test and it have also reduced the risk of failure during the main qualification tests. However the insufficient reliability of analytical methods, mainly the risk of humain mistakes, requires to maintain the main qualification tests.

1 - INTRODUCTION

Avec les progrès des techniques de calcul par Elements finis réalisés depuis 20 ans, on peut considérer qu'il est possible d'appréhender dans les moindres details les champs de contrainte de nos structures. Nous avons montré à plusieurs reprises (réf 1) que ces facilités de calcul ont déjà permis une meilleure optimisation et une diminution des aléas au niveau de la phase de developpement de nos cellules. I1 reste à discuter ici le point de savoir si ces possibilites de calcul conduisent à une amelioration de la procedure de qualification de nos structures, cette amelioration pouvant se traduire, soit par une meilleure qualité de la démonstration, soit par une reduction du coût de cette demonstration, en particulier en diminuant le nombre et la complexité des essais.

Pour cela, faisons ici un rapide parcours des potentialites, des faiblesses et des procedures correctives des méthodes de calcul utilisées dans les trois principales branches du calcul des avions :

- le calcul des champs de contrainte.
- l'evaluation des contraintes admissibles en statique et en fatigue,
- les calculs des charges, de l'aéroélasticité et du flutter.

Nous montrons qu'au depart le risque d'erreurs est omnipresent, et que la stratégie de développement et de qualification des structures doit obligatoirement inclure un processus de detection et de correction de ces erreurs.

Nous exposons l'organisation de conception/qualification des structures à laquelle Dassault est arrivé ; l'analyse sur ordinateur y Drend une Dart considerable tout en restant contrôlée par un nombre minimal d'essais judicieux.

2 - LE CALCUL DES CHAMPS DE CONTRAINTE

Actuellement pour le calcul des contraintes statique et dynamique à basses fréquences, nous recommandons une organisation de calculs imbriqués les uns dans les autres presentee planche 1 ; Elle comprend :

- Un modele general Elements Finis de l'avion complet

Il sert de base à la resolution de l'hyperstaticité générale de l'avion, au calcul des charges, de l'aéroélasticité et des vibrations basse fréquence.

L'ambition de ce modele est de fournir les "flux" d'efforts internes et non le detail des surcontraintes locales, sa taille optimale est de 30000 à 100000 degrés de liberté (exemple planches 2 et 3).

- des modeles de details qui s'imbriquent dans le modele general ou dans d'autres modeles intermédiaires (voir planche 4). Par une technique de condensation par "Super-éléments", ils prennent des conditions aux limites "exactes" (charges condensées et rigidités) dans les modeles amonts.

Citons quelques uns des plus typiques de ces calculs de details :

- Calculs dit "boulon par boulons", avec des maillages en Elements de flexion d'un pas environ deux fois plus raffiné que celui des fixations. Ces modèles permettent d'accéder directement aux efforts transmis par les fixations : ils prennent en compte la plupart des effets non-linéaires géométriques (post-flambage, matelassage etc..., voir planches 5 et 6).

- Calculs de "point stress" dans les composites qui prennent leurs conditions aux limites dans les precedents et génèrent les champs de contraintes locales autour des fixations (voir planche 7)

- Calculs de ferrure tridimensionnels (planche 8) qui doivent intégrer la non-linéarité de contact et de plasticite.

Nous rattachons les calculs de stabilite à la famille des calculs de contrainte. L'approche de l'analyse des charges critiques du flambage lineaire est certes particulière, mais la tendance est de ne l'utiliser que pour le prédimensionnement. Pour la justification fine de qualification, nous faisons aujourd'hui tres largement appel au calcul d'équilibre en post-flambage (il inclut la plasticite pour les materiaux metalliques).

La réalisation ordinaire de ces calculs n'est possible que par la disponibilité de logiciels ergonomiques et bien rôdés comme notre système CATIA-ELFINI. Les maillages et analyses Elements Finis y sont complètement intégrés dans un système de CAO. Cet outil offre à la fois l'avantage d'une complete intéractivité et celui de la possibilité de recuperation des "historiques" des commandes d'un calcul, ce qui permet d'élaborer très rapidement les données des calculs comportant des similitudes. Ces "historiques" sont donc une façon de mémoriser le savoir faire, et par là de réduire les risques d'erreurs dans les calculs ultérieurs.

L'autre point essentiel est le fait que le coût de tout ces calculs est devenu modeste (moins de 1' CPU sur IBM 3090 VF pour le calcul statique du modèle general présenté planche 2).
Même pour les calculs non linéaires les plus sophistiqués avec les techniques de minimisation du potentiel élastique biquadratique par "Line Search Exact" et celle de la "plasticité implicite" (Réf. 2 et 3), on arrive à un coût de quelques fractions à quelque fois celui du premier calcul linéaire.

Sur le tableau 1, nous résumons, les modes de defaillance les plus typiques et le processus de corrections des differents elements de ces calculs.

Nous donnons notre appreciation de la criticité des modes de defaillance :
- initiale (c.a.d. pour un premier calcul)
- finale (c.a.d. en appliquant les mesures correctives et avec l'organisation de conception/qualification presentee §5)

Les conventions sont les suivantes :

Q	Risques nuls ou négligeables
**	Risques faibles à consequence mineure
***	Risques d'avoir à modifier les structures en service
	Risques graves.

TABLEAU 1

	Types de defaillance	Criticité Initiale	Criticité Finale	Modes de détection/correction
Logiciels Eléments Finis	- Erreurs de théorie (souvent sur la formulation des E.F.) - Erreurs de programmation - Modes d'utilisation trop compliqués	*** *** ***	0 0 0	- Comparaison solutions analytiques - Comparaisons autres solutions E.F. - Tests convergence - Patch test - Verification des résidus de l'équilibre - Comparaison resultats d'essais - Utilisation de logiciel ergonomique
Maillage & autres données	- Maillage insuffisamment raffiné - Type d'E.F. inappropriés - Erreurs charges, C.L. et toutes autres données. - Non representation de la structure réelle.	** *** *** ***	0 0 0 0	- Formation du personnel - Experience du maillage optimal de chaque type de structure - Calculs redondants (notamment comparaison modele general - Modèle de detail) - Comparaisons aux résultats d'essais (essais partiels, essais généraux) - Test de convergence mathématique. - Ergonomie des logiciels (maillage, visualisation resultats, detection d'erreurs).
Analyses des résultats	- Erreurs humaines dans les interpretations par R.D.M. - Lissage des extrémums (E.F. trop grossiers ou par le logiciel de visualisation) - Oubli d'examens de certains types des contraintes - Impasse dans la justification.	** ** *** *	* * * *	- Formation des personnels - Ergonomie des logiciels de post-traitement - Micro analyse E.F. - Système I.A. supportant le dépouillement - Exigence de qualité pour les dossiers de justification - Essais statique generaux

A l'expérience, nous n'avons connu que très peu d'erreurs graves liées purement aux fautes de logiciel. Cela résulte vraisemblablement, avec notre système CATIA-ELFINI, de l'application stricte des règles de detection d'erreurs citées plus haut et surtout de plus de 20 ans d'utilisation avec une verification systematique de non regression à chaque version.

C'est au niveau du maillage, de la confection des données et de l'interprétation des résultats que l'erreur humaine doit être considérée comme quasiment inevitable, cela simplement du fait du nombre de transaction nécessaire pour élaborer la série de nombre par laquelle une structure est representee dans un ordinateur.

Ce caractère inévitable de l'erreur au niveau d'un seul calcul doit être corrigé non seulement par tous les moyens cités plus haut mais surtout par une organisation intégrant des calculs et des essais se recoupant, tel que nous la présentons § 5.

3 - CONNAISSANCE DES CONTRAINTES ADMISSIBLES

Beaucoup d'entre nous ont fait l'expérience malheureuse de ne pas savoir quoi faire des champs de contraintes obtenus après un calcul Elements Finis tridimensionnels raffiné.

En effet, avec les calculs lineaires on obtient des surcontraintes locales dépassant largement le niveau théorique de rupture alors qu'elles sont écrêtées en fait par la plasticite.

En fatigue avec la possibilité de calculer des formes complexes, on ne dispose plus des références classiques (courbe de Wohler), car elles font intervenir le coefficient concentration de contrainte Kt qui n'est plus défini.

Par extension, nous assimilons les calculs de tolerance aux dommages à la connaissance des contraintes admissibles. Les défauts de précisions de l'analyse de l'initiation des fissures sont d'ailleurs pour partie responsable de l'intérêt pour les méthodes de tolerances aux dommages, le calcul de la propagation des fissures en fatigue étant globalement mieux maîtrisé que celui de leur initiation. (voir réf.4 et P1.9).

Avec les materiaux composites ce qui pose problème c'est l'extrême sensibilité du critère de rupture 6 la configuration locale (proportion, empilements et direction des plis, diamètre des trous de fixation, type de fixation, matage, etc. ...).

'analyse des modes de défaillances et des procédures correctives est la suivante :

TABLEAU 2

	Types de défaillance	Criticité Initiale	Criticité Finale	Mode de détection/correction
Critère de rupture statique des métalliques	- Non prise en compte de la plasticité - Méconnaissance des caracteristiques materiaux et des lois rhéologiques	* ***	0 0	- Calcul avec plasticite - Identification rhéologie matériau - Essais éprouvette élémentaire representative.
Critère de rupture en fatigue (initiation de fissure)	- Difficult6 liée à la notion de Kt - Non validite de la règle de Miner (spectre complexe) - Défaut de representation théorique des phénomènes complexes (état de surface, fretting, corrosion, etc.. .) - Dispersion matériaux méconnaissance des caractéristiques. - De façon générale, difficult6 à extrapoler les données expérimentales disponibles vers le cas envisage.	** ** *** *** **	0 0 * * 0	- (Calcul E.F. avec plasticité et cyclage) - Dimensionnement conservatif - Essais partiels représentatifs (configuration et spectre de charge). - Essais de fatigue générale - Inspection en service
Critère de rupture statique des composites	- Extreme complexité des critères de rupture (surtout en zone de fixation) - Difficult6 des analyses de tolérance aux dommages	*** **	0 0	- Verification systématique de la conservativité des critères de rupture. - Essais statique partiel des zones complexes - Essais statique general avec dommage - Dimensionnement pour réparabilité.

TABLEAU 2 (suite)

	Types de defaillance	Criticité Initiale	Criticité Finale	Mode de detection correction
'ropagation de fissure iétallique 'olérances aux dommages	- Mauvaise prediction propagation des petites fissures	*	0	- Dimensionnement conservatif en tolerance au dommage
	- Mauvaise modélisation des propagations en spectre complexe	*	0	- Identification des vitesses de propagation avec spectre reel.
	- Difficult6 d'estimation des longeurs critiques	**	0	- Essais de fatigue partiel
	- Oiagnostique d'arrêt de propagation des fissures "explosives"	***	0	- Essais de fatigue globale
				- Essais spécifiques de tolerance aux dommages (sur CES, CEF, et/ou éprouvettes spéciales)
				- Inspection en service

Pour les matériaux métalliques, le dimensionnement statique à partir des contraintes calculees par Elements Finis ne pose souvent pas de problème de sécurité, cela résulte du fait que les calculs linéaires surestiment les contraintes.

Dans tous les cas, la garantie de qualification est assurée par l'essai statique general.

La justification en Fatigue reste par contre dominée par l'empyrisme ; la démonstration est essentiellement fondée sur des essais, le calcul sert pour l'essentiel à justifier la validite des conditions aux limites ou la conservativité des essais.

Les méthodes de calcul de l'initiation des fissures basées sur l'historique reel des contraintes en tenant compte des écrouissages plastiques locaux, sont certes prometteuses, mais elles ne sont pas encore aujourd'hui suffisamment rodées pour fonder une qualification.

En fatigue, malgré l'application de la procedure experimentale il reste certains aléas, ils sont dus pour l'essentiel aux differences possibles entre les structures essayées et les Cellules en service (en dehors des problèmes de connaissance des charges évoqués §4).

Pour les composites, la situation n'a pu être dominée qu'avec l'utilisation de critères semi-empyrique type "point stress" qui à défaut d'un fondement physique rigoureux, présentent l'avantage de donner une influence aux principaux paramètres (nappage, diamètre fixation, matage, etc...) ; l'idée directrice est d'ajuster systématiquement les paramètres du critère de façon à être <u>conservatif sur l'ensemble des essais disponibles.</u>

Par ailleurs, toutes les situations complexes font l'objet d'essais partiels (dont les C.L. sont vérifiées par calcul). Un essai statique general est de rigueur (incluant les effets de tolerance aux dommages si nécessaire).

4 - CHARGES, AEROELASTICITE, FLUTTER

Les calculs de contraintes et de durée de vie exposes précédemment n'ont de sens que si on connait les charges appliquées sur les avions.

Ces dernières résultent principalement :

- des manoeuvres effectuées
- de la turbulence atmosphérique rencontrée (avion civil)
- des charges de pression aérodynamique
- des repartitions de masse.

Actuellement, nous disposons dans notre système CATIA-ELFINI d'une branche Aéroélasticité (présentée en detail à l'AGARD dans la reference 5) couplant directement les calculs d'aérodynamique stationnaire et instationnaire avec le calcul de structure par Elements Finis ; cet outil fournit principalement :

- Les coefficients aérodynamiques avion souple,
- Les calculs de vitesse critique de divergence statique et de flutter
- Le calcul de réponse structural en manoeuvre et à la turbulence
- Les cas de charges "enveloppes".

Le logiciel donne un grand nombre de possibiliti de recalage des champs de pression aerodynamique sur les resultats de soufflerie ou sur le vol (référence 6).

L'analyse des principaux modes de defaillance est la suivante :

TABLEAU 3

	Types de défaillance	Criticité Initiale	Criticité Finale	Mode de détection/correction
Logiciel & théorie	- Limitation de la validité des calculs aérodynamiques (transsonique, Gcoulements décollés etc.. .)	***	*	- Calibration des calculs aéro. sur soufflerie en vol - Augmentation des marges de calcul
	- Non linéarité servo commande mal representee	**	*	- Simulation avec non linéarité mécanique
	- Linéarisation abusive	**	*	- Ouverture progressive du domaine de vol
	- Mode d'utilisation trop compliqué, manque de test d'erreur	***	0	- Ergonomie du logiciel, test de validité
Procédure de calcul & données aérolasticité	- Maillage aérodynamique inapproprié	***	0	- Formation des personnels - utilisatioii de procédures de calcul reconnues
	- Effet de' troncature de base	**	0	- Comparaisoii soufflerie
	- Mauvais lissage des formes aerodynamiques	***	0	- Etude de convergences
	- Erreur sur les répartitions de masse	**	0	- Utilisation d'une "base de charges" (réf.5) - Calcul alternatif - Recalage du modele sur essais de vibrations - Recalage sur le vol
Sollicitation à considérer	- Normalisation des manoeuvres dimensionnantes	**	0	- Maintien de l'avion dans son domaine de "resistance limite" par CDVE (réf.7)
	- Reconnaissance du spectre d'utilisation de l'avion 6 priori	**	*	- Utilisation fatiguemètre
	- Incertitude des modeles de turbulence	**	?	- Pertinence pour le choix du modèle de turbulence.

La plus grosse difficult6 du calcul des charges vient de l'aérodynamique, cela spécialement dans le domaine transsonique ; il faut admettre que les calculs d'aujourd'hui, même menés de façon conservative, n'interviennent que dans la diminution du risque programmatique. La qualification pour les charges n'est acquise qu'avec le recalage du modèle sur les essais en vol. Nous avons développé des procédures sophistiquées présenté à l'AGARD dans la référence 6.

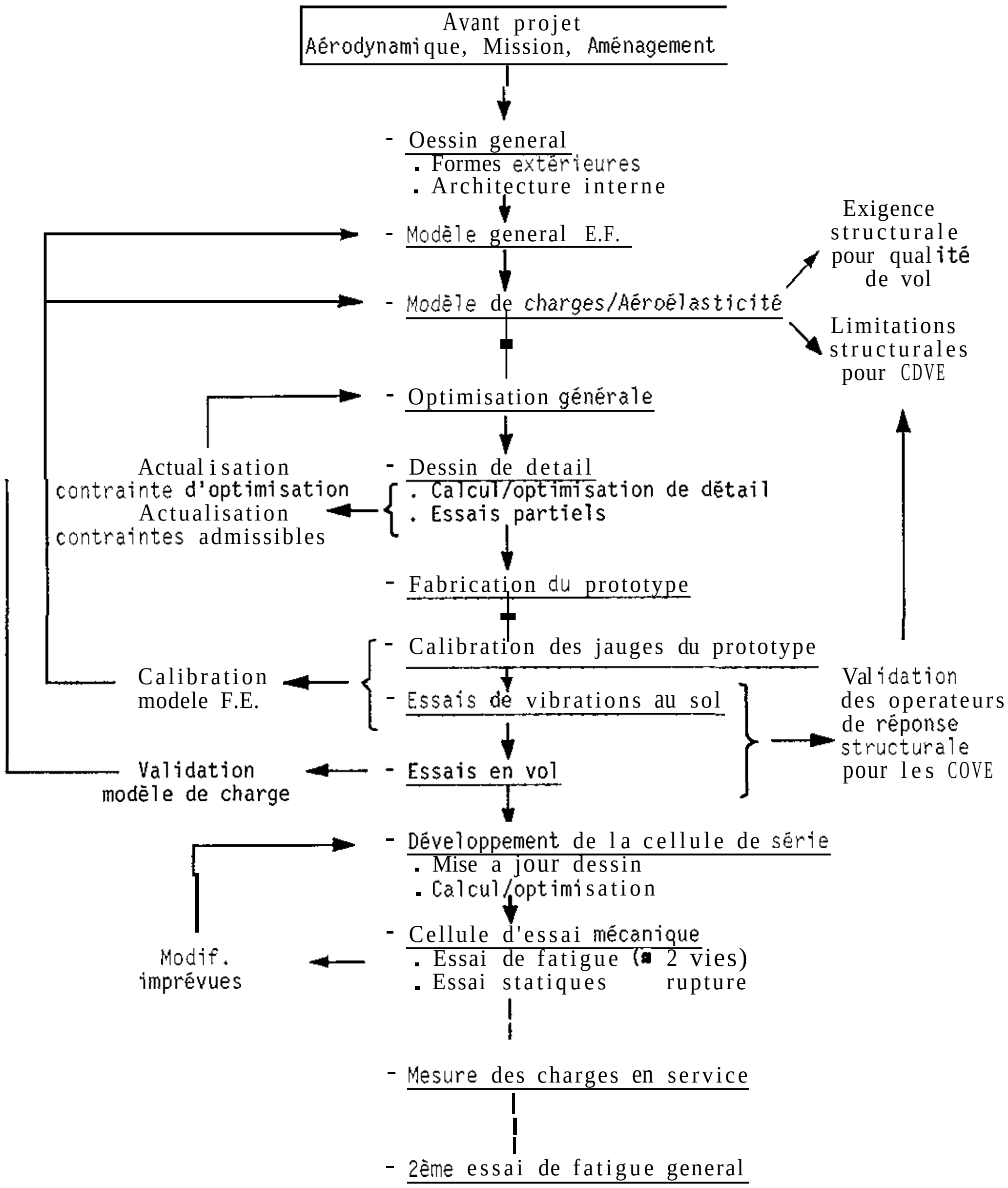
Pour les avions militaires 6 commande de vol électrique (CDVE), nous avons montre 6 l'AGARD (réf.7) que la definition de manoeuvres dimensionnantes "normalisables" Gtait quasiment impossible, et qu'on pouvait avantageusement remplacer cette notion per celle d'enveloppe de resistance limite de la structure dans lequel l'avion est maintenue par les CDVE quoi que fasse le pilote ; nous avons monte que nous pouvions en faire une demonstration rigoureuse, 6 partir du moment ou le modele aeroélastique de l'avion est calibré sur les essais en vol.

Pour la turbulence qui est un évènement extérieur indépendant de l'avion et du pilotage, on ne peut que rester dans le cadre des modèles normalisés, avec la grosse diffilculté qu'ils correspondent ã des situations impossibles 6 rencontrer en essais en vol. Il faut donc faire confiance au modèle de calcul qui doit être validé indirectement (essais de vibration, identification de fonction de transfert à la turbulence mesurée en vol, réponse de jauge de contrainte en manoeuvre, etc.. .).

Les inconnues sur l'utilisation réelle de l'avion au moment de sa conception sont des facteurs d'incertitude évidents pour l'evaluation de la durée de vie. Cela nous amène 6 préconiser l'usage systematique de fatiguemètre sur tout ou partie des avions en service et éventuellement de refaire des essais de fatigue pour pouvoir prolonger des cellules anciennes (voir § 5).

5 - INSERTION DE LA DEMONSTRATION DE QUALIFICATION DANS LE PROCESSUS DE CONCEPTION

Un compromis entre les considerations de qualification précédentes, les impératifs de conception optimale, de délais et de coût nous a conduit à l'organisation suivante :



Les points clef de la validation des modeles et de la qualification sont les suivants :

1er point - Essais partiels

Ils concernent principalement la determination des contraintes admissibles dans les pièces complexes (en fatigue pour les métalliques, en statique pour les composites).

En s'appuyant sur des modèles de calcul E.F. détaillés, on peut à la fois simplifier notablement les éprouvettes et ajuster exactement les conditions aux limites pour représenter au mieux la configuration visée.

2ème point - Essais de calibration des jauges de contrainte du prototype

Les charges subies par l'avion sont identifiees (voir réf.6) au travers de la réponse en contrainte de l'avion mesurée en vol par des jauges.

I1 importe donc absolument de valider le modele Elements Finis de l'avion qui fournit l'opérateur charges appliquées-contraintes.

Pour cela, on soumet au sol le prototype à plusieurs dizaines de chargement unitaire ; on effectue ensuite la correlation entre contraintes mesurées et contraintes calculées cela pour tous les chargements. L'expérience nous montre que cette corrélation est généralement assez bonne même avec le premier modèle, et devient excellente quand tous les défauts sont corrigés.

3ème point - Essais de vibration au sol

Les mesures directes des caracteristiques modales permettent la validation du modele dynamique de l'avion. La corrélation est tres souvent bonne même avec le modele initial, nous disposons de toute façon d'une méthode tres efficace de calibration automatique du modele E.F. sur les resultats d'essais de vibration (sous produit des techniques d'optimisation)

Remarques importantes :

Nous considérons la correlation de l'essai de vibration et celle des réponses des jauges mesurée avec les calculs sous differents chargements statiques comme les meilleurs arguments de la validation du modèle Elements Finis statique et dynamique.

4ème point - Essais en vol

Ils permettent la calibration simultanée des modèles de mecanique du vol, de charges et d'aéroélasticité, à partir de la réponse mesurée en vol des parametres de vol et des jauges de contraintes.

Nous avons present6 la méthode en detail que nous utilisons dans la réf.6. Les influences de chaque effet aérodynamique unitaire sont identifiées à partir des réponses sur des manoeuvres d'oscillation de tangage et de roulis à fréquences variables dites "stimulus". Une technique d'identification mathematique permet ensuite de procéder directement au recalage des champs de pression.

Les essais en vol qualifient aussi les operateurs de "surveillance" des contraintes structurales par le système de commande de vol électrique.

5ème point - La cellule d'essais mecanique

C'est 6 ce niveau que l'influence des possibilités de calcul actuelles se concrétise.

Nous pensons qu'il est possible sur la même cellule de procéder successivement 6 un essai de fatigue (environ 2 vies) et un essai statique menés à charge extrême puis à rupture.

L'idée directrice est que nous ne pouvons pas abandonner ces types d'essais. car nous ne pouvons pas garantir qu'aucune faute de dessin ne passe nos filtres d'analyses et d'essais' partiels. Cependant, nos previsions de calcul sont suffisamment bonne pour que le risque de retard de l'essai statique du fait de defaillance en fatigue soit tres faible (par exemple les essais statique et de fatigue du Mirage 2000 et du Rafale A se sont déroulés sans incidents sérieux).

6ème point - Charges en service

L'expérience nous a montré, surtout pour les avions militaires, que l'utilisation de l'avion pouvait être différente de celle qui était prévue lors de la conception. C'est pourquoi. il est utile d'identifier assez finement les charges en service 6 1 aide de fatiguemètres. Ceci devient encore plus nécessaire si le client demande la prolongation des cellules au dela de la vie prévue.

7ème point - Deuxième essai de fatigue

Cette idée résulte du fait qu'on peut quelquefois mettre en cause la qualité de l'essai de fatigue effectué en début de programme, cela pour deux raisons principales :

- Le spectre d'utilisation n'était pas encore bien connu,
- I1 y a toujours un certain nombre de modification des cellu'les les premieres années de production, qui font que l'avion de série tend à ne plus devenir tout à fait conforme à la CEF.

Il en résulte un intérêt 6 effectuer un essai de fatigue quelques années après la mise en service de l'avion.

Avec cette politique, les premieres années resteit garanties par le ler essai de fatigue sur la cellule d'essais mécaniques.

6 - CONCLUSION

Les progrès des méthode de calcul servent principalement à l'optimisation des structures et à iminuer les risques de surcoût et de retard du programme resultant de defail lances pendant les essais de qualification.

La fiabilité des calculs n'est pas encore suffisamment totale pour éviter d'avoir à pratiquer les grands essais de qualification (essai statique general, essai de fatigue général).

Les causes les plus remarquables de ce relatif manque de confiance dans les calculs sont les suivantes :

- Un certain défaut de precision des critères de rupture,
- Les difficultés des calculs d'aéroélasticité en transsonique,

et surtout :

- Les risques d'erreur humaine de toutes sortes dans les calculs notamment lors de l'élaboration des données et de l'interprétation des résultats.

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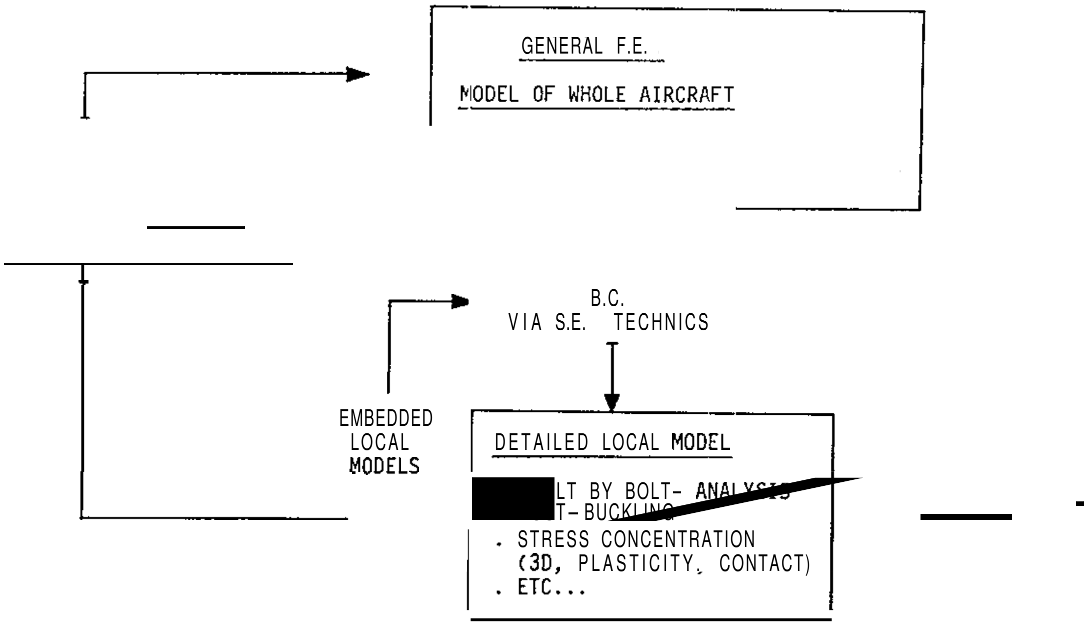
ORGANIZATION OF CALCULATIONS



– PLATE 1 –

GENERAL MESH OF COMBAT AIRCRAFT

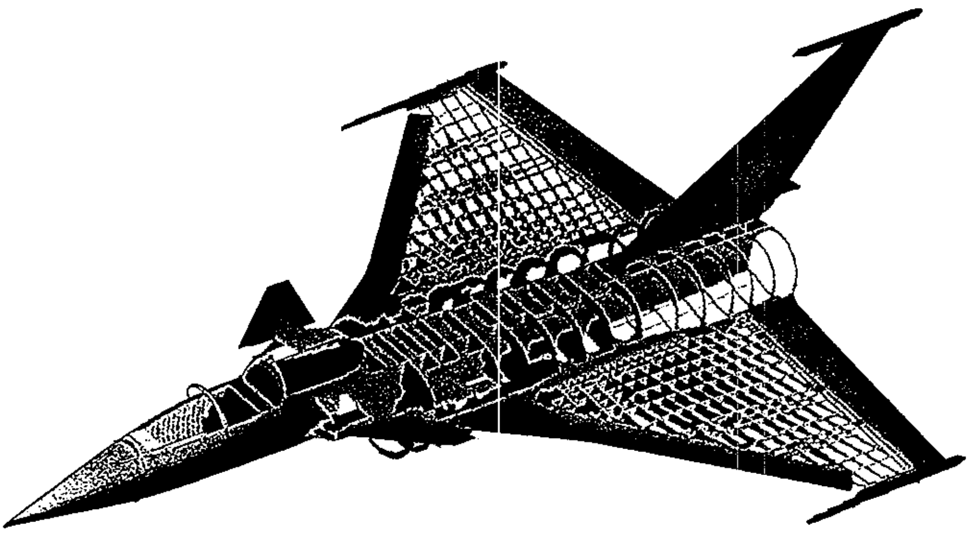
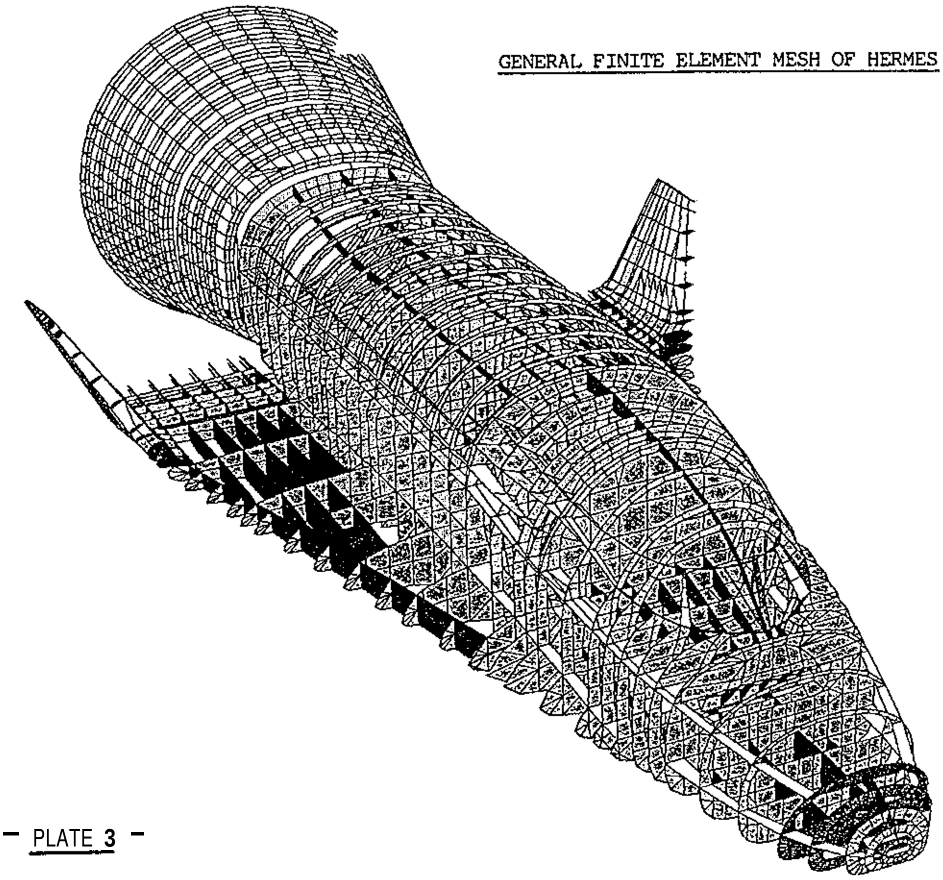


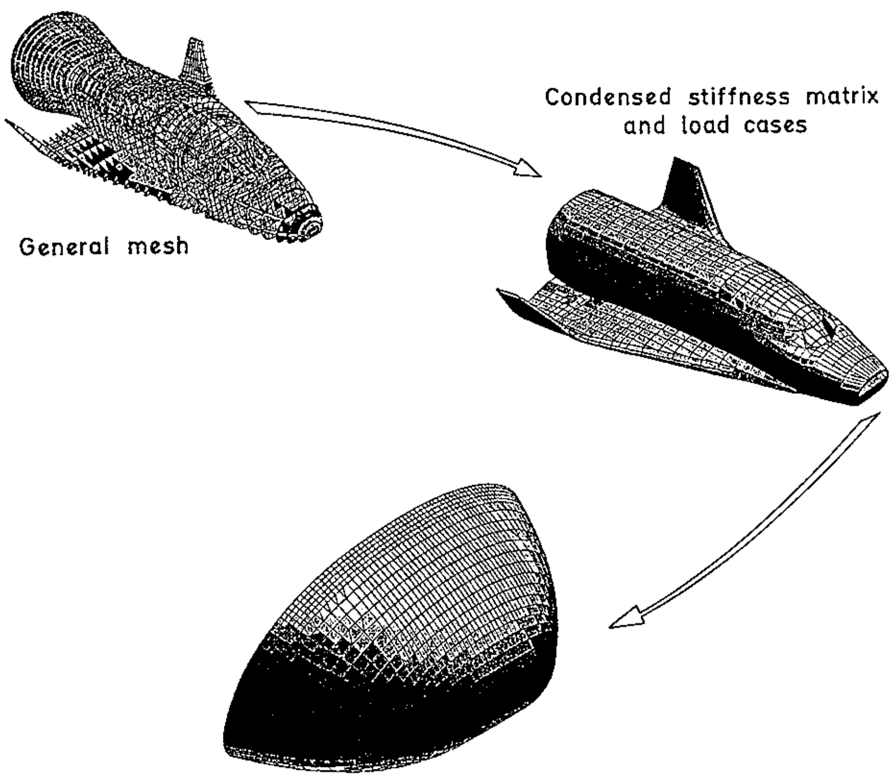
PLATE 2

GENERAL FINITE ELEMENT MESH OF HERMES



– PLATE 3 –

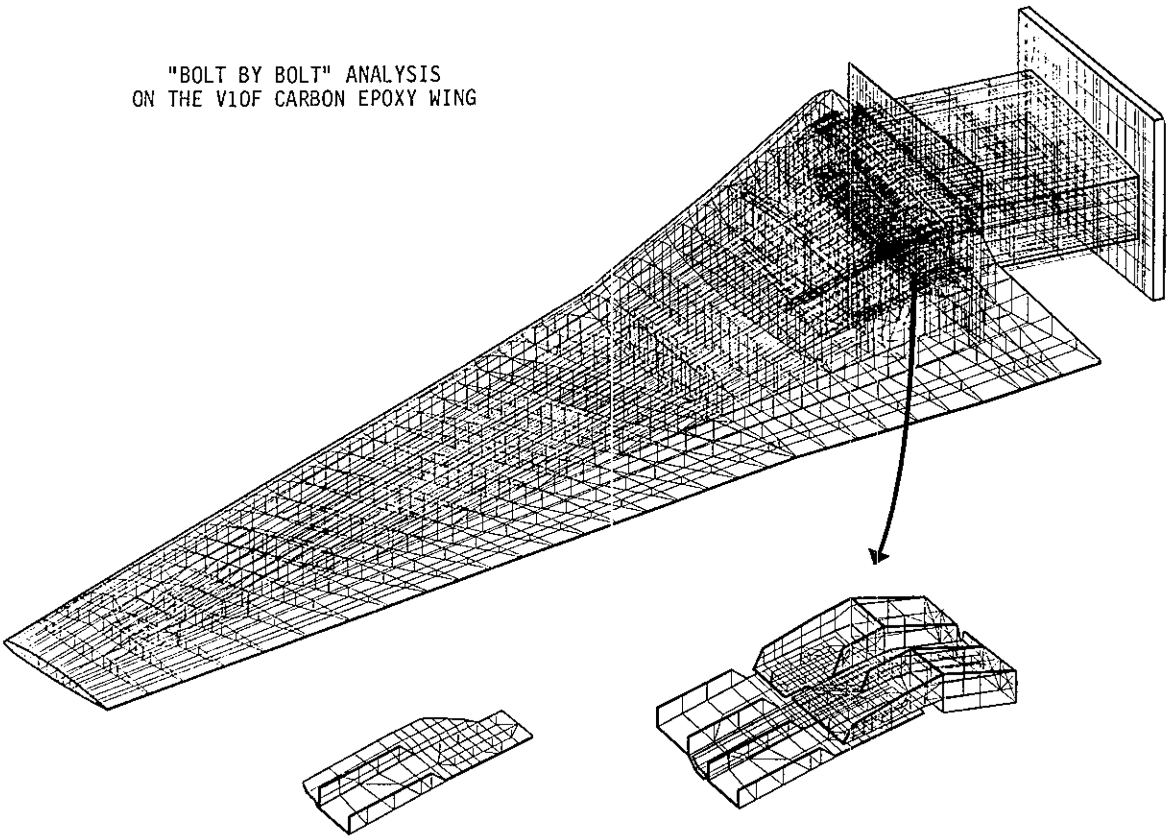
GENERAL F.E. MESH DELIVERS PROPER B.C.
TO SECTION PARTNERS VIA S.E. TECHNICS



– PLATE 4 –

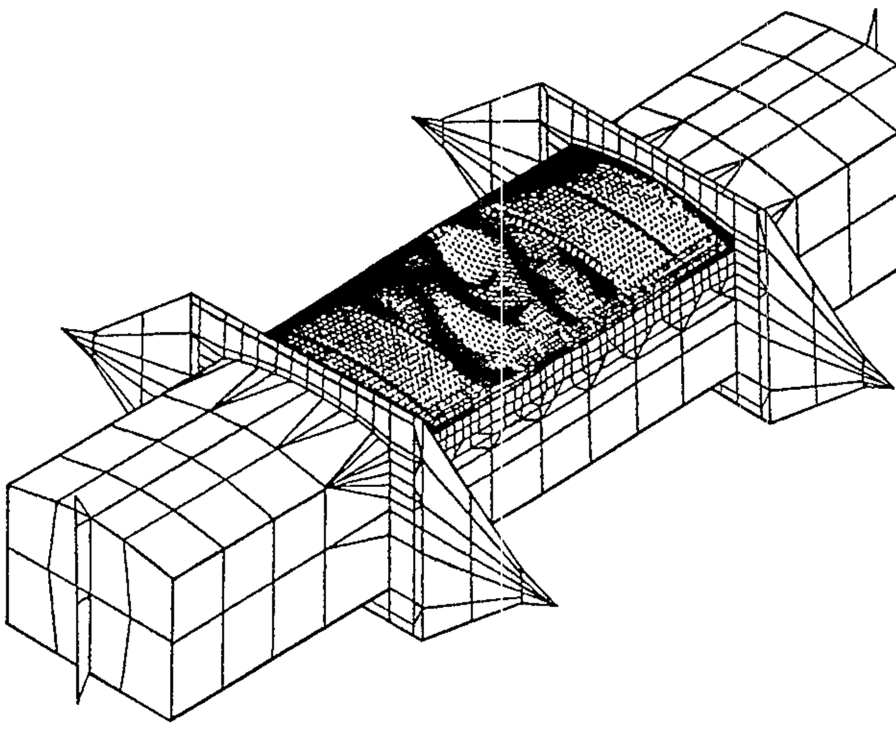
Refined local analysis

"BOLT BY BOLT" ANALYSIS
ON THE V10F CARBON EPOXY WING



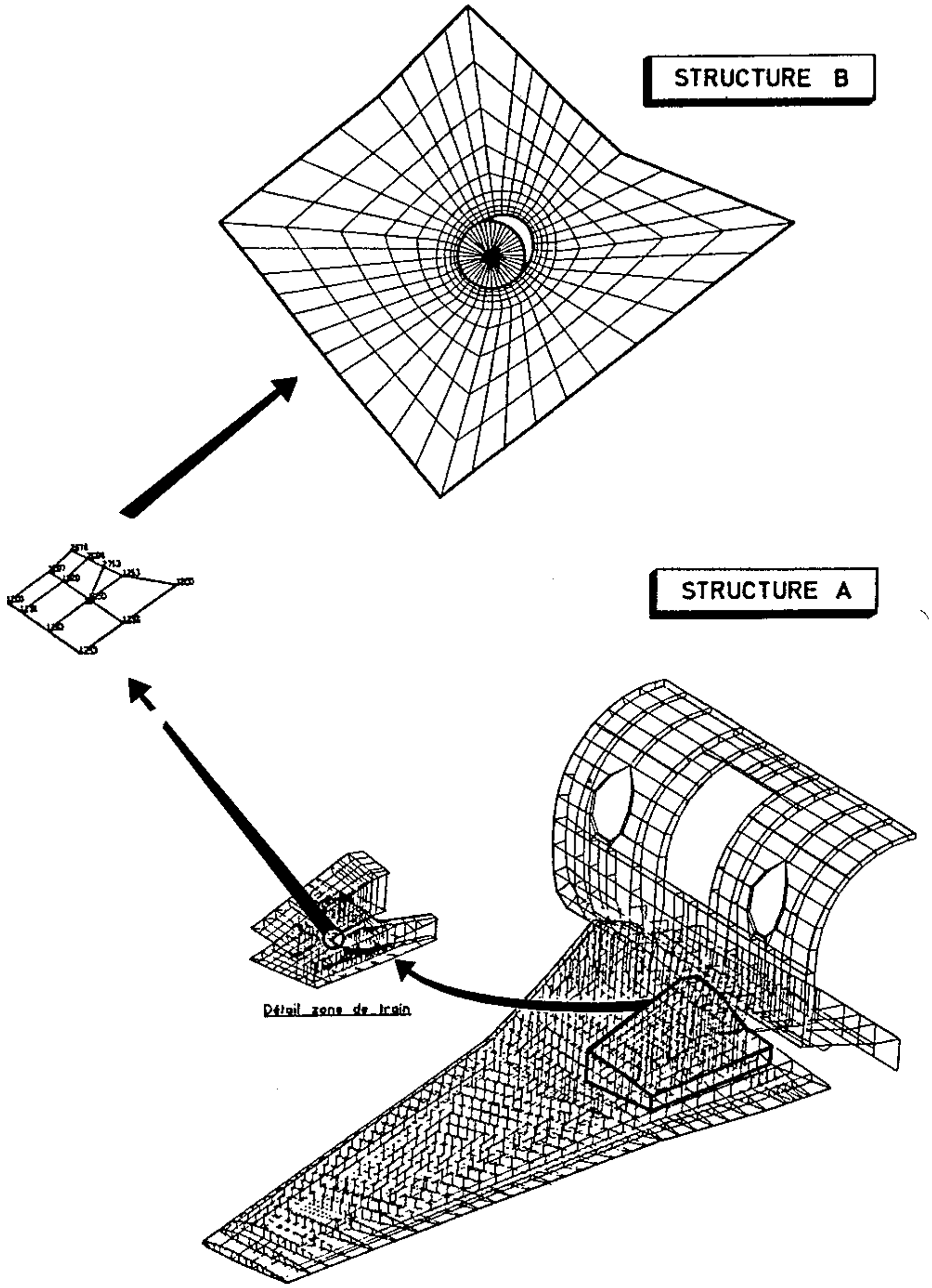
— PLATE 5 —

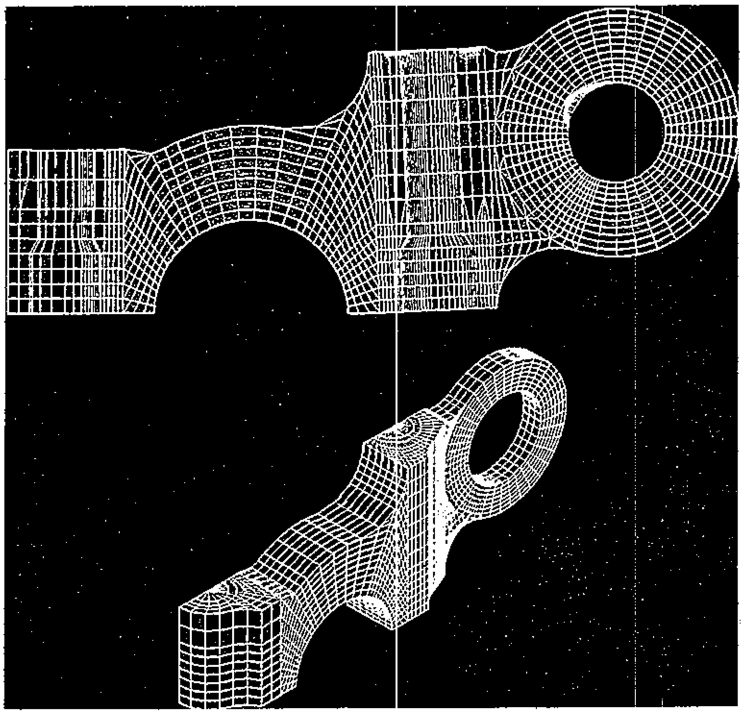
POST-BUCKLING ANALYSIS OF A CURVED CARBON EPOXY PANEL
(TEST ON FUSELAGE PANEL OF COMBAT AIRCRAFT)



— PLATE 6 —

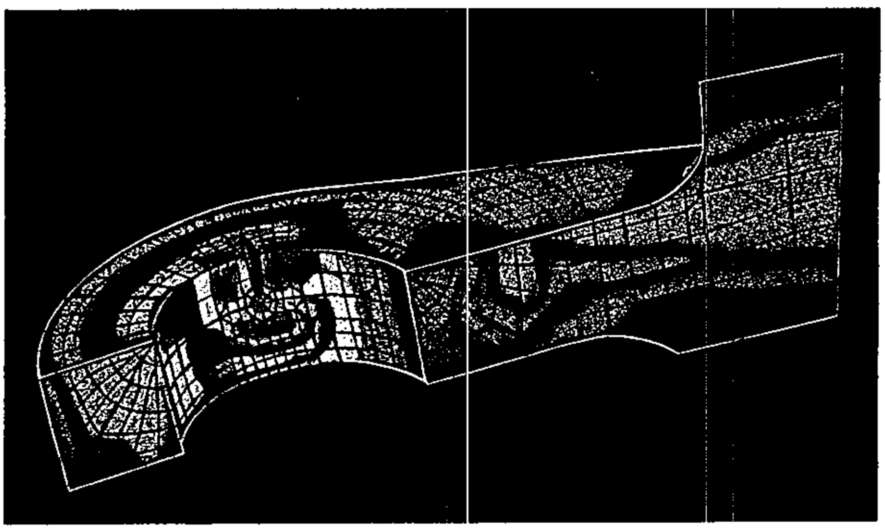
"AUTOMATIC POINT STRESS" ANALYSIS
AFTER "BOLT BY BOLT" ANALYSIS





FITTING TRIDIMENTIONAL ANALYSIS

— PLATE 8 —



TRIOIMENTIONAL ANALYSIS OF CRACK PROPAGATION

— PLATE 9 —

ANALYTICAL METHODS FOR THE QUALIFICATION OF HELICOPTER STRUCTURES

by

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SUMMARY

Analytical methods are more and more becoming the primary means of demonstrating structural integrity, durability, and crashworthiness of helicopter structures, both for civil and military use, as the costs of experimental testing are increasing, when representative conditions of, e.g., temperature, moisture, cyclic loading, and impact have to be considered.

At MBB, the airframes of derivatives of the basic BO 105 have been qualified in the past without retestina. as it could be shown that the structure conformed to those for which experience has shown the structiral analysis to be reliable.
During the development of a composite fuselage for the BK 117, both an analytical and an experimental strength substantiation was performed, which corresponded very well and formed the basis for airworthiness qualification of this experimental helicopter under flight testing now.

A large number of components in the dynamic system are designed primarily so that they will provide adequate fatigue strength, defined in terms of an endurance limit, or in terms of fatigue life. The analytical qualification of these fatigue critical structures, on the basis of measured fatigue loadings and calculated working S-N curves, is state of the art in the helicopter industry.

In the nonlinear domain, analytical methods have been applied for highly laminated elastomeric bearings and for the crashworthiness qualification of both crushable subcomponents and complete helicopters.

The results gained so far allow the application of analytical methods, partly in combination with coupon or component testing, for the qualification of helicopter structures.

1. INTRODUCTION

The objective of the structural qualification is to assure and to demonstrate that the helicopter has adequate structural strength, durability and safety.

Helicopter structures, both for civil and military use, have to be qualified with respect to airworthiness standards, which prescribe the stiffness and strength to be provided in the rotorcraft as a whole and in its components.

Since the potential for accidents of military helicopters is great, due to their mission and the environment in which they must accomplish that mission, crashworthiness requirements must be fulfilled as well

Without doubt, full scale tests with series production components under realistic loading and environmental conditions can be regarded to be best suited of demonstrating compliance with the mandatory design requirements. Structural tests, however, are expensive and the cost of experimental testing is even increasing, when representative conditions of temperature, moisture, cyclic loading, and impact have to be considered; therefore analytical methods are more and more becoming the primary means of demonstrating structural integrity and crashworthiness. The analytical predictions for stiffness, stress, and deformation are validated by structural tests on coupons or components.

As finite element analysis has become both cost effective and easy-to-use in linear as well as in nonlinear applications, this tool can be regarded as state of the art not only in the project and design stage, but also in the qualification phase of helicopter structures.

2. AIRWORTHINESS QUALIFICATION

Airworthiness qualification of a new model helicopter is dependent upon demonstration of compliance with specific design requirements and airworthiness standards. Since many requirements are applicable to individual subsystems of the helicopter, it is appropriate that the adequacy of the major subsystems be demonstrated individually. Many individual components used in such subsystems, however, must be qualified separately in addition to being qualified in the subsystem installation. The methods of qualification vary greatly since the requirements differ significantly among subsystems and components.

The airframe structure, e.g., is subject mainly to static loads, where yielding generally is not permitted at limit load and a positive margin against ultimate load is to be maintained.

A large number of helicopter components, as found, e.g., in the dynamic system, are subject to oscillatory or repeated loadings of sufficient magnitude and frequency that fatigue rather than static strength is the critical structural design consideration.

2.1 Airframe Structure

The basic airframe structure is loaded, in general, by a complex system of external loads, e.g., rotor loads that are reacted by the inertia of the helicopter and its contents. The internal member loads generally are distributed among redundant load oaths in a manner deoendent upon the relative stiffnesses of the various load paths.

Fatigue considerations usually are less significant for airframes than for dynamic components. When the vibratory loads,transmitted to the airframe,are reduced to the levels necessary for pilot comfort, the fatigue stresses are at a level where good design, proper material selection, and sufficient fabrication quality will result in an adequate fatigue life. Also, except at the major load application points, the airframe is sufficiently redundant to provide a reasonable amount of fail-safety should a fatigue crack develop. Steady stress levels also are lower in the airframe structure than in dynamic system components because most of the structure is critical for elastic stability rather than for material strength.

For airframe structures, both made of metals and composites, the overall approach for structural analysis is essentially the same. Finite element method:, are used to calculate the load paths as well as the distribution and magnitude of the field stress or strain within the main structural components of an airframe. Normally a fully three-dimensional analysis of the structure is necessary, due to the fact that helicopters with their large fuselage cut-outs often are asymmetric and are subject to asymmetric loading. Traditional methods are used to analyze local regions of concentrated loadings, as e.g., joints, lugs, and cross-section changes.

If once the analytically predicted load paths, local strengths and failure modes have been verified by structural testing, it is common practice and an accepted means to qualify the airframe structure by analytical methods only, if the helicopter is modified subsequently.

At MBE, the BO 105 CBS and BO 105 LS, both of them derived from the basic version EO 105 CB, have thus been qualified without retesting. The BO 105 CBS is a slightly stretched version and the EO 105 LS additionally has a higher gross weight and increased engine power.

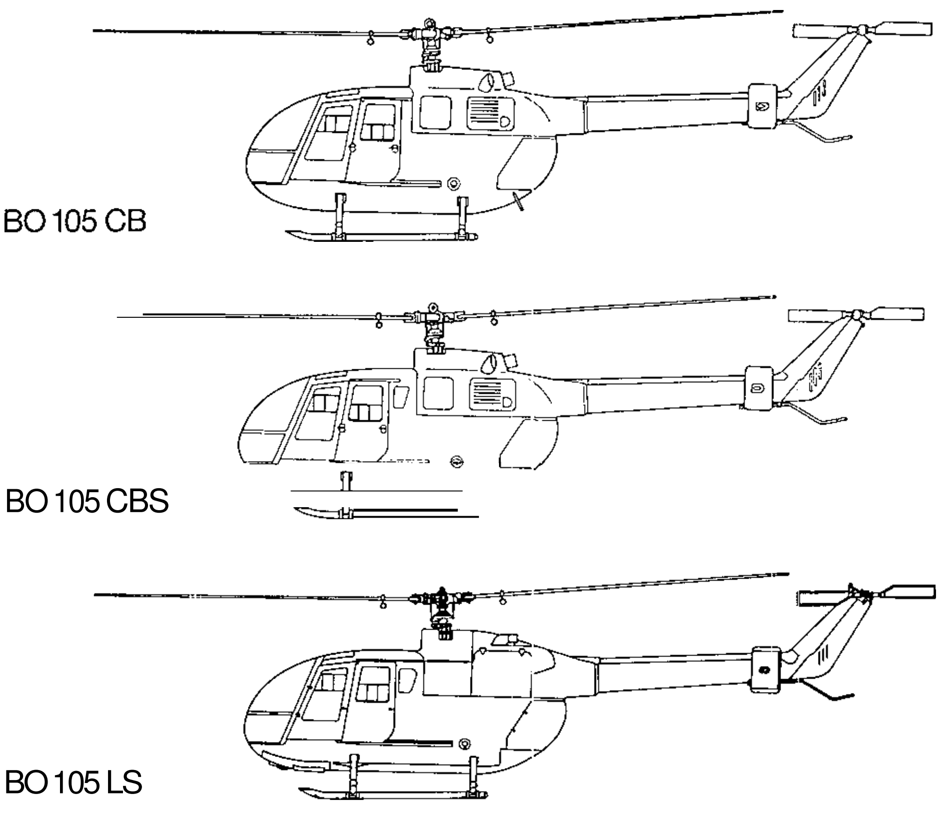

Figure 1. EO 105 Familiy

The finite element method can be regarded as state of the art to be used for structural analysis of helicopter airframes. Already at the project stage comparative analyses with varying designs can be carried out very rapidly to fix the design. In the following design phase structural optimization can be achieved to a certain degree, supported by design development tests to validate the basic design concepts, material selection, and critical or new configuration details. Finally, the integrity of an airframe structure can be validated by establishing the minimum margins of safety in the different structural components for all critical loading conditions under consideration of the respective allowables.

A composite airframe programme was started at MBB in 1985 to design, manufacture, and flight test a composite helicopter fuselage on the basis of a BK 117 111. The complete primary fuselage structure below the engineltransmission deck is substituted, as shown in Figure 2.

Figure 2. BK 117 Composite Airframe

The resulting composite fuselage is 75 per cent carbon fibre composite, because of its high strength and stiffness, for monolithic frames and beams as well as for highly loaded sandwich panels in the subfloor structure in combination with aramid fibre composite to improve the crash impact behaviour. Aramid fibre composite is also used in low loaded sandwich panels.

Based on a three-dimensional finite element model which was available from the metallic fuselage, an internal load analysis was carried through after the model had been updated and the composite relevant properties had been introduced. The difference between the metallic and the composite fuselage is mainly the different design concept, where the sheet/stringer panels are replaced by composite sandwich panels, and the anisotropic material behaviour of the composites as compared with the isotropic behaviour of metals. The complete finite element model consisted of about 2000 structural nodes and more than 4000 elements, as shown in Figure 3. A linear analysis was conducted with NASTRAN.

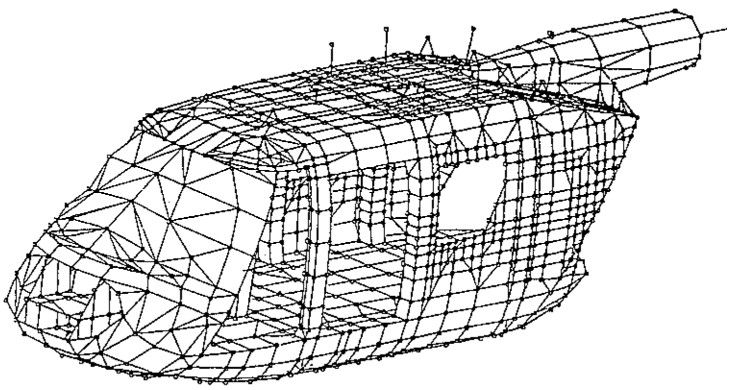

Figure 3. Finite Element Model of BK 117 Composite Fuselage

The analysis showed that the pullup manoeuvre to 3.5 g is critical for the forward fuselage and the level landing with drag is critical for the rearward part.

Based on the internal load analysis, structural adequacy was substantiated by comparing the critical stresses or strains, computed for each structural component against the relevant allowables, where the following parameters are of importance:

- fibre constituency,
- resin formulation and cure
- fibre alignment and distribution within a ply,
- fibre to matrix volume fraction,
- ply orientation and stacking seauence within the laminate.
- section geometry and
- environmental condition.

Besides values taken from literature, material properties and design allowables were established by conducting a test programme that comprised about 800 coupons, including rivet joints, tested at different environmental conditions.

The following criteria have been considered for the determination of individual margins of safety:

- no first ply matrix failure up to limit load,
- no first ply fibre failure up to ultimate load,
- 6 mm/m maximum tensile strain at ultimate load,
- 4 mm/m maximum compressive strain at ultimate load,
- no local instabilities up to ultimate load,
- no global instabilities of sandwich panels up to ultimate load.

In cooperation with IABG in Ottobrunn, an experinental strength substantiation was carried out to verify the airworthiness of the airframe, but also to validate the applied analytical tools by comparing calculated values with measured results. Tests have been conducted on component as well as on complete airframe level.

The component consisted of the right-hand composite side shell, as shown in Figure 4, and a part of the original metallic engineltransmission deck as well as of a dummy subfloor structure.

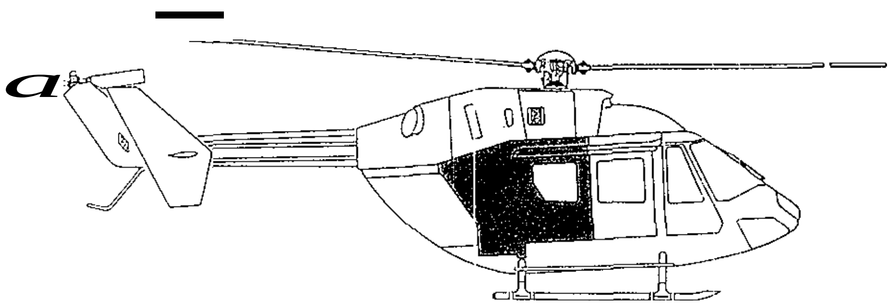

Figure 4. Side Shell Test Conponent in BK 117 Airframe

To simulate realistic boundary conditions, the dummy subfloor structure was attached to a test fixture and the engineltransmission deck was supported at each frame by a pair of struts. The loads were applied on the original load application points of the engine/transmission deck and the rear landing gear. A schematic view of the test set-up is shown in Figure 5 [2].

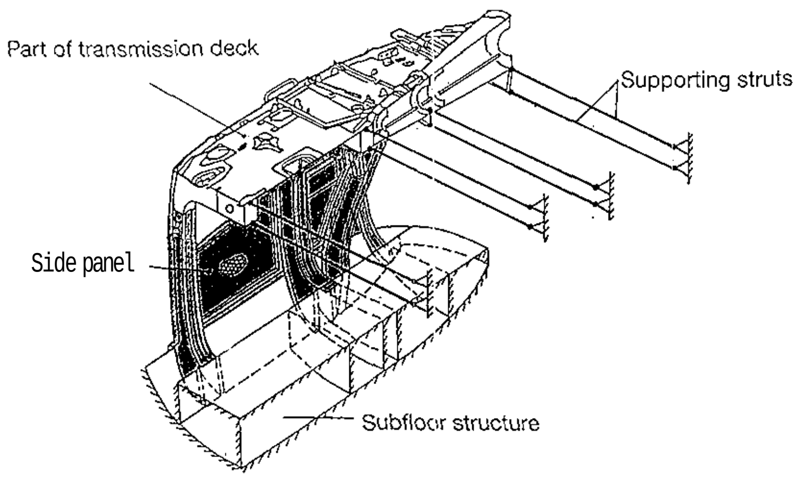

Figure 5. BK 117 Composite Side Shell Test Set-Up

The side shell component was analyzed and tested under the following loading conditions:

- pullup to 3.5 g,
- level landing with drag,
- shear in the sandwich panel and
- compression in the rear landing gear frame.

A comparison of calculated and measured strains of the landing load case at limit load and room temperature in the oblique frame, as the highest loaded menber, can be seen in Figure 6 and shows a sufficiently good conformity.

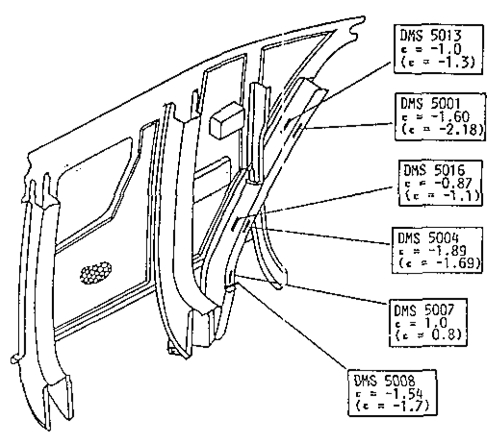

Figure 6. Measured and Calculated Strain:: in the BK 117 Composite Side Shell

After tests up to limit load at room temperature without any failure or unusual behaviour, the component was tested at 70°C. The following failure modes were observed (Figure 7):

- in the flight load case the front frame failed at 155% limit load in the connection to the subfloor structure (failure 1),
- in the landing load case the oblique frame failed at 200% limit load in the middle between subfloor structure and rear frame (failure 2),
- in the shear load case the sandwich panel buckled between the window and the rear landing gear frame with subsequent local delaminations at 200% limit load (failure 3),
- in the compression load case the rear landing gear frame failed at 220% limit load near the connection to the engine/transmission deck (failure 4).

The component test showed that the analytical tools, used for sizing of the individual structural members, are able to predict local stress and strain levels with sufficient accuracy.

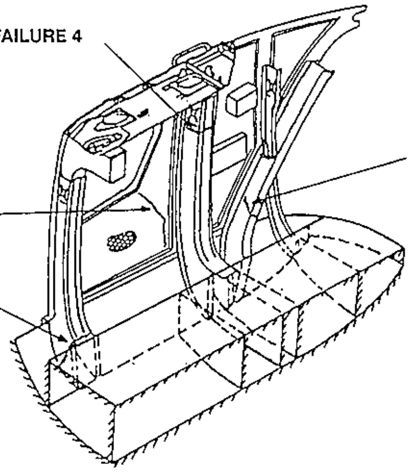

Figure 7. Failure Modes in the BK 117 Composite Side Shell

The complete airframe consisted of the fuselage as well as of the original metallic tailboom with empennage and a dummy landing gear for load application purpose. The primary structure of the fuselage comprises the cockpit without windshields, the side shells, and the subfloor structure, all made of composites, as well as the original metallic floor and the engine/transmission deck, where dummies of the hydraulics, the transmission and the engines were attached for realistic load application, as shown in Figure 8.

Figure 8. Complete Airframe Test Component

The airframe, which was slightly reinforced in the area where the previously tested side shell failed first, was tested at room temperature under the two critical loading conditions up to 150% limit load without any failure or unusual behaviour. The test set-up is shown in Figure 9.

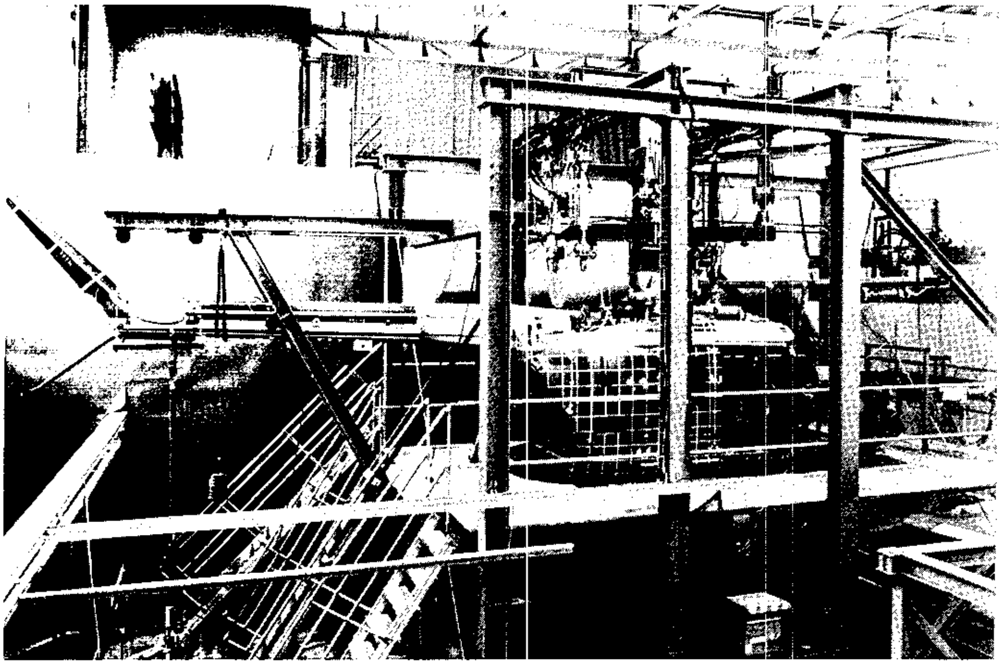

Figure 9. BK 117 Composite Airframe Test Set-Up

From the measured strains, from coupon tests, and analytical investigations the following failure modes can be expected at 70°C and 70 per cent relative humidity:

- in the flight load case the front frame of the side shell to fail near the connection to the engineltransmission deck at about 180% linit load,

- in the landing load case the oblique frame to fail in the middle between subfloor structure and rear frame at about 175% limit load.

Besides the qualification of the BK 117 composite airframe for the "3.5 g pullup" flight load case and the "level landing with drag" landing load case, the complete airframe test validated the finite element model which was used to analyze the airframe and whore sufficiently good conformity with the measured stress and strain values was found. As it is a practical impossibility to qualify by test an airframe structure for every loading condition a helicopter might encounter, the validated finite element model was used to analyticallv oualify the BK 117 comoosite airframe for all other critical loadina conditions. such as "yawing conditions', "level landing with side load", "one-skid landing", and "emergency landings".
Based on the above shown tests and analyses as well as on damage tolerance investigations, additionally performed at the side shell component and the complete airframe, airworthiness of the BK 117 with a composite fuselage was qualified and the helicopter is under flight testing (Figure 10) since spring 1989 without any problems.

Figure 10. BK 117 Composite Airframe Test Aircraft

2.2 Dynamic System

As a helicopter operates in a severe fatigue environment, because it has a rotor that constantly generates a cyclic load input, a large number of components in the dynamic system are designed primarily so that they will provide adequate fatigue strength. But in addition, of course, a static substantiation is necessary that uses a limit load envelope which includes any peak loads that would be experienced in extreme, but seldom encountered, manoeuvres or that is calculated on the basis of structural design criteria.

The frequencies of the cyclic input loads range from once or less per flight to multiples of the rotor speed. Because the number of cycles accumulates rapidly at the higher frequencies, fatigue damage cannot be permitted for the load levels that occur at these frequencies. High loads, on the other hand, should not occur at frequencies high enough to accumulate a critical amount of fatigue damage.

Fatigue substantiation of helicopter structures has to insure that the components have sufficient fatigue strength throughout their design life. Fatigue strength of a given component can be defined in terms of an endurance limit, or it can be stated in terms of a fatigue life. The endurance limit is the maximum value of alternating stress to which the component can be subjected for an infinite number of cycles without failure. Fatigue life is that number of stress cycles that can be sustained prior to failure.

In the helicopter industry the fatigue strength is usually determined by full-scale fatigue tests for all critical structural components, defined as components subjected to significant fatigue loading, the failure of which would contribute to, or cause a failure condition which would prevent the continued safe flight and landing of the rotorcraft.

For components, such as control system parts, for which stiffness is a primary design criterion, all alternating stresses can be below the endurance limit and thus the fatigue life would be infinite. In such cases no fatigue testing is necessary to demonstrate infinite life if allowable stress levels are established by acceptable means, and the stresses measured in flight are lower than these established levels. The following method is considered acceptable at MBB for establishing allowable stress levels.

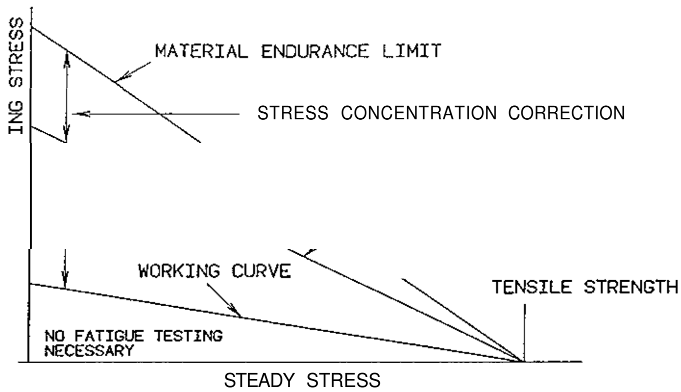

Figure 11. Analytical Qualification of Infinite Fatigue Life

As many helicopter components are subjected to steady loads with alternating loads superimposed, the relation between steady stress and alternating stress is important. The modified GOODMAN diagram which represents the endurance limit as a straight-line relationship of the steady stress, as shown in Figure 11, is commonly used in industry and is constructed from laboratory test data determined on smooth specimens. This endurance level for a given material will be reduced by a calculated theoretical stress concentration factor for the configuration of the component and the imposed loads. A factor of safety of 3 is then applied to the component endurance limit to arrive at a working curve which provides the allowable stress level.

Stress concentration is always present in cases of cross-sectional changes such as notches, grooves, fillets, holes, corners, undercuts, cut outs, etc., but it is also found in unsymmetrical and eccentrically loaded parts that must necessarily bend with each application of load. The theoretical stress concentration factor can be derived from handbooks or data sheets, if the component is simple or of a standardized shape. If, however, a complex part is to be considered, computational mechanics, e.g., finite element analysis is becoming the primary means to calculate the distribution and magnitude of stresses, and this analytical method is superseding photoelasticity in the establishment of stress concentration factors.

Pre-processing is the first step in finite element work. The geometry, the finite element mesh, material properties, loads, and boundary conditions must be loaded into the computer. At MBB we use either CAEDS or MENTAT for pre-processing activities. Depending upon the problem to be analyzed, either NASTRAN or MARC will be applied. NASTRAN for linear applications and MARC for nonlinear material properties and large deformations. Postprocessing is a way of representing finite element results in plots or graphs that make them easier to interpret. Here again CAEDS or MENTAT are used. Figure 12 shows pre- and post-processing plots as well as a fatigue test result of an elastomeric bearing housing.

The finite element model, as shown in Figure 12, also will be used to estimate the detailed stress fields on the component under all relevant static loadings and thus enables the analytical qualification by establishing the minimum margin of safety under consideration of the material design allowables.

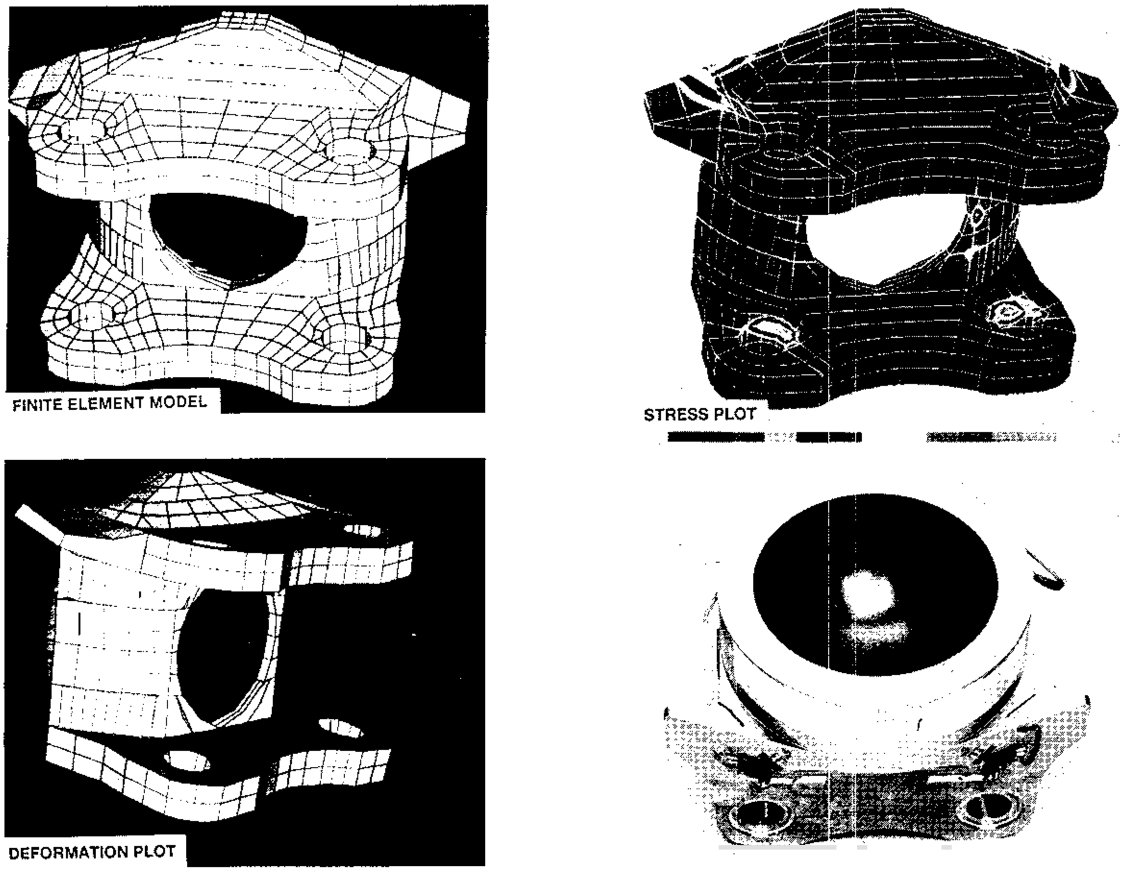

Figure 12. Finite Element Analysis and Fatigue Test Result *of* an Elastomeric Bearing-Housing

When the maximum measured fatigue stress lies above the working curve (Figure 11), the component must be considered to be fatigue critical and hence subject to fatigue analysis. In this case the loading must be defined in detail, and the fatigue strength of the component must be determined. The fatigue life of the component then can be predicted analytically by applying the well known "Linear Cumulative Damage Hypothesis" or Miner's Rule. The fatigue loadings cannot be determined without flight tests. To establish a component's fatigue strength, at MBB, the fatigue test results are condensed into the following analytical expression, proposed by WEIBULL, which covers tie whole range from $N = 1$ (static strength S_u) to $N = \infty$ (endurance limit S_∞):

$$S = S_\infty + (S_u - S_\infty) \cdot e^{-\alpha(\log N)^\beta}$$

Best estimates of the unknown constants S_u, S_∞, α and β are obtained in fitting the above equation 'of the S-N curve to the fatigue life test data by applying the method of least squares. For an acceptable level of risk, the above found mean curve must be reduced by an appropriate reduction factor to define a working S-N curve or to establish an S-N-P diagram. This factor is based on the type of material being tested, past service experience with the material, and type of 'design. It includes consideration of the number of specimens tested and the variability of the fatigue results. Figure 13 shows the mean and working S-N curves of a hypothetical fatigue life problem *of* the American Helicopter Society [3], which where established according to the MBB method, as shown in detail in [4].

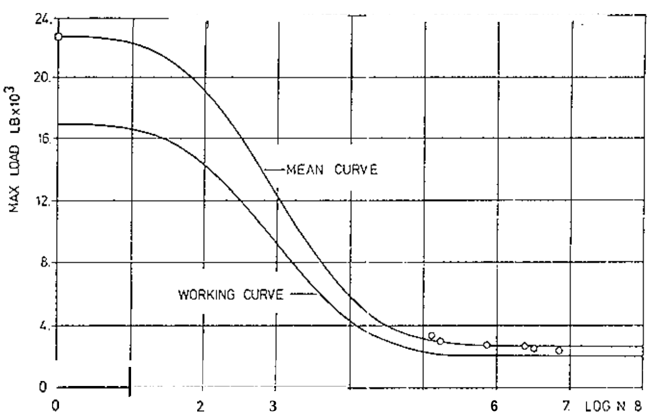

Figure 13. Mean and Working S-N Curves of a Hypothetical Pitch Link

On the basis of measured fatigue loadings and calculated working S-N curves, the analytical qualification of fatigue critical structures is state of the art in the helicopter industry.

The rotor hub is the central structure of both main and tail rotors, and plays a fundamental role in the helicopter dynamic system. It supports the rotor blades at rest and in rotation, transmits the drive torque to blades, as well as the control inputs in terms of blade ptich change, and transmits to the fuselaae the blade lift loads and rotor moments generated by the rotating blades. High capacity laminated elastomeric bearings, in combination with composite materials, are used to fulfill these primary functions in modern helicoper design, such as MBB's FEL rotor, a rigid rotor with elastomeric bearings (Figure 14). In the FEL rotor, two-elastomeric bearings, one radial and one axial, perform the function of a feathering hinge, whereas the hinges for flapwise and lead-lag motion are replaced by the elastic behaviour of the blade root.

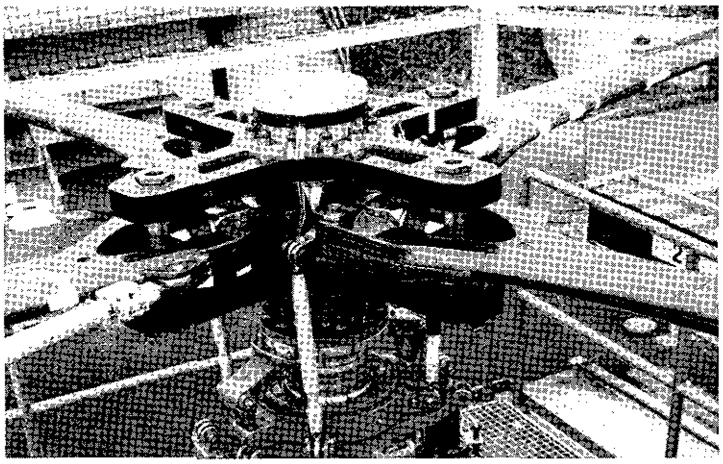

Figure 14. MBB's FEL Rotor

Due to their low polymerised chain molecules, elastomeric materials are able to undergo large but reversible deformations. This capability, in combination with the nearly incompressible material behaviour is used in designing elastomeric bearings. In the special case of a low stiffness requirement around the feathering axis, together with a high stiffness requirement in the other directions, the elastomeric bearing is constructed from alternating rubber and metal layers (shims). The demands of stiffness and strength may be accomodated by the designer using shims in the form of disks, cylindrical, conical, or spherical shells. In designing elastomeric bearings, questions with regard to stiffness, static and fatigue strength as well as damping arise for different loading and environmental conditions.

The capability for analyzing components constructed of elastomeric materials, which can be idealized using nearly incompressible hyperelastic and viscoelastic material properties, is a feature in a number of standard finite element programmes. At MBB the MARC finite element code is used to realize the nonlinear analysis [5].

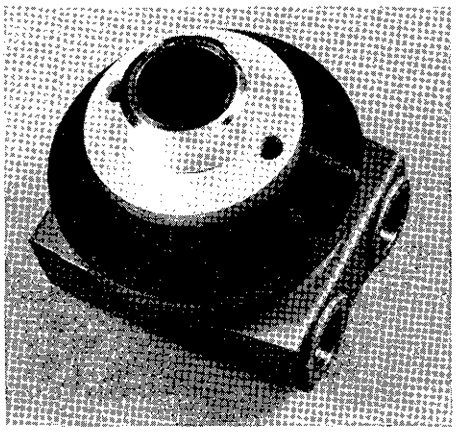

Figure 15. Axial Elastomeric Bearing

The axial bearing, as shown in Figure 15, has to support

- axial compression, caused by centrifugal forces and
- radial shear, caused by drag and rotor thrust,

while yielding

- torsional shear, due to pitch motion and
- cocking shear, due to flap and lead-lag motion.

Axial bearings are generally built up of several conical and/or spherical shells, consisting of alternating elastomeric and metallic layers attached to inner and outer support members.

For numerical analysis a finite element model, as shown in Figure 16, was used, with major emphasis placed on a realistic representation of the elastomeric layers and the shims.

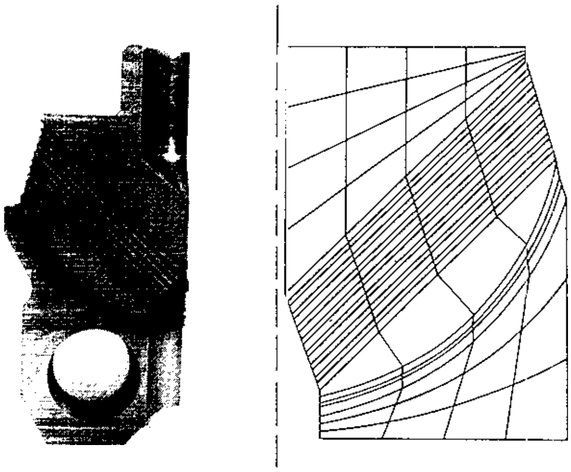

Figure 16. Finite Element Model of Axial Elastomeric Bearing

The finite element analysis was checked by establishing the stiffness of the bearing in various modes and comparing it to the test results. A fairly good agreement was found, as shown in Table 1.

Table 1. Stiffnesses—analysis vs test

Stiffness / Error	Axial	Radial	Torsional	Cocking
$\frac{C_{TEST} - C_{FEM}}{C_{TEST}}$	− 5.8%	+9.6%	+5.7%	+4.2%

Strain gauges were installed in hoop direction at different positions on the outer end of conical and spherical shims as well as on the housing. The measured strains are compared to the numerical results for various loading conditions in Figure 17. The test results and the finite element results are found to be in good agreement.

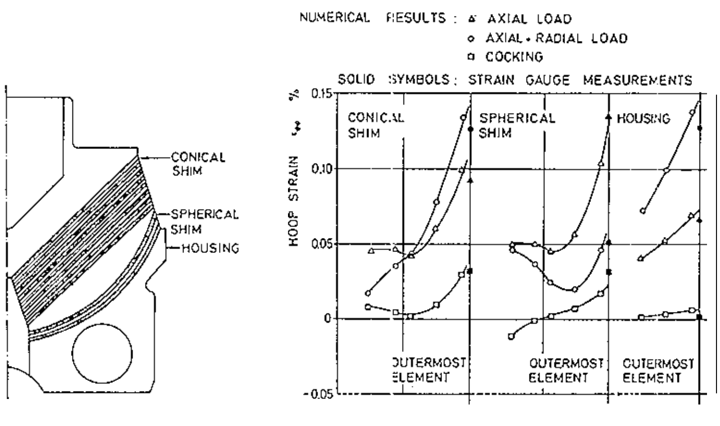

Figure 17. Hoop Strains-Analysis vs Test

From the above it can be deduced that for the analytical qualification of elastomeric components there are finite element codes available which can be used to analyze the nonlinear behaviour of rubber-like structures applying near incompressible and/or viscoelastic material properties. The investigations can be summarized as follows:

- the stiffnesses of laminated elastomeric bearings in various modes can be calculated satisfactorily,
- stresses and strains in the shims and in the rubber layers can be analyzed with sufficient accuracy,
- more detailed information can be provided than can be measured on an actual component, e.g., local internal stresses and strains, internal energy and temperature distribution, as well as influence of assumed damages on internal stress and temperature distribution.

3. CRASHWORTHINESS QUALIFICATION

Since the potential for accidents, especially of military helicopters, is great due to their mission and the environment in which they must accomplish that mission, it is imperative that they be engineered to minimize damage and enhance occupant survival in crashes. This requires that the various parts of the helicopter be designed so that the occupant will not be exposed to incapacitating injury prior to or after the airframe expends all of its required energy absorption capacity, The decelerative forces on the occupants and large masses must be reduced by energy absorbing systems, a habitable space must be maintained around the occupants, and postcrash hazards must be avoided. Crashworthiness design thus requires the management of the crash energy, as shown in Figure 18, primarily through:

- stroking of the landing gear,
- crushing of the fuselage subfloor structure, and
- stroking of the seats.

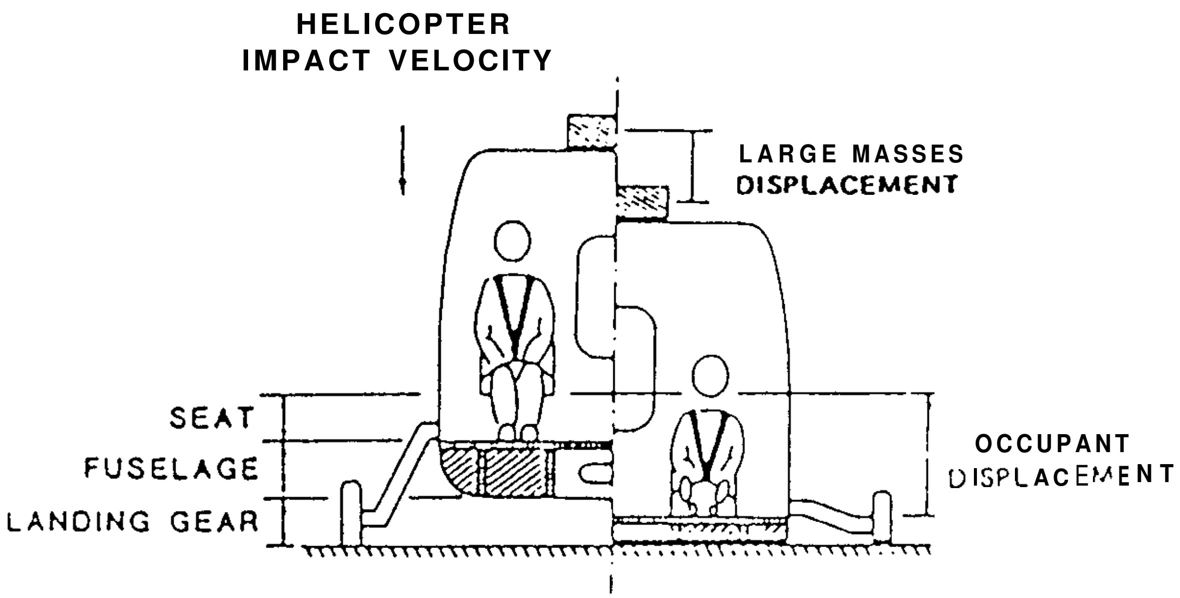

Figure 18. Crash Energy Management System

According to crashworthiness standards, e.g., MIL-STD-1290, there are several sets of crash conditions that must be investigated in support of the design process. In terms of fidelity, the dynamic testing of full-scale structures most closely approximates actual crash conditions, especially if velocity components and impacted surface conditions can be realistically represented. However, full-scale testing would be extremely expensive, but more significant than cost is the fact that only one test parameter data set per test article is available.

As a result of expanding computer capability, a number of digital computer programmes for analysis of helicopter structures in a crash environment have been developed. These mathematical crash simulation computer codes can provide a means of qualifying helicopter structures with respect to a set of crashworthiness criteria. The computer programmes vary widely in their modeling characterization and mathematical treatment of the model equations.

Crash impact simulation, i.e., predicting of the structural behaviour of a helicopter and the decelerations to which the occupants are subjected in a crash environment, must include extensive plastic deformations, large deflections and rotations, and the ability to handle nonlinear boundary conditions required by variable contact/rebound.

At MBB, we have implemented the computer programme KRASH, well-known in the helicopter community, which predicts the structural response of a helicopter to multidirectional crash environments. The programme solves the coupled EULER equations of motion for n interconnected lumoed masses. each allowed six dearees of freedom. The interconnecting structural elements represent the stiffness characteristics, both linear and nonlinear, of the structure between the masses and must be defined by user input data.

3.1 Subcomponent Crushing

The nonlinear stiffness behaviour of interconnecting beams in crash impact simulations are frequently found directly by tests. But due to cheaper computing power the nonlinear properties of subcomponents will more and more be derived from separate refined finite element analysis, taking into account effects, such as section distortion, shell folding, and rivet popping [6].

For an aluminium-alloy sheet-stringer concept, in cooperation with Engineering System International (ESI), a mathematical model has been developed with shell elements and special rivet elements and ESI conducted a nonlinear dynamic analysis using the explicit finite element code PAM-CRASH,
The sheet-stringer structure has a vertical plane of symmetry, therefore only one symmetric half was considered in the finite element model, as shown in Figure 19.

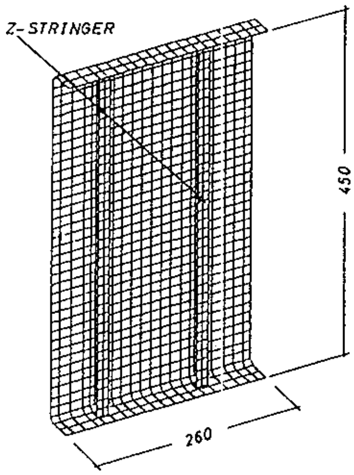

Figure 19. Finite Element Model of Sheet-Stringer Panel

Loading of the finite element model was applied by two moving rigid walls at top and bottom of the structure. Each wall travelled at 5 m/s axially toward the structure, thus producing a 10 m/s crushing velocity. The first 15 milliseconds of the crush were simulated. This corresponded to a 150 mm crushing of the 450 mm high structure.

The following results have been presented by ESI:

- load versus deformation diagrams,
- absorbed energy versus deformation diagrams,
- sequence of deformed shape plots (Figure 20),
- force versus time diagrams for rivets in the no-failure-condition, and
- time of rivet failure, when failed within the simulation time.

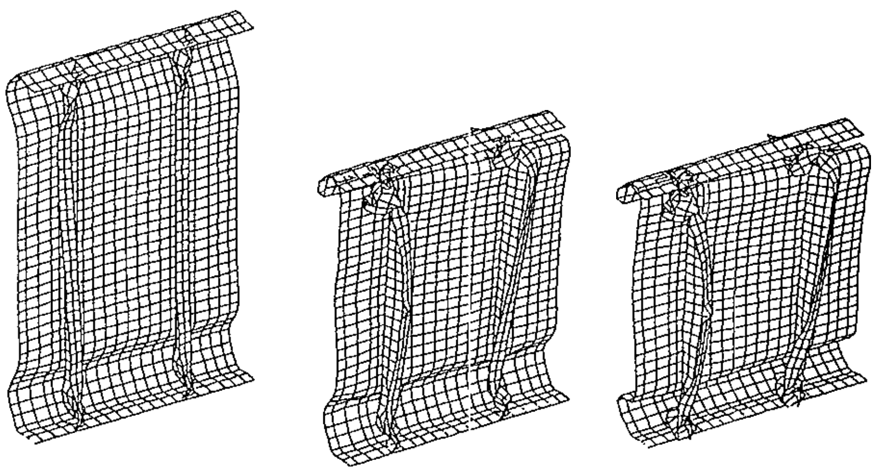

Figure 20. Deformed Shape Plots of Sheet-Stringer Panel

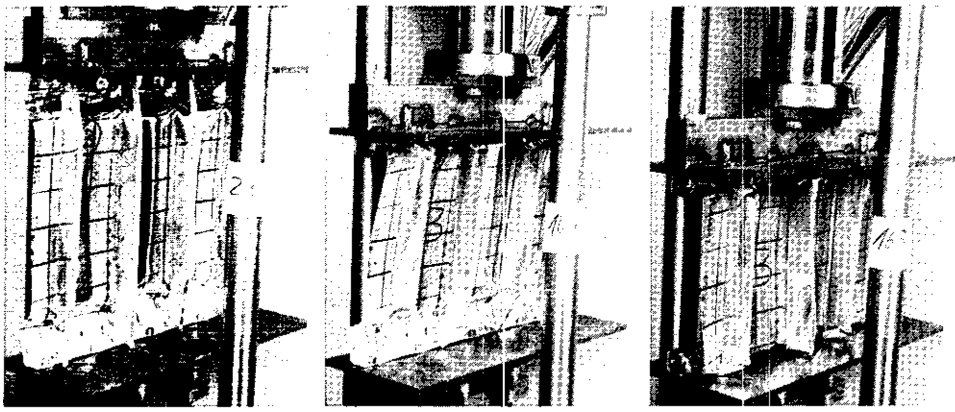

Figure 21. Deformed Shape of Sheet-Strirger Panel in Quasi-Static Crushing

Quasi-static crushing tests have been conducted at MBB and dynamic impact crushing tests at DLR, Stuttgart, to compare the analytical results with experrmental findings [7].

From the experience gained with analytical simulation of subcomponent crushing, using finite element programmes, and from comparison with test results, the following can be concluded:

- the sequence of deformation can be predicted fairly well (Figures 20 and 21),
- the load-deformation characteristic is predicted with sufficient accuracy (Figure 22),
- the total energy absorbed at 150 mm stroke is very well predicted,
- the available computer codes can be a viable tool to simulate realistically structural detail behaviour to establish input data for the programme KRASH, and
- the amount of experiments required can.be reduced at reasonable computer time.

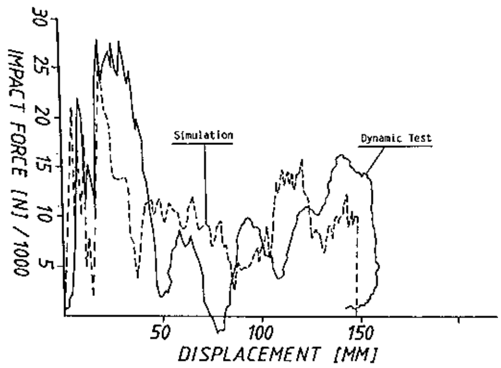

Figure 22. Crushing Characteristic of Sheet-Stringer Panel

3.2 Crash Impact Simulation of a Complete Helicopter

Once, the nonlinear behaviour of the inerconnecting structural elements, required as input data for the programme KRASH, is established either by test or analysis, the response of the complete helicopter structure to crash conditions can be evaluated.
The computer programme KRASH has the following capabilities:

- determination of the time histories of
  - o mass accelerations, velocities, and displacements,
  - o beam loads and deformations;
- provisions for load-dependent properties, as
  - o nonlinear stiffness,
  - o permanent deformation,
  - o load-limiting devices;
- determination of failure behaviour, as
  - o failure mode,
  - o time of failure,
  - o load redistribution;
- provisions for dynamic effects, as
  - o mass penetration into an occupiable volume,
  - o ground contact by external structure,
  - o internal damping,
  - o sliding friction, and
  - o rebound.

From working with KRASH we found that it enables the representation of a helicopter structure by a relatively small number of beams, which facilitates data evaluation and result interpretation. The calculation of a measure of occupant injury potential (Dynamic Response Index), by modeling occupant/seat systems, allows the evaluation of the probability of spinal injury. The programme KRASH can be used for studies of structural design parameters and energy dissipation in subassemblies for two- and three-dimensional geometry and motion (Figure 23).

The programme KRASH can be regarded as an acceptable means for the qualification of helicopter structures with respect to a set of crashworthiness requirements.

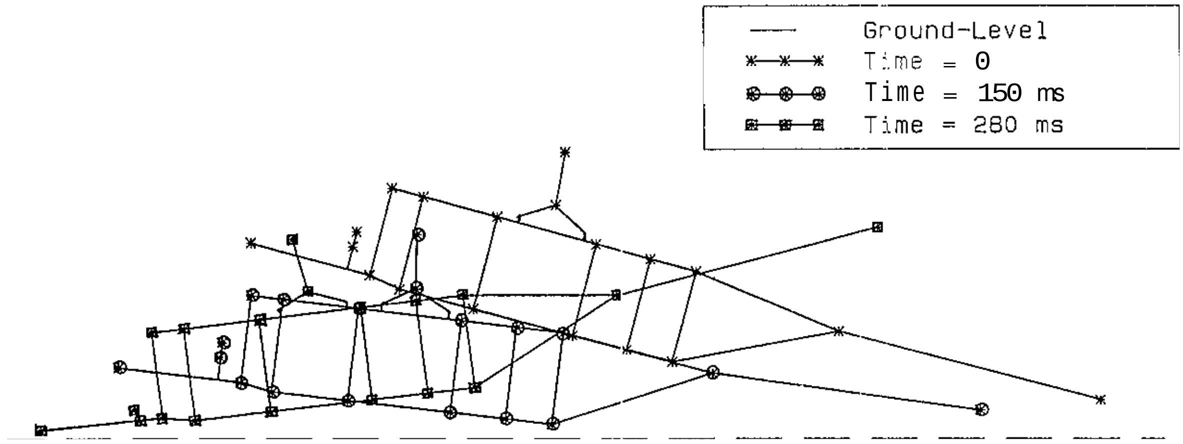

Figure 23. KRASH Mass Point Position Plot

4. CONCLUSIONS

With the currently available analytical tools, mainly the finite element method, the cost of the structural qualification programme can be significantly reduced.

Starting from preliminary design, analytical methods are applied with great success in the design phase for structural optimization of airframe structures. Based on these investigations and supported by developmental tests on coupon or component level the final qualification for all relevant load cases can be achieved with sufficient confidence, even under consideration of environinental conditions.

Complex dynamic components, designed primarily to provide adequate fatigue strength, can be checked whether fatiaue testina is necessary. An analytical determination of the endurance limit is acceotable when the fatigue margin of safety is high, or when the service loads are clearly defined and the'primary mode of failure is evident and the precise stress distribution is known. In the other case, the application of the Miner's Rule is state of the art in the helicopter industry, For the analytical qualification of fatigue critical structures.

During the last decade, considerable progress has been made in nonlinear finite element methods. The analysis of high capacity elastomeric bearings or elastomeric dampers which usually undergo large deformation, exhibit a nonlinear stress-strain relationship and have a nearly incompressible material behaviour, can now be carried out by engineers without greater difficulties. Energy dissipation and temperature distribution, as significant properties, both for bearings and dampers can be taken into consideration for analytical qualification.

Large deformations and a nonlinear stress-strain relationship are also found in subcomponent crushing for energy absorption. Nonlinear load-deformation characteristics can be predicted with sufficient accuracy for the interconnecting structural elements of a lumped mass model to qualify helicopter structures with respect to a set of crashworthiness requirements.

Pre- and post-processing have advanced in such a way that the geometry may be transferred directly from a computer aided design system, a reasonable mesh pattern can be generated automatically, and the output results may be arranged in plots or graphs for easier interpretation.

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Solar Array Qualification through Qualified Analysis

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1 Abstract

-What is a solar array and why is its qualification through analysis relevant to you?

-Why and haw did Fokker Space and Systems perform this qualification through analysis?

-Recommendations for those who want to accept the challenge of qualification through analysis.

2 The problem

To achieve qualification is in general a very expensive exercise. For solar arrays this is done by a dedicated test program through which final qualification is achieved. Due to severe competition on the solar array market, cheaper means are looked for to achieve a qualified product for our customers. One of the methods is to drastically limit the environmental test program and to qualify the solar-array structure against its environmental loads by analysis.

3 What is a solar array?

Solar arrays transfer the light of the sun into electrical energy.
It forms the primary power source of gcostatianary communications spacecraft.

These spacecraft come in two versions:

spin stabilised (cylindrical solar arrays)

three axes stabilised (planar solar arrays)

We will concentrate on the latter and more specific on rigid panel solar arrays. Fokker Space and Systems build and deliver a family of solar arrays called Advanced Rigid Arrays (ARA).

During the launch phase these solar array panels are stowed, folded like a concertina, exposing only the outboard panels to generate power for house keeping of the spacecraft. Figure 1 shows a solar array in the stowed configuration. In this phase, the stowed solar array will experience high structural loading, as we will explain later.

When the spacecraft is in its final orbit, these stowed wings unfold "gently" and will provide the spacecraft with sufficient power to fulfil its mission. Figure 2 shows the deployment phase of such an array up to the fully deployed configuration (figure 3).

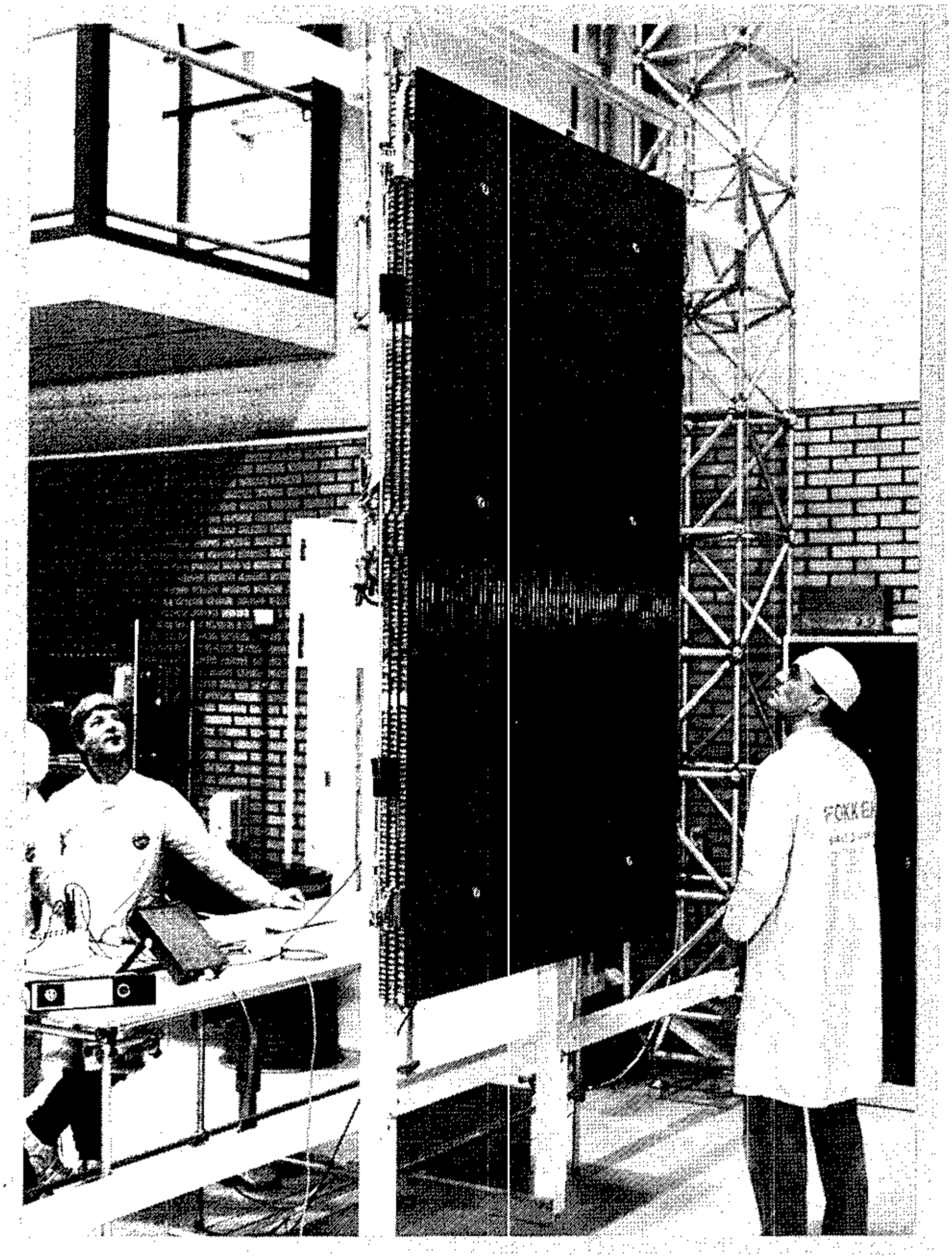

Fig. 1. A solar array in stowed configuration (ref. ARA-Solar arrays; Fokker Space and Systems).

4 Environmental loads in solar arrays

In the stowed configuration, the rigid panels are fixed to the spacecraft on a discrete number of support points. In this configuration, the solar array will be exposed to:

- quasi-static loading (from the launch)
- low-frequency (0-100 Hz) loading (transients of engine cutoff, flexibility of launcher/spacecraft)
- high-frequency (100-8000 **Hz)** loading (acoustic noise from engines and aerodynamic noise)
- thermal loading due to aerothermal heating after fairingjettison

The solar array contractor is responsible for proving that the solar array will survive these loadcases with sufficient margins.

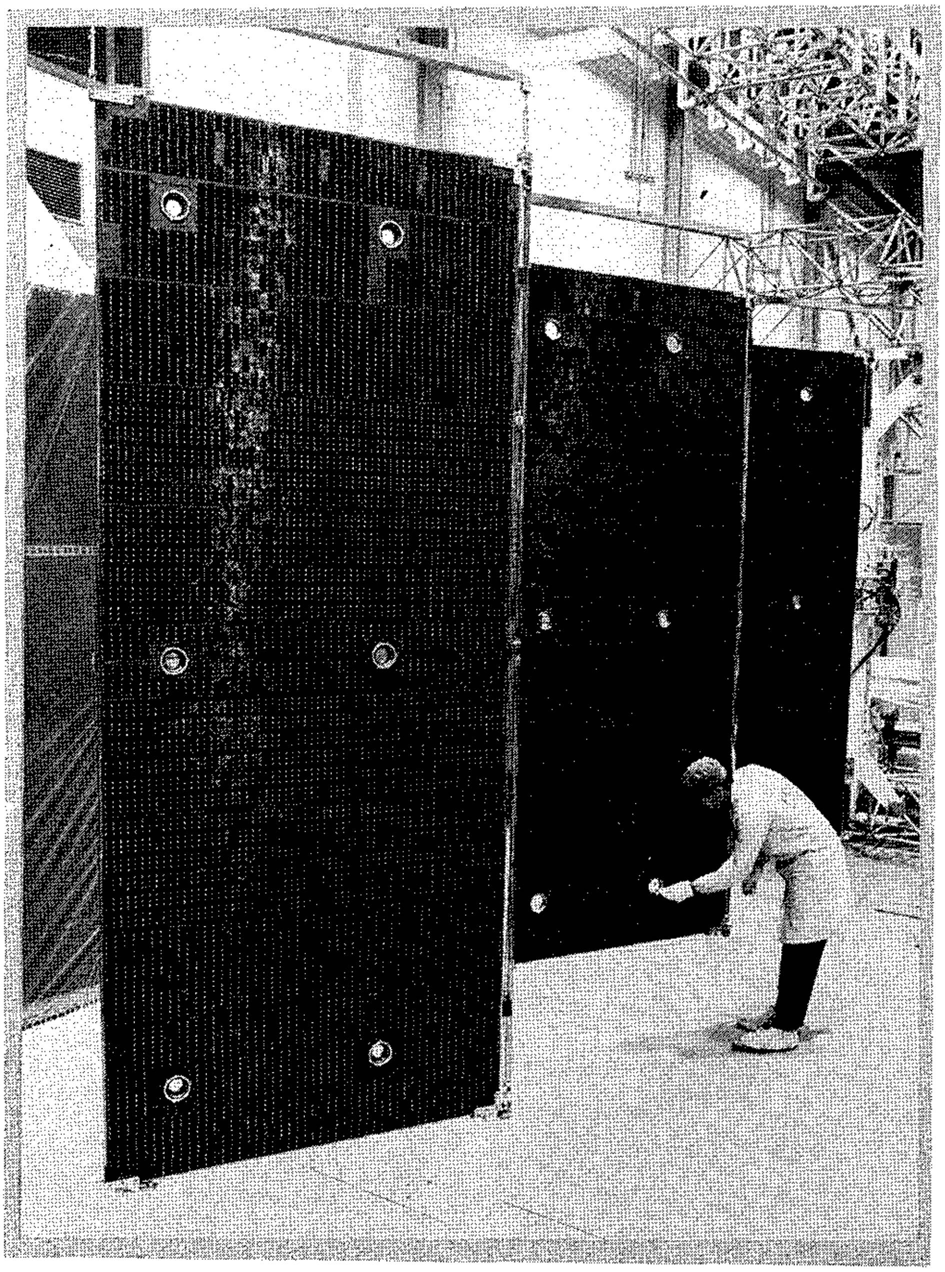

Fig. **2.** Solar array in deployment phase **(ref.** EURECA-Solar arrays; Fokker Space and Systems).

5 The problem in perspective again

To understand why Fokker Space and Systems have chosen for "qualification by analysis" of their Advanced Rigid Array (ARA), wherever possible, there are two extra topics you need to know:

1. The **ARA** is not a specific solar array for a specific spacecraft, but a generic solar array (see figure 4). It is a family of solar arrays, to suit individual customer requirements with, each time, an optimum design as it would be very expensive to qualify all possible variants by testing.

2. The time between award of a contract for a solar array subsystem and delivery of the qualified product is usually less than two years, while the throughput time for manufacturing and integration is approximately 1.5 years! This implies that you have to freeze your design and verify it in about half a year. You can only meet that time dead-line in an approach of preprocessing, analysis and postprocessing, for what concerns, the structural qualification against the mechanical environment, and not by testing a structural model.

The following aspects must bring the solar array into perspective for you:

- Rigid lightweight structures
- Quasi-static and dynamic loading
- Semi-rigid interface
- A customer requiring a qualification product, and a verified finite element model (FEM), to allow coupled analysis with the spacecraft structure.

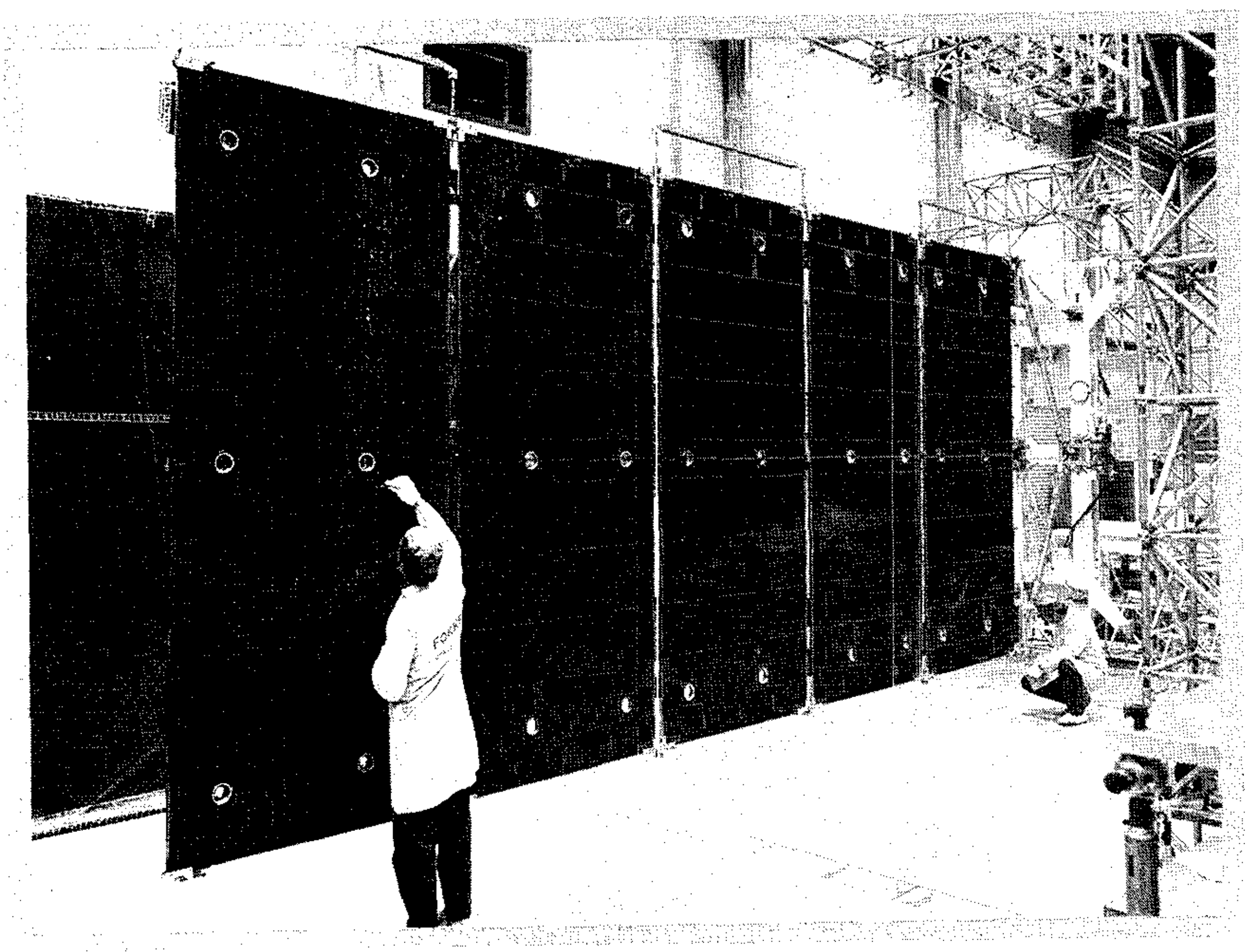

Fig. 3. Solar array in fully deployed configuration (ref. EURECA-Solar arrays; Fokker Space and Systems).

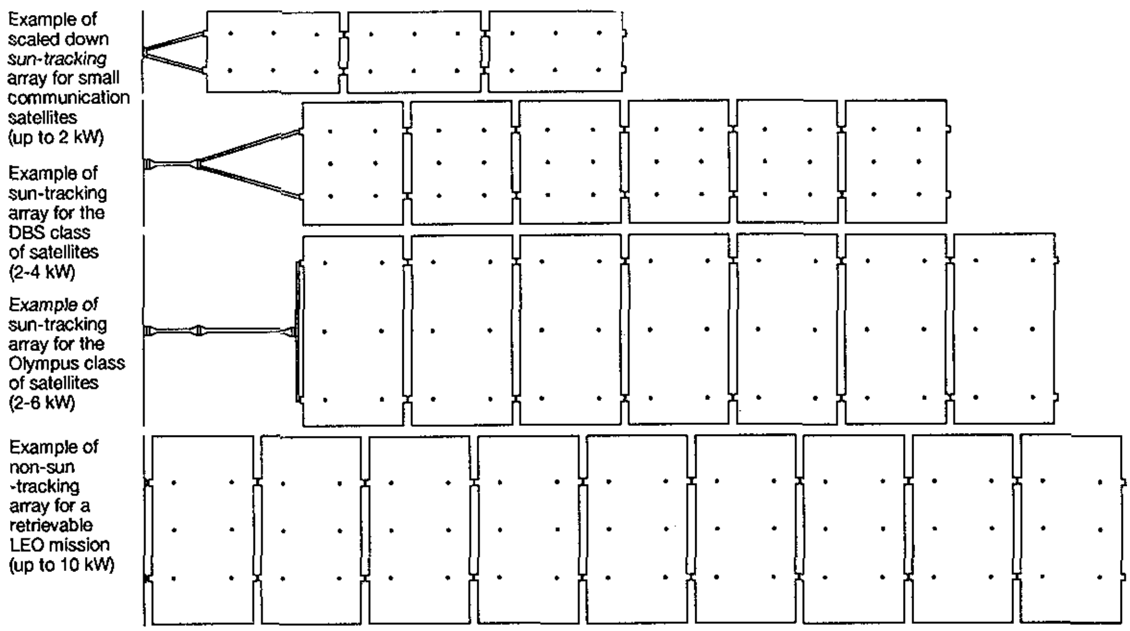

fig. 4 Family of generic design Advanced Rigid Array (ARA) showing various configurations

6 Qualification by qualified analysis

You will understand that before a customer accepts the approach of "qualification by qualified analysis", he will need to understand the procedures. This can be a time-consuming process.

The approach, which Fokker Space and Systems adapted was:

6.1 -to design and build some typical wing configurations

6.2 -to have their allowable dynamic input spectra calculated by our pre-processor /finite element analysis (FEA)/ post-processor combination

6.3 -to test these configurations to the analysed allowable limits

6.4 -if all configurations survive the test, you have qualified your analysis. You have then demonstrated that your analysis-tool can be readily used for other solar arrays from the ARA-family. Furthermore vibration-tests could now be deleted on request. In theory this approach certainly works. In practice there were some refinements to be made.

6.1 Selected wing configurations for ARA development program

From the sales aspect one would like to have some configurations with favourable interfaces in order to demonstrate high performance. From the viewpoint of the qualification of the analyses capabilities one would prefer some worst-case configurations, with the least favourable interfaces, to demonstrate the range of the analysis and to cover worst-case environment. In order to be cost efficient, one would like the typical wing configurations to be built out of the same components. Within the ARA program, we have vibration tested three configurations :

- A three panel configuration without further appendages (3 panel clean wing, see figure 5).
- A three panel configuration with a yoke structure (connection between solar panels and spacecraft in deployed position) and a solar sail (3 panels complete wing). In stowed position the I/M (intermediate) hinge was connected to the shaker.
- A five panel configuration with a yoke structure, a solar sail and some thermal hardware, normally used for spacecraft thermal housekeeping (5 panels complete wing). In stowed position the I/M hinge was supported by the inboard panel.

All panels had the same interface points to *six* holddowns to keep the array in stowed position and transfer the launch loads. These holddowns were located very close to the positions that yield the highest natural frequencies.

6.2 Calculation of allowable input level

Normally analysis tools are aimed at calculating the margins of safety for a given structure under given loads. One will start with conservative calculations and refine the calculations when low or negative margins are found. This approach cannot just be inverted to find allowable dynamic input level. One must start with expensive refined analysis from the beginning. Much emphasis was therefore laid on the analysis methods:

A preprocessor to generate the finite element model (FEM) of the solar array

A postprocessor based on hand-stress calculations for the holddownpoints

An enforced displacement method (EDM) postprocessor (ref. 4) tu calculate stresses in the panel, because the dynamic FEM model was too coarse.

A postprocessor to determine the dynamic interpanel gap (tu predict panel-touching)

This effort yielded the theoretical allowable input levels, as a function of the frequency, for a given constant modal damping value. This preprocessing and postprocessing was not only meant to gain time, but also to take out the "human factor" in the FEA. With this human factor, we mean the freedom of the analyst in modelling a structure.

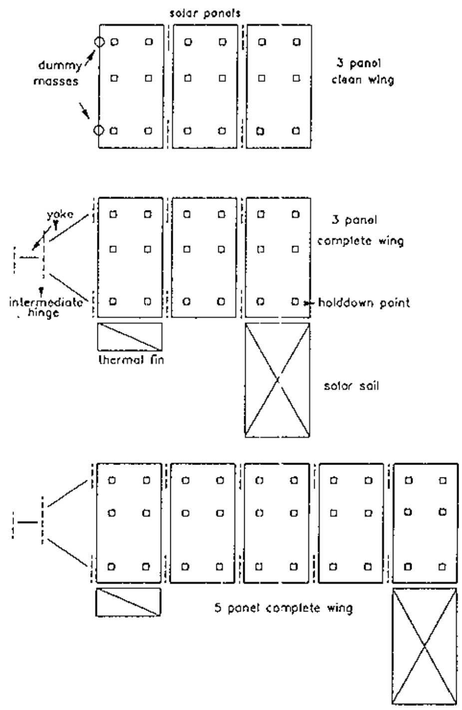

fig. 5 Solar array configurations for the ARA vibration test, shown in deployed configuration

6.3 The test philosophy

The theoretical allowable input level, as a function of the frequency and modal damping will have sharp notches at the natural frequencies. The location of these notches is not very accurately known, if one realises that natural frequencies could vary by at least 5 percent as a function of build tolerances and FEM errors.

The shape of the notches will depend on the damping that, as a function of mode shapes and load levels, can vary by a factor of 2.

So that even if the vibration control equipment was capable of accurately controlling the input level to the capricious curves, predicted by the analyst, this would be an extremely dangerous test philosophy. By having a slight deviation in frequency and damping, the actual applied "notch" in the input, could miss the actual natural frequency. This would result in serious overtesting. This was already discovered during the development testing.

"Notching" is a well-known procedure in aerospace dynamic-testing. The purpose is often to prevent overtesting of the testobject and to obtain a more realistic input spectrum in accordance with the spacecraft environment. The input to the test-object is reduced from a certain constant level in a specific frequencyband in order to achieve a constant response output on a selected measuring channel. The input level will shows a sharp "notch. If during the vibration control procedure this "notching" is applied around a resonance frequency of the test-object, the "notch" of this input level will have the shape of the (inverted) response-peak of the test-object at the natural frequency (see figure 6).

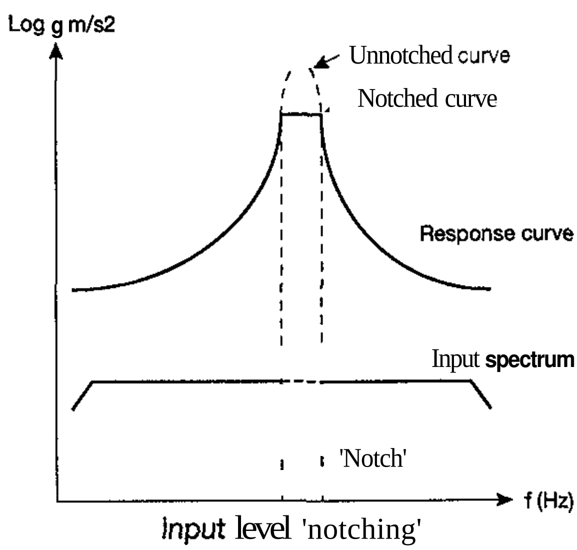

fig. 6 "Notching" phenomena (schematic)

It was found essential that the vibration testing takes place with active "notching", based on the straingauges and accelerometers, and that the correlation takes place in retrospect. This is shown schematically in figure 7.

All testing of the different wing configurations was performed at the Estec facilities in Noordwijk Figure 8 shows a general overview of the 3 panel clean wing on the dual shakerhead.

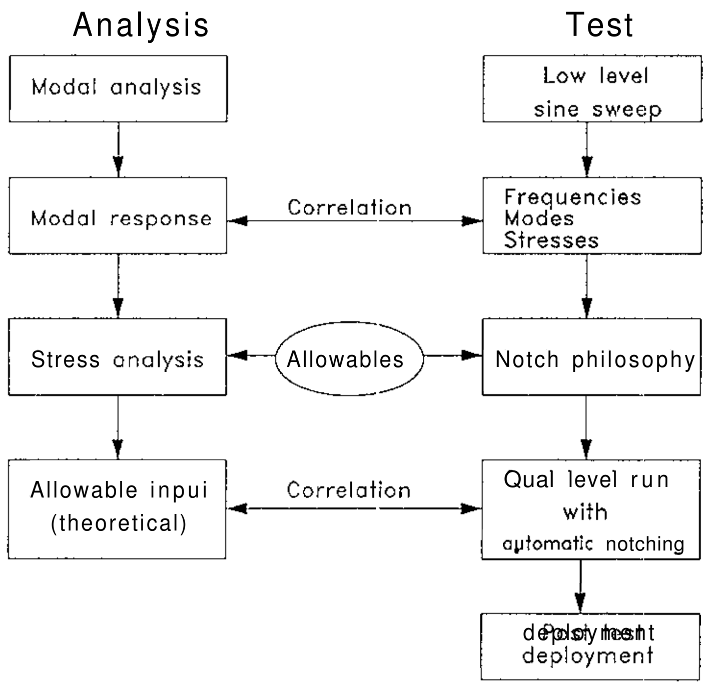

fig. *7* ARA Vibration Test Approach (ref. **1)**

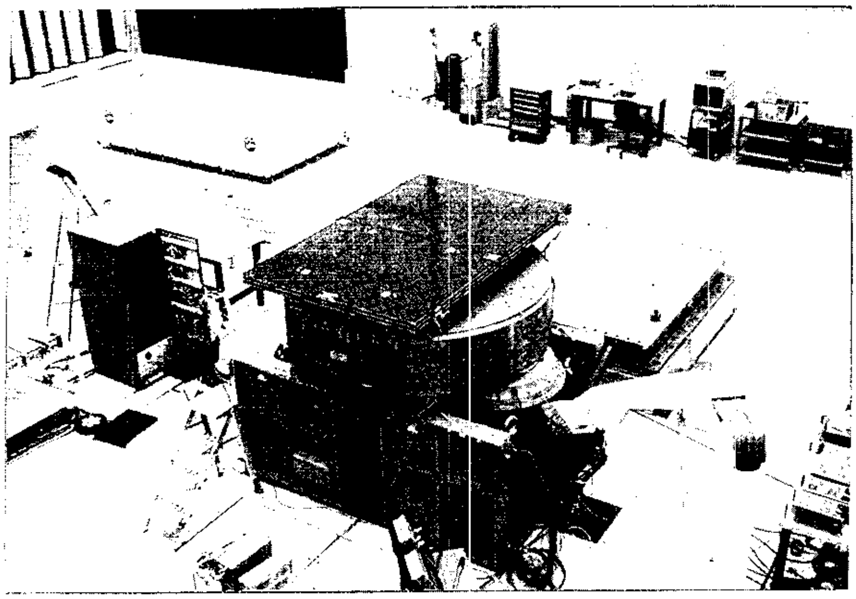

fig. *8* General **overview** of the 3 panel clean wing on the dual shakerhead.

7 The correlation: Test versus Analysis

As survival of the hardware was assured by a good testphilosophy (again requiring extensive analysis effort, even in between tests) the analysis qualification no longer follows directly from the survival of the hardware. Therefore a correlation betweenpre-test analysisand the test results remains necessary. For this, several methods were followed

7.1 Correlation of the Model (analysis input):

The dynamic FEM was verified with aid of modal surveytest results (Modal survey at Estec, University of Kassel and Fokker Space & Systems ref. 2). All of these have the disadvantage that they verify the finite element method and the vibration modes, but not the responses. Modal survey testing can be performed in several ways. For the ARA-lest program three methods were used on the 3 panel "clean" wing:

Method **1:**
Random input from the shaker on to which the stowed wing was mounted. With the responses of the accelerometers that were there on the wing for thevibrating testing, the Estec facility was able to reconstruct some major modes. With the base excitation, perpendicular to the panels, some 5 modes were found between 0 and 150 Hz. Correlation between these modes and the MSC/NASTRAN finite element model, or between these modes and the results of other modal survey methods was poor (ref. 2).

Method 2
The same as method 1, only now the input was a very slow sine sweep *(U2* octave/minute) at low level (0.3 **g).** The data were processed by a team at the Kassel University, headed by prof. Link. With this method 8 modes were found between 0 and 150 Hz. Especially on the major 2 modes, that **in** further testingpractically determined the allowable input level, quite reasonable correlation was found between the results of this modal testing and the MSC/NASTRAN finite element model (ref. 2).

Method 3
At Fokker the same hardware model was placed on a seismicblock. Modal survey testing using a single point excitation was carried out in the frequency range from 15 to 150 Hz. The modal parameters were extracted with LMS/GMAP software on a HP9826 desktop computer. The eight most reliable modes were selected. The first two modes' correlated very well with the analysis results (ref. 2)

7.2 Correlation of the Responses (analysis output):

The analytically derived modal responses, their resonance frequencies and associated vibration modes were compared with those, as derived during low level testing **(As** only the low level runs have a constant input level, these are the only ones that allow an easy correlation). The first step in this activity, the determination of response frequencies and the lowest Margin of Safety at that frequency, is already performed as a necessary part of the vibration control procedure.

An easy method to determine the correlation of the important modes is described in ref. **3.** This approach is a kind of orthogonality check not taking into account *the* phase of the modes. From the test acceleration plots as function of the frequency the following properties of the main modes are taken:

Frequency of the excited modes; they *are* recognized by *the* peaks in the graphs of the resonant plots.

Imaginary part of the acceleration to obtain the amplification factor.

For each mode the predicted and measured amplification factor of each measurement point must be collected. They form two vectors, $\{a\}$ and $\{t\}$.

To correlate the measured and the calculated modes the following is defmed ,ref. 5:

$$D = \frac{[\{t\} - C\{a\}]^T [\{t\} - C\{a\}]}{\{t\}^T \{t\}} \tag{1}$$

with C = the correction factor and D = difference factor

The analytical mode shape vector $\{a\}$ is scaled linearly with the correction factor C to get a best fit with the testvector $\{t\}$. Physically this means a correction factor of C^{-1} on the damping value assumed in the analysis. The optimum value for the correction factor is:

$$C = \frac{\{t\}^T \{a\}}{\{a\}^T \{a\}} \tag{2}$$

(C = 1 in case of perfect correlation)

Using the optimum value of C the "difference" is:

$$D = 1 - \frac{(\{t\}^T \{a\})^2}{(\{t\}^T \{t\})(\{a\}^T \{a\})} \tag{3}$$

(D = 0 in case of perfect correlation)

This approach applies only for those modes that will be excited in a vibration test, i.e. the modes with high effective mass , because they show clear peaks in the frequency response plots.

7.3 The results

For all ARA test configurations this method 2 in combination with the analytical correlation was used to correlate the test results to the predictions. The results for the main modes, the yoke and panel modes, showed good correlation. The difference :actor $D \leq 0.1$, meaning a fairly correct prediction of the mode shapes. The modal damping of 5 % as used in the analysis appeared to be rather good for most modes (ARA). In some cases, however, the main panel mode appeared to have a lower damping **(3.4%)**. The results of all these modal survey testing and the correlation exercises must be seen in the following perspective:

1) The number of modes below 150 Hz, predicted by the finite element analysis was 30. Many of these modes, were known to have little effective mass, and were not expected to be found other than by very detailed modal survey testing, ref. 6. These modes are in general not important.

2) The number of response frequencies that needed active control during the qual-level test (up to 100 Hz) was **5.** This is exactly the number of modes found by both the Kassel University method and the LMS method in the applicable frequency range and the frequencies corresponded very well. A thorough evaluation of the low level sine test results is a very good first step in correlation between test and analysis results.

8 Recommendations

In order to follow the approach of "qualification by analysis" as a possible method of reducing the development effort for ARA type solar arrays, the following points are of importance:

1. If you want to qualify by analysis, study hard, and talk to your customer, to determine what he will accept as qualification of the analysis. Work it out in detail and take out the influence of the analyst by strict procedures and pre- and postprocessing.

2. If you determine your hardware examples for the qualification of your analysis, you must realize that you are doing your stress analysis the wrong way around. Optimization criteria may turn out false. Good can be bad, and bad can be good, if you want to qualify your analysis capabilities for the whole range of applicability

3. Do not test your hardware to the limits, predicted by your analysis, before you have qualified this analysis. Make sure that you protect your hardware by proper notching and control procedures. Our experience was that test facilities can give good advice here. Notching on straingauges **is** easier than on accelerometers, as the allowable strain is not frequency dependent in contrast with the allowable accelerations.

4. Compare the calculated allowable input to the actual test input determined by active notching in the shaker control loop. Because of **all** the uncertainties and inaccuracies it is not likely that this correlation will show a perfect match, but at least it can demonstrate whether the predictions were sufficiently conservative to compensate for the inaccuracies.

5. Qualify your analysis for a conservative damping value. Mass and stiffness are quite well predictable. Damping values can easily **vary** by a factor 2. Determine the damping by development tests.

6. In your correlation exercise, consider responses rather than normal modes. Responses will eventually determine the loads, *the* loads will determine the adequacy of your design.

9 Conclusion

Qualification by analysis is possible. The benefits are that a significant amount of development effort can be saved in case such a powerful tool is available. Extensive testing can be avoided thus saving time and money. For successful use the above mentioned recommendations should be adhered to.

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Probability Approach for Strength Calculations

by

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SUMMARY

The use of probabilistic structural analysis methods (PSAM) to predict structural reliability is the subject of an on-going NASA research program. The paper reports on the elements of the new technology developed to date. Applications of the developed software to structural problems are demonstrated for simple validations problems and for large scale application problems. Future. on-going research to support component and system reliability predictions suitable for analytical certification of aerospace structures is then briefly reviewed.

LIST OF SYMBOLS

Symbol	Description
B	strain operator
C_{ij}	covariance matrix
d	perturbation operator
D	elasticity operator
E	Young's modulus; elastic constant
$f_{\underline{x}}$	probability density function of x
g	limit state function
k	r_o/r_i
K	stiffness matrix
ln	natural log function
p_f	probability of failure
P	applied forces
r	cylinder radius
R	residual nodal forces
S	bending stress
T_i	internal temperature
T_o	external temperature
u	displacements
v	Poisson's ratio; elastic constant
$\underline{x}$	random variables
X_i	design variables
Z	response function
α	coefficient of thermal expansion; sensitivity factor
δ	tip displacement
ϕ_i	normalized mode shapes of independent random variable fields
σ	stresses
σ_i	variance of i-th variable
σ_θ	hoop stress

1. OVERVIEW OF PROBABILISTIC STRUCTURAL ANALYSIS

Program Goals

The Probabilistic Structural Analysis Methods (PSAM) project funded by NASA has the central goal of developing a comprehensive structural analysis system capable of considering general forms of uncertainty in loading, geometry, material behavior. and boundary conditions. The purpose of this analysis system is to support the design of advanced space propulsion hardware capable of operating in severe environments with adequate reliability. The PSAM methods are sufficiently general, however, that they can be applied to a very wide range of structural reliability concerns.

The PSAM project is being completed in two phases. The first phase is complete and has focused on the prediction of the uncertainty in structural response variables such as stress, natural frequency, forced and random vibration response. buckling load, etc. Such predictions of the probabilistic distributions of "stress"-types of variables are just a part of the prediction of structural reliability, albeit an important and previously-missing component of the complete problem. The second phase is underway and concerns the probability that the "stress' variables will exceed some structural 'resistance' variable, such as strength, fatigue life, or crack growth limit. This second phase must then also be concerned with the interactions of multiple and progressive damage processes that affect structural "resistance."

The ability to analyze complex structural response problems and predict the reliability of the structural component. based on uncertainties in design and manufacturing variables, is an essential ingredient in being able to certify aerospace structures based on analytical (or numerical) modeling. The designer concerned with reliability needs to be able to identify those key variables which most affect the component or system reliability, and to be able to quantify the relationship between design variable uncertainty and component or system reliability. The work to date on PSAM demonstrates the analytical means for satisfying this need.

The current paper primarily reports on the work done to date on the first phase of the PSAM effort. The status of design approaches to analytical prediction of component and system reliability for aerospace structures is then briefly reviewed in the final section of this paper. The completion of future PSAM tasks will provide the tools needed for analytical certification, when coupled properly with critical component and system testing. The methodology will be reported in future progress reports on this NASA-funded effort.

NESSUS Coda Structure

Figure 1 highlights the major modules currently used by the NESSUS (numerical evaluation of stochastic structures under stress) software system, illustrated for the finite element method (FEM). These modules include an expert system which facilitates the user interface [1], a preprocessor for random data fields (NESSUS/PRE), a structural modeling module (e.g., NESSUS/FEM) [2], a database for intermediate analysis results, and a module for making fast probability calculations (NESSUS/FPI) [3].

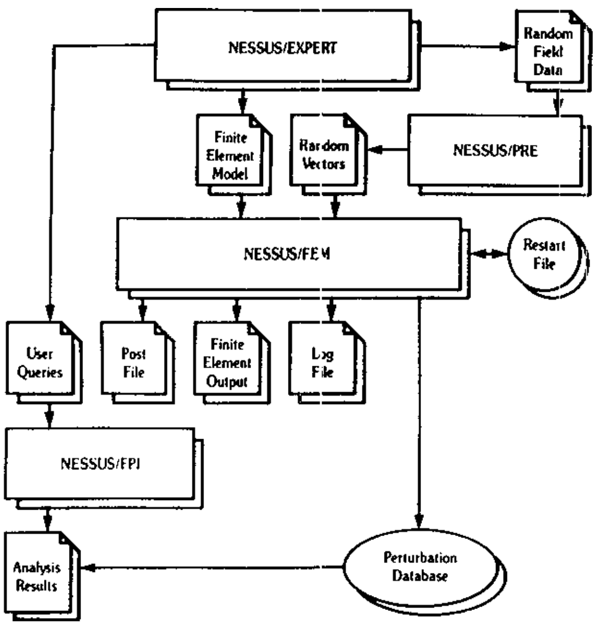

Figure 1. Overview of the NESSUS Code for PSAM

Probabilistic analysis requires that each random variable must be an independent random variable. For random fields, some correlation of the variables is expected from point to point. Random fields include descriptions of the probabilistic pressure or temperature loads, probabilistic thickness variations, or probabilistic material prcperties within the structure. As an example, consider aerodynamic loading. Aerodynamic pressure loading is likely to have a high degree of statistical correlation from one region to another, owing to large-scale order in variables such as velocity and pressure. On the other hand, turbulent boundary layers generally have statistical aspects that may only be locally correlated, related to the size and distribution of the turbulent eddies.

The decomposition of random fields with partial spatial correlations into independent random variable fields is presented in [4]. The approach is to represent each field in terms of a modal decomposition

$$[C_{ij}] = \sum_{i=1}^{N} \{\phi_i\}\lambda_i\{\phi_i\}^T \tag{1}$$

where the $\{\phi_i\}$ are normalized mode shapes of independent random variable fields. The algorithm was originally developed on the basis of normally distributed random variables, but has been extended to approximate non-normally distributed variables. NESSUS/PRE performs the normalized modal decomposition for random fields, based on a user-specified degree of isotropic correlation over the field. More general representations are theoretically allowable using this same approach.

Analysis Algorithns

Figure 2 represents the key element of the PSAM algorithms. The algorithm is represented, for purposes of example, in terms of frequency dependence on material modulus. The random variable (nodulus) is described empirically or in terms of standard cumulative distribution functions. The structural sensitivity of the frequency to changes in modulus is estimated by a numerical model, experimental data, or an approximate, closed-form solution. The cumulative distribution function (CDF) for the response natural frequency is directly computed by the fast probability integration (FPI) algorithm.

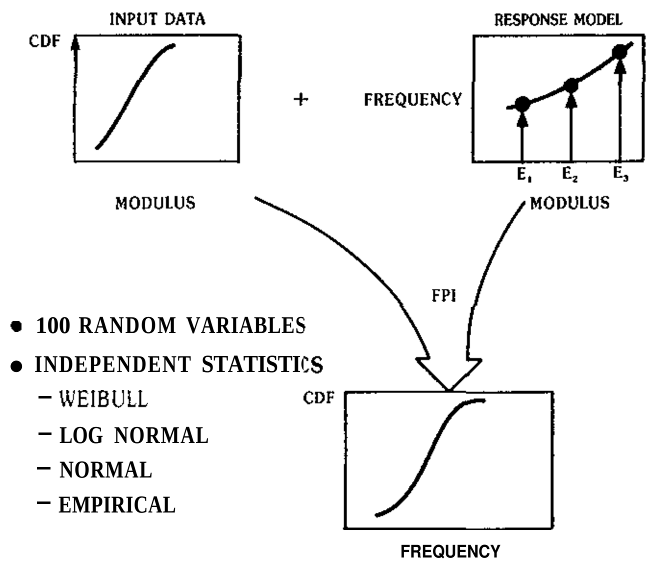

Figure 2. Fast Probability Integration (FPI) Algorithm Uses Structural Sensitivity Data

The FPI algorithmwas developed [5] for basic reliabilitycalculations for explicit, closed-form system models. In the PSAM project. FPI has been adapted very effectively to implicit response models through numerical analysis of complex structures [6].

In general, the response model is a numerical analysis model (e.g., FEM) of the structures subject to the usual deterministic loading and boundary conditions. For linear analysis, we are generally concerned with such variables as the maximum stress or the natural frequencies of fundamental modes. Sensitivity analysis seeks to estimate how much the stress or natural frequency depends on each of the design variables (loads, geometry, material properties. boundary conditions). By perturbing the random design variables a small amount, the solution variable(s) are perturbed. A hyperplane or hyper-(quadratic)surface is then fitted to the solution data at the mean-value (MV) design point and a suitable number of perturbed-solution points, and the results stored in a database for later FPI analysis.

The NESSUS code has specially tailored algorithms for obtaining the perturbed solutions for static and dynamic problems, as described in the reference by Dias and Nagtegaal [2]. In general terms. the structural analysis routines maintain the full element description of the unperturbed problem, and place the effects of perturbations on the right-hand side of the system equations as pseudo-forces. for the FEM case. Iteration of the system equatfons is based on the use of the reduced unperturbed solution stiffness equation, as an iterative preconditioner matrix. This approach is summarized in the equations below, and results in a significant time savings for static problems over the full resolution at the perturbed condition. A smart sub-space iteration method is used to obtain efficient perturbed eigenvalue solutions for dynamic response and linear buckling problems. Typical perturbation solutions are achieved in roughly 60% to 70% of the solution time for a full generation and resolution of the system equations.

The following equation summarizes the static solution algorithm for perturbed variables. The quantities with hats are perturbed from their Initial state. The perturbed $\{\hat{\sigma}^i\}$ values and $[\hat{B}]$ matrix are used to compute a residual vector $\{R^i\}$ for the revised load vector. Perturbed displacement and stress terms are then obtained by iteration, using the unperturbed $[K]^{-1}$ matrix as a preconditioning matrix.

$$
\begin{aligned}
&\{u\} = [K]^{-1}\{P\} \\
&\{\sigma\} = [D][B]\{u\} \\
&(i^{TH} STEP)\{\cdot\} + \Delta\{\cdot\} = \{\hat{\cdot}\} \\
&\{\hat{u}^i\} \\
&\{\hat{\sigma}^i\} = [\hat{D}][\hat{B}]\{\hat{u}^i\} \\
&\{R^i\} = \{\hat{P}\} - \int_v [\hat{B}]^T\{\hat{\sigma}^i\}dv \\
&\{d\hat{u}^{i+1}\} = [K]^{-1}\{R^i\} \\
&\{\hat{u}^{i+1}\} = \{\hat{u}^i\} + \{d\hat{u}^i\}
\end{aligned} \tag{2}
$$

Probabilistic analysis of the structural problem requires the prediction of the probability that the structural response exceeds some level or limit, as a function of uncertainty in all of the random design variables. Such a problem may be written in terms OF the joint probability function of the random variables, as shown in Figure 3. below.

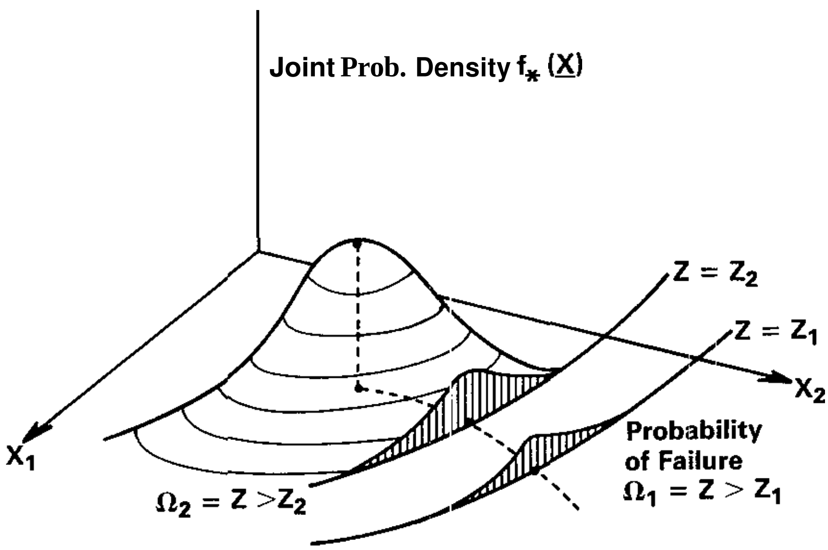

Figure 3: Integration of Joint Probability Function

The Probability of exceedance (i.e., $Z > Z_2$) is then written as the solution of a multiple integral written as

$$P(Z > Z_2) = \int_{\Omega_2} f_{\underline{X}}(\underline{x})d\underline{x} \tag{3}$$

Here $Z(\underline{X})$ is a response function such as stress or displacement, Z_2 is a limit value of the random variable Z. $\underline{X} = (X_1, X_2, \ldots, X_n)$ is the vector of basic design factors, and $f_{\underline{X}}(\underline{x})$ is the joint probability density function of $\underline{X}$ limit. The X_i are the individual random variables input to the problem (loads, dimensions. material properties, etc.). each with its own statistical distribution. Equation (3) describes the cumulative distribution function (CDF) of the response function as Z_2 is varied over the appropriate range. In structural reliability, the integral would be carried out over the failure region Ω, the design space in which some "performance function' or "limit state function" $g(\underline{X})$ satisfies the condition $g(\underline{X})<0$. Solution of that integral determines the probability of failure p_f.

$$g(\underline{X})=Z_2-Z \tag{4}$$

Exact solution of this multiple integral is. in general, extremely complicated. An alternative is the Monte Carlo method, but this is expensive relative to its accuracy, especially for engineering problems for which low p_f values are required. As a result, current approaches for determining structural reliability are based on analytical approximation methods, including first- and second-order reliability methods (FORM and SORM) [7].

The PSAM code employs a new fast probability integration (FPI) algorithm, an extension of the first-order reliability method (FORM) of Rackwitz-Fiessler [8] and Chen-Lind [9], which has been proposed and demonstrated by Wu and Wirsching [10] to consistently provide fast and accurate estimates of point probabilities for typical engineering response functions. This algorithm establishes quadratic polynomial approximations to the limft state at the most-probable-point (MPP) and then transforms the quadratic form into a linear one. The MPP at $Z=Z_2$ is given by the maximum value of the joint-PDF along the limit defined by $Z=Z_2$, as indicated in the cross-section depicted in Figure 3. An optimization routine is employed to approximate non-normal variates as equivalent normals. thereby approximating the limit state as linear in normally distributed design factors. Solution of this simplified problem is referred to herein as the mean-value, first-order solution (MVFO).

Probabilistic Sensitivity Factors

One of the most important results from application of the FPI methodology is the prediction of probabilistic sensitivity or importance factors for the design. These factors can be used to identify the key design variables affecting structural reliability. Structural sensitivity has been defined above as the rate of change of a structural response variable with respect to change in a random design parameter (e.g., material stiffness, part tolerances, loads, etc). Probabilistic sensitivity includes this factor but also takes into account the uncertainty of the given structural parameter.

The details of the calculation method for these sensitivity factors (α_i) are treated in [11]. Generally speaking, the α_i are normalized such that

$$\alpha_1^2+\alpha_2^2+\cdots+\alpha_n^2=1 \tag{5}$$

The values of α_i are given approximately as the sensitivity of the solution (g) to the physical variable (X_i)

$$|\alpha_i|\propto\left(\frac{\partial g}{\partial X_i}\right)\sigma_i \tag{6}$$

where σ_i is the normal standard deviation. Thus, both the physical and the statistical variables are taken directly into account.

The utility of the probabilistic sensitfvity factors will be seen in the applications problems. Essentially. the factors represent a numerical ranking of the importance of the individual physical variables on the component (un)reliability. The higher the ranking, the more important is that variable to component (un)reliability. Weak physical variables with large uncertainties may have probabilistic sensitivity factors more important than strong physical variables with small standard deviations. The values of the factors must be reduced in order to make the component more reliable.

2. *NESSUS* APPLICATIONS

Validation Example

An example problem demonstrating the results of the application of the NESSUS software is presented below. A number of other validation problems have successfully been executed with NESSUS [6]. Other validation analyses can be found in [12].

The example validation problem herein is for a cantilever plate subjected to correlated random static loadings as shown in Figure 4. The other random variables are elastic modulus, thickness, width and a base spring stiffness as defined in Table I. The response functions considered are the bending stress. S, at the base and the tip displacement, δ.

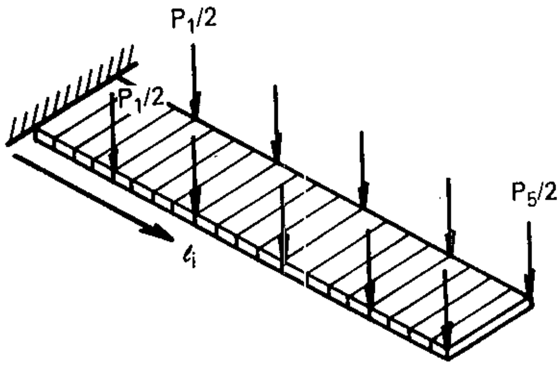

Figure 4. Cantilever Plate

Table I. Variables for Validation Problem

Variables	Distribution	Median	COV
Modulus, E	Lognormal	10E6 psi	0.03
Length, L	Lognormal	20.0 in	0.05
Thickness, t	Lognormal	0.1 in	0.05
Width, w	Lognormal	1.0 in	0.05
Base Spring, K Loads'	Lognormal	1E5 lb-in/rad	0.05
P_1 to P_5*	Normal	0.1 lb (mean)	0.1

'Note: Correlated with correlation coefficients = exp (-distance between loads / 20)

The finite element model consisted of 20 fliit-plate elements with 42 nodes. The NESSUS deterministic solution results were 0.7648 inch for the tip displacement and 3657 psi for the maximum stress. These values agreed well with theory – 0.7692 inch and 3600 psi, respectively. The differences are 0.5% for the displacement and 1.6% for the stress. For either the displacement or the stress, the probabilistic solutions were checked by selecting two points in the right tail of the distribution (i.e., cumulative probability > 50%).

The solution approach used by NESSUS, in this {example, is first to calculate the first-order (FO) sensitivity factors for root stress and tip displacements (i.e., $\partial g/\partial X_i$) evaluated at the mean-values (MV) of the design variables (X_i). NESSUS then uses FPI together with the statistical models in Table I to predict the mean-value, first-order (MVFO) probabilistic distributions shown.

At each reliability level, FPI estimates the MPP. or the values of X_i that are most probable for that reliability level. Back-substitution of these variables into the NESSUS model corrects the value of the solution state (root stress and tip displacement), assuming that the local reliability does not change. This update at the MPF is referred to as the advanced-MVFO solution (AMVFO). New sensitivity factors ($\partial g/\partial X_i$) at the MPP may then be calculated in order to update the reliability value at this solution point. This iteration step generally results in very little change in the local solution, demonstrating that the AMVFO solution is sufficiently accurate for most design purposes. This process is illustrated in Figure 5. The MVFO, the AMVFO. and the first-iteration solutions for the displacement and the stress, respectively, are shown in Figures 6 and 7. The exact solution shown in the figures was generated by applying Monte Carlo simulation (sample size = 100,000) to the theoretical solutions.

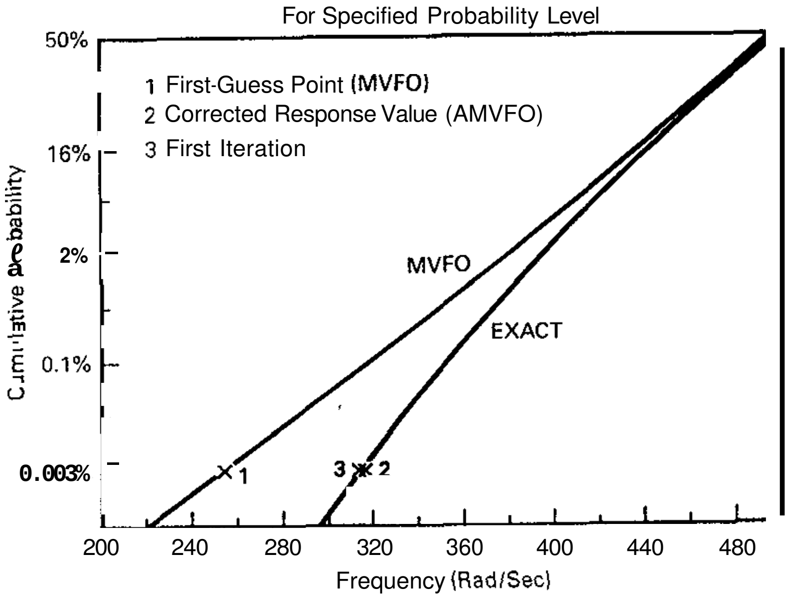

Figure 5. Iteration Algorithm

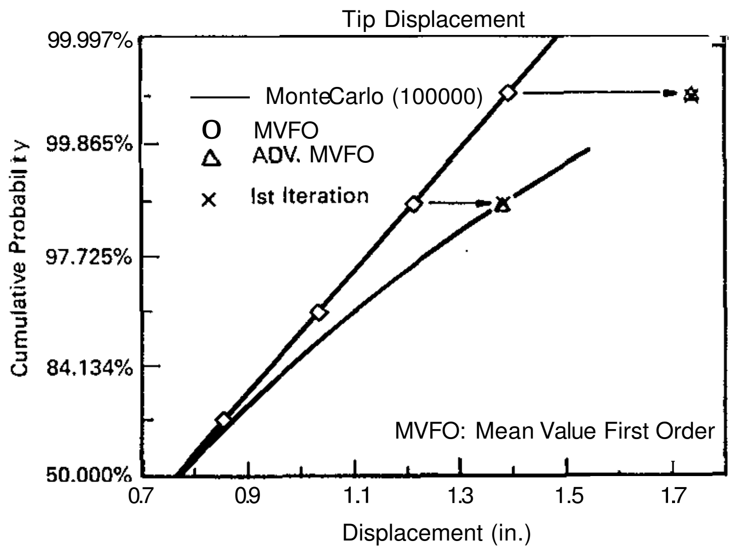

Figure 6. CDF of the Plate Tip Displacement

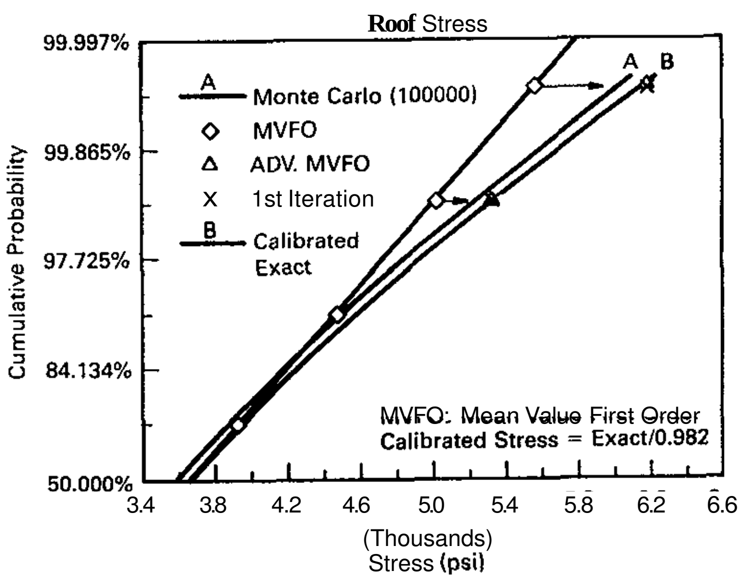

Figure 7. CDF of the Cantilever Plate Root Stress

Approximate Methods

The applicability of approximate methods for for rapid evaluation of probabilistic structural analysis is addressed by the Probabilistic Approxiriate Analysis Methods (PAAM) module in NESSUS. The basic idea of PAAM is simple: make an approximate calculation of system response, including calculation of the associated probabilistic distributions, with minimal computation time and cost, based on a simplified representationof thegeometry, loads, andmaterial. Ihe resulting deterministic solution should give a reasonable and realistic description of performance-limiting systemresponses, although some error will be inevitable. If the simple model has correctly captured the basic mechanics of the system, however. including the proper functional dependence of stress, frequency, etc.. on design parameters, then the response sensitivities calculated may be of significantly higher accuracy than the solution itself. In other words, the calculated probabilistic distribution of the response variable may be in significant error only by some offset of the mean value.

The PAAM response functions are defined in an independent user-written subroutine. To execute the code. a user needs to prepare a data set which defines the selected component, the input random variables, and the probabilistic analysis options. The output of the analysis includes the CDF values, the most-probable-points, and the sensitivlty factors. For checking purposes, the code also includes a Monte Carlo option which implement:, an Importance sampling scheme.

An example problem of a thick cylinder subjected to different interna'l and external pressures and temperatures is a classic elasticity problem; exact solutions are readily available in standard textbooks [13]. For simplicity we consider here only the hoop stress. which is typically the largest stress component. The pressure solution is given by

$$\sigma_\theta = -\frac{(r_o/r)^2(p_o-p_i)}{k^2-1} + \frac{p_i - p_o k^2}{k^2-1} \tag{7}$$

$k = r_o/r_i$, and the subscripts i and o denote Internal and external pressures or radii. The thennal solution for an appropriate steady-state temperature field is given by

$$\sigma_\theta = -\beta\frac{(r_o/r)^2+1}{k^2-1} + \beta\frac{1-\ln(r_o/r)}{\ln(k)} \tag{8}$$

where

$$\beta = \frac{\alpha E(T_i - T_o)}{2(1-\nu)} \tag{9}$$

Here E is Young's modulus, v is Poisson's ratio, and a is the coefficient of thermal expansion.

Probabilistic data values for the thick cylinder model were selected for testing and are listed in Table II . The material is Haynes 188. Note that FPI can easily accommodate a wide variety of distribution types, including truncated distributions.

Table 11. Definition of Input Variables for Thick Cylinder Model

		Mean	COV
r_i	truncated Normal (itl.003)	0.094 in.	1.06%
r	truncated Normal	0.110 in.	4.55%
E	Normal	3.40E+7 psi	2%
v	Normal	0.3594	2%
		5.65E-6/R	5%
p_i	Lognormal	3077 psi	4%
		3232 psi	4%

Both an FPI analysis and a Monte Carlo simulation (based on 10,000 samples) were performed. Results for the hoop stress at the inner radius are presented in Figure 8 as a cumulative distribution function, described in terms of standard deviations of the equivalent normal distribution of the hoop stress. These calculations suggest that the mean value of the hoop stress at the inner radius will be approximately 193 ksi, and that the hoop stress will be less than 220 ksi at a probability level of 0.99 (about +2.3 standard deviations). Actually, these stress levels are above yield, so the elastic assumptions in the formulation are invalid and a nonlinear analysis should be performed. This would require only the development of new, nonlinear equations for the hoop stress, but would not cause any difficulty for the FPI algorithm or its implementation. There is excellent agrement between the FPI and Monte Carlo calculations. The primary difference between the two is speed of execution. The traditional Monte Carlo computations for this problem require more CPU time by one to two orders of magnitude, depending on the probability level required.

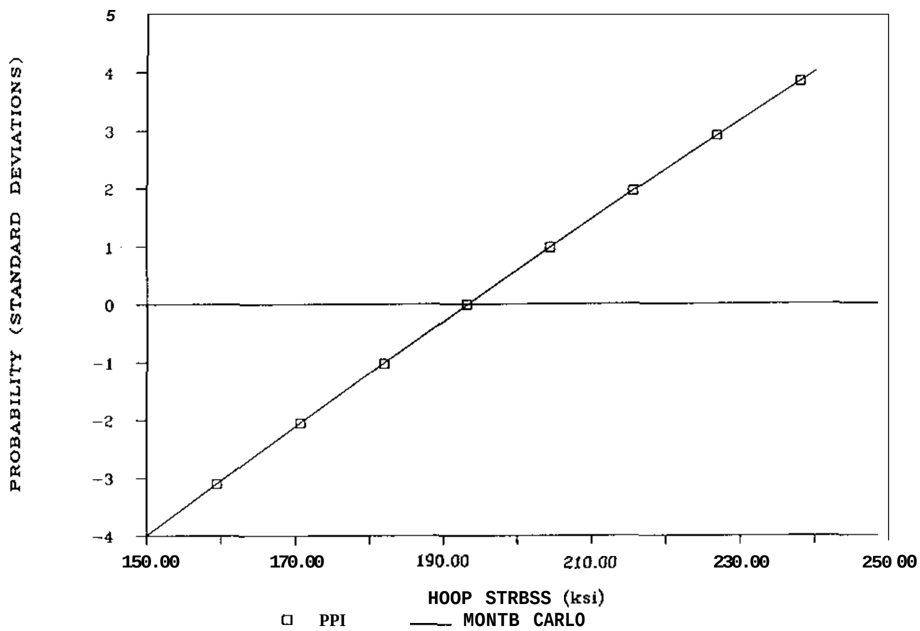

Figure *8*. Cumulative Distribution Function of the LOX Post Hoop Stress at the Inner Radius

Figure 9 shows the probabilistic sensitivity factors of the random variables for the hoop stress at the internal radius, computed here at the design point corresponding to +3 standard deviations. From these results it is seen that *a* clearly dominates, followed by the elastic constants E and v and the external temperature T_o. This is due in part to the larger COV's for these variables, which are based on default values rather than problem-specific information. It may be of particular benefit to obtain improved statistical data for these parameters. Furthermore. in view of the significant influence of these three material properties, further analysis should address the known temperature dependence of each. an effect which was neglected in this first PAAM analysis. On the other hand, the tight tolerances on the internal and external radii are shown to be sufficient to make dimensional variations of little consequence to the hoop stresses. so further work in that area is not needed.

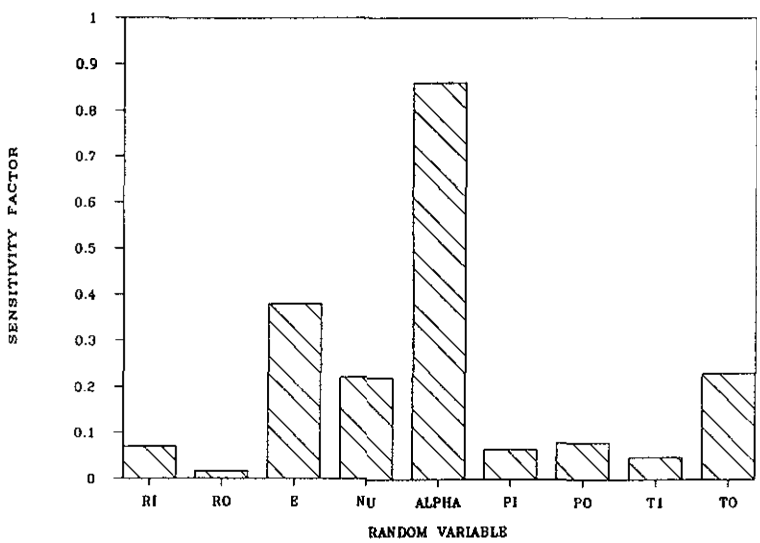

Figure 9. Sensitivity Factors for Random Variables in PAAM Analysis of LOX Post (Thick Cylinder Model, Calculation of Hoop Stress)

SSME Turbopump Blade Application

The NESSUS/FEM software was applied by the space shuttle main engine (SSME) manufacturer (Rocketdyne) to the structural response of the turbopump blade shown in Figure 10. The finite element model shown has about 6.000 degrees of freedom. The random design variables for the blade included nine manufacturing variables and nine loading variables. The manufacturing variables included three rigid-body orientations of the airfoil, associated with the process of machining used on the blade root, three orientations of the cubic single crystal being evaluated, and three material properties for the single crystal. The nine loading variables concerned operating conditions for the turbopump. each of which had some predictable effect on the rotor speed, blade temperature, and pressure loading conditions (steady-state operation was considered).

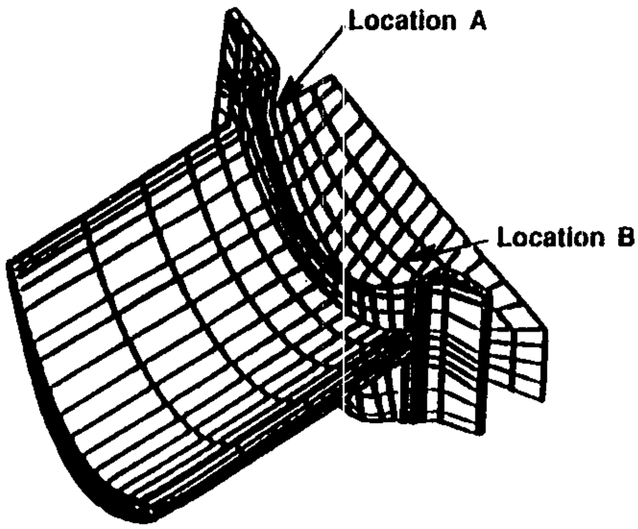

Figure 10. FEM Model of Turbopump Blade

Figure 11 illustrates the use of engine simulation to define the effect of each of the 'primitive' operating condition variables on the three loading conditions. A primitive variable is taken to be a statistically independent variable whose effect on a random loading variable must be independently calculated or estimated. In the current application, Rocketdyne correlated the results of the engine test data to define the statistics of the primitive variables, as given in Table III.

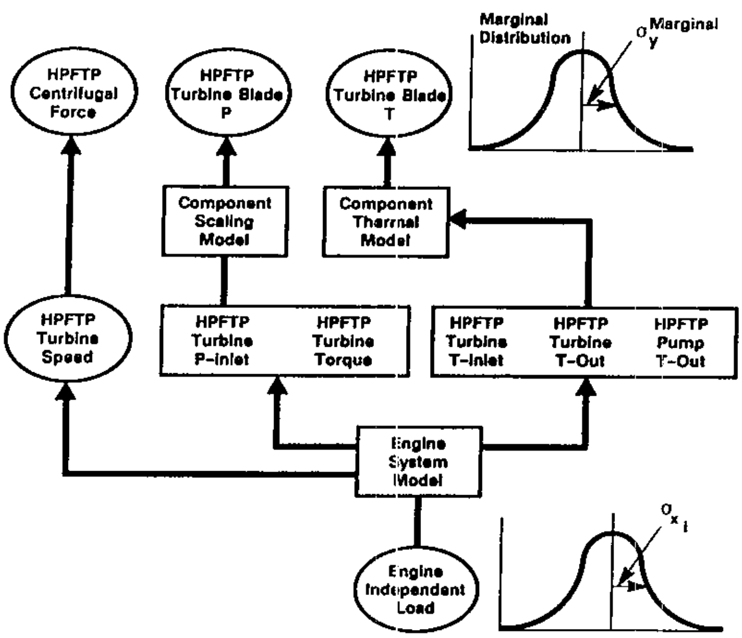

Figure 11. Engine Simulation Modeling

Table 111. Random Variables for Turbopump Blade Modeling

No.	Random Variable	Type	Affected FEM Quantities	Mean	Standard Deviation
1	Material axis Z	Material	Anisotropic	-0.087266 radian	0.087544
2	Material axis Y	orientation	material	-0.034007	0.081544
3	Material axis X	Effects	Orientation angles	-0.052380	0.067544
4	Elastic modulus	Material	Elastic constants	18.38E6 psi	0.4595E6
5	Poisson's ratio	properties		0.388	0.00065
6	Shear modulus			18.63E6 psi	0.046575E6
7	Geometric lean	Geometrical	Node coordinates	0 deg	0.14 deg
8	Geometric tilt	variations		0 deg	0.14 deg
9	Geometric twist			0 deg	0.30 deg
10	Mixture ratio	System	Pressure, temperature,	6.0	0.02
11	Fuel inlet pressure	independent	centrifugal force	30.0 psi	5.00
12	Oxidizer inlet pressure	loads		100.00 psi	26.00
13	Fuel inlet temperature			37°R	0.50
14	Oxidizer inlet temperature			164°R	1.33
15	Pump efficiency	Component	Pressure, temperature,	1.00	0.008
16	Head coefficient	independent loads	centrifugal force	1.024	0.008
17	Coolant seal leakage	Local effects	Temperature	1.0	0.10
18	Hot gas seal leakage			1.0	0.05

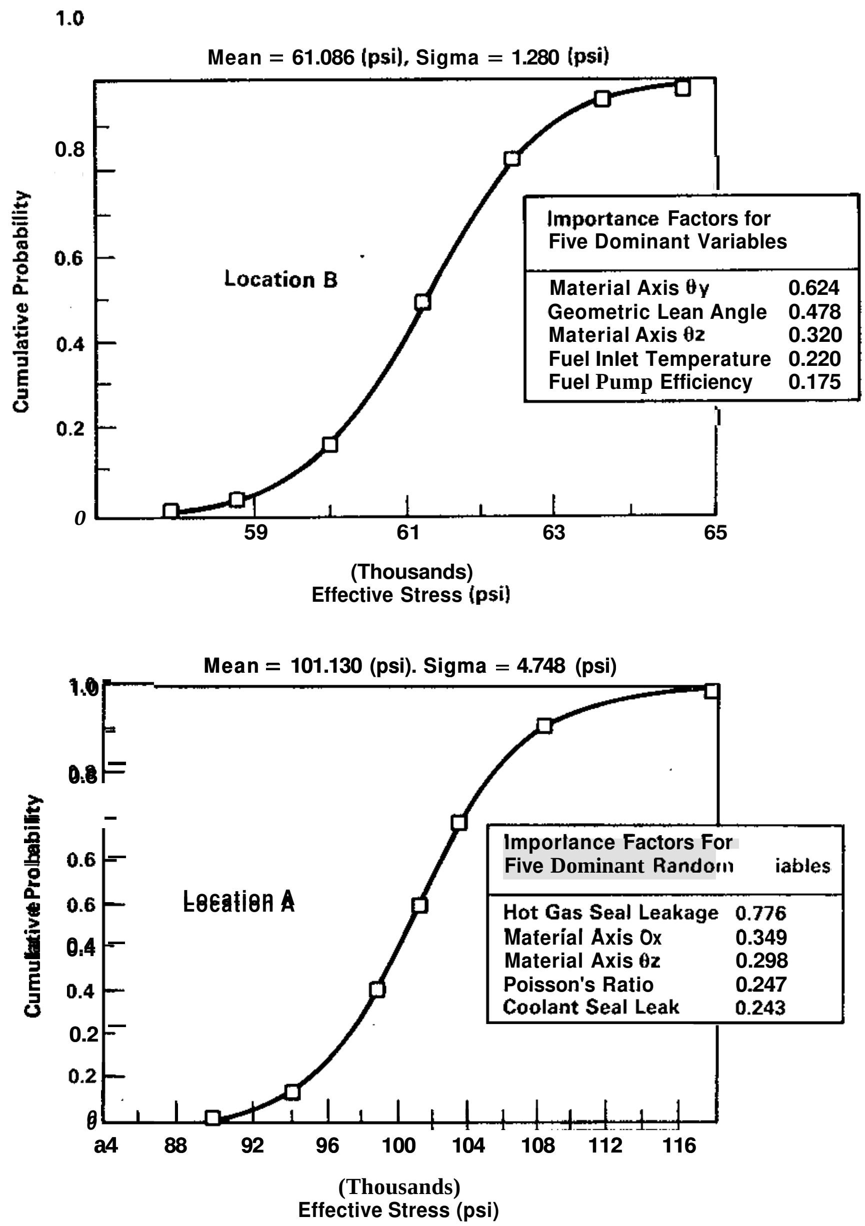

Figure 12. CDF and Sensitivity Results for Two Locations

The results of the NESSUS analysis for the turbine blade problem are plotted in Figure 12. The two parts of the figure correspond to the two limiting locations identified on the finite element model. The plots depict the CDF of the effective stress at the two locations, using the MVFO option. Of most importance in the design study are the probabilistic sensitivity or importance factors shown. At location A, the dominant factor is the hot gas seal leakage, which allows hot gas exposure of that location, resulting in the highest blade stress. At location B, two of the single crystal orientations are among the most dominant factors. These factors can be used to quantitatively assess the importance of design changes or process controls for the variables on the reliability of the design.

3. STRUCTURAL RELIABILITY

Current Approach

The current practice in design of flight structures does not include design-in reliability, except through the use of past. good experience on reliable designs. Rather, load envelopes and biased design factors for such items as weights, thermal gradients, tolerances, and material properties are used to provide a result for which a safety or life factor can be applied.

Statistical approaches to support the desigri process are used to evaluate the uncertainty in loading and material properties, for example, but not to predict the ireliability of the final design. Some selected design problems such as fracture critical engine hardware may include a probabilistic fracture mechanics model for defining design curves. Also, probabilistic risk assessment is becoming widely used in the evaluation of airworthiness directives for selected flight safety evaluations. These applications, of course, occur after the design, when premature cracking of a critical component is found in the fleet.

The design of civil structures is the only place where probabilistic methods are used more often for predicting structural reliability. Examples include the response of structures to stochastic load models due to earthquakes and sea loads. However, in these problems only the mean and the variance in the solution are usually calculated.

The results presented so far have been for tie "stress" variable distributions and not for the 'strength' variables. In order to compute the design reliability of the structure it is necessary to take the joint probability of the "stress" and "resistance" distributions for appropriate material or structural failure mode. Considerable ,research has been done to date on probabilistic "resistance" models. These models are easily integrated with the results for the "stress" variables to predict the probability of failure of the structural component. Automation of the strength evaluation is now being made in the NESSUS code.

Future Directions

The research reported to date has focused on the assessment of uncertainty in the "stress" or response variables of the design. Much work exists in the literature on the subject of stochastic life modeling, which focuses on the material "resistance" or strength and life variables. The extension of the reported research effort includes the integration of these two technologies. to be able to predict component reliability in terms of strength, life, or cost.

Component reliability in its simplest form addresses individual, independent failure modes. In reality, multiple failure modes or sites may be involved in structural failure. Structural redundancy and damage progression may also be very important. These issues bring in problems of system reliability, especially if there is some form of dependence between the damage processes. The integration of system reliability concepts into the NESSUS software is one of the challenging areas of future work.

Othermajor issues for the future include probabilistic considerations for component inspection and system health monitoring requirements. The information gained from such studies can be directly used in the assessment of the probabilistic residual strength or performance of a system, and will be developed as well in the NESSUS system.

Certification Issues

Certification has been achieved in the past by the assurance of margins of safety or of life for the structure. No suitable level of risk has yet been defined to correspond to the success of past designs. Much work needs to be done to support the use of reliability methods for flight or propulsion system design certification. The use of limited numbers of flight test aircraft or test-cell engines does not support the demonstration of component or system reliability, suitable for certification. The use of past experience is only valid if there is considerable replication of the data-base structure in the new design. New analytical methods, such as reported herein, and supported by critical experimentation, are required for certification to a defined level of reliability.

Fortunately, new tools are now becoming available for the certification task. Integration of reliability modeling methods in the design and test phases of major, new aerospace systems is critical for the future. Cost savings and performance improvements will be achieved though the detailed application of the new tools throughout this development process. Furthermore, the tools provide for the first time, quantitative assessments of the role each of the design variables is playing in the reliability of the component, and hence the system.

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RECORDER'S REPORT OF FINAL DISCUSSION

by

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During the last years the cost of structural testing has been skyrocketing due to the following reasons:

- Introduction of composite materials for primary aircraft structures and hence the possibility of complicated failure modes and the variability of these materials to temperature, moisture and impact.
- Cost of labour is very high.
- Test equipment is sophisticated and therefore costly.
- With aeroelastic tailoring a balanced design is found – therefore ideally there should be no inherited strength which makes the search for the weakest test condition very difficult since the test article collapses in all failure modes at once.

On the other side the use of finite element codes has made structural calculations more reliable and the advent of CFD codes also produces better airloads which may not be represented well enough in a test.

Therefore a rational approach to control development cost for new airplanes is to increase emphasis on analytical certification with necessary (but limited) experimental validation.

The objective of the Workshop had been to address the role of structural analysis design in relation to the qualification procedure, in order to establish guidelines for the future and to seek out those areas where there exists a commonality in approach.

The attitude of the participants, from the various groups, can be summarised as follows:

Aircraft Industry

- Reduce certification costs by eliminating nonessential development tests.
- Explore acceptability of combining major tests.
- Reduce economic risk of developing a non-viable, un-certifiable airframe structure.

Certification Agencies

- Demonstrate airworthiness by tests.

Analysts

- Analytical and computational procedures and capabilities are improving with time.
- Robust designs for structures can be assured with applicable data bases.

Experimentalists

- Some failure modes and mechanisms in composite structures are not yet well enough understood to predict reliably by analysis.
- A test is more likely to find a design problem than an analysis that does not model the problem.

The following general points were made:

1. Analysis is getting better and less expensive.
2. Analysis should complement tests to improve the qualification process and reduce testing costs.
3. Analytical models should be verified or certified by test and then used for redesign.
4. Analysis will not eliminate the need for testing for some time to come, if ever.
5. Composite structures are more sensitive to secondary effects than metallic structures and, as a result, require more detailed local analysis than is used for metallic structures to model critical failure mechanisms and to predict failure reliably.

Some common issues and concerns were identified

- Computational costs
- Software fidelity and user qualifications
- Modelling accuracy and appropriate analysis

- Material property variability and manufacturing tolerances
- Availability of appropriate loads
- Reliable damage tolerance and durability predictions
- Understanding of failure mechanisms and availability of appropriate failure analyses
- Improved analysis capabilities and full-scale testing are needed.

The following recommendations were agreed:

- Improved analytical methods are needed for predicting complex structural behaviours such as non-linear response and crack growth.
- Software should be evaluated and verified to provide quality assurance.
- Training should be provided for analysts and certifiers to assure that structural problems and analysis methods are understood.
- Modelling skills should be improved to reduce the risk of generating misleading analytical results.
- Validated reference analytical results and benchmark problems would help analysts improve their understanding and verify software.
- Open exchange of lessons learned would help others avoid repeating a design problem or mistake.
- Round-robin analysis challenges would help software developers and analysts to verify software and modelling skills.
- **A** common certification approach would help all concerned with design and certification.

It **is** hoped that the workshop — bringing together the various views of the aircraft industry and certification agencies — has served in achieving the goal of showing a way to reduce development cost and increase reliability of structures.

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The Workshop provided a very useful forum for the exchange of information which is reflected in the papers presented in this report.

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