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Aircraft Integrated Monitoring Systems

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TELEDYNE DATA ACQUISITION AND RECORDING TECHNOLOGY ON AIRBUS AIRCRAFT

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Abstract

This paper describes a graphical Human-Computer-Interface for specifying triggering conditions and event definitions that are to be automatically translated for execution in a computer.

This specification method is being developed for use in the Teledyne User programmable systems, specifically the Flight Data Interface and Management Unit (FDIMU) installed on Airbus Single Aisle and Long Range aircraft.

Introduction

The Teledyne FDIMU designed for the Airbus Single Aisle and Long Range aircraft families is an airborne unit that performs the following three functions:

- Flight Data Interface function that acquires, formats and supplies data to the Flight Data Recorder for accident / incident investigation.
- Data Management function for aircraft condition monitoring and flight operations quality assurance.
- Raw data recording for maintenance and flight operations analysis.

The data management and raw data recording needs differ from one operator to another and a single hard-coded solution can not satisfy even a small number of customers. Thus the specifics of the data management and raw data recording function are designed to be user programmable.

The Teledyne Application Generation Station (AGS) is the tool the operators use to program the Data Management and Data Recording functions of the Teledyne FDIMU.

The AGS allows the user to specify or modify many of the data processing characteristics of the airborne units, such as the format of the generated reports, the sampling rates of the desired parameters, the recording rates of the data, the format of screens presented on the cockpit displays etc.

The user of the AGS specifies all of the above characteristics along with other functional requirements of the airborne unit by entering data into forms, or composing formats of reports and screens by the use of the keyboard and the mouse of a ground based personal computer.

The aircraft condition monitoring and to some extent the recording functions are real-time functions, and as such are driven by events. Thus, in addition to the characteristics, mentioned above, that must be entered into the AGS tables, the user of the AGS must also specify the conditions that are to be monitored and the actions that are to be taken when defined events happen.

The remainder of this paper presents some of the different methods currently available for defining conditions that lead to events. In addition a new specification method and a new Computer Human Interface (CHI) is introduced for specifying such conditions and events.

Existing Systems

Currently there are two significantly different methods of specifying conditions that when satisfied define an event. One is based on a software model and the other hardware.

Consider the following simple example represented in figure 1.

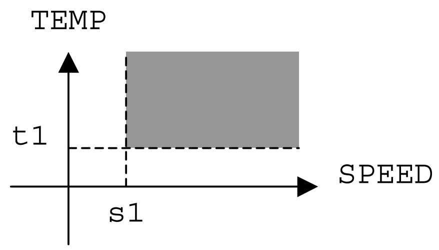

Figure 1.

The shaded region represents an “exceedence” condition where speed (SPEED parameter) is greater than s1 and at the same time temperature (TEMP parameter) is greater than t1. Suppose entering this region is an event that should cause an alarm.

In a “software oriented “ user programmable system the user might be expected to enter a statement similar to figure2.

```
IF SPEED > S1 AND TEMP > T1 THEN SOUND ALARM
```

Figure 2.

Similarly in a “hardware oriented” system the user might be expected to draw a diagram similar to figure 3, using a library of symbols and lines.

Other methods of specifying conditions, events and actions are, for most part, based on one of the above two methodologies.

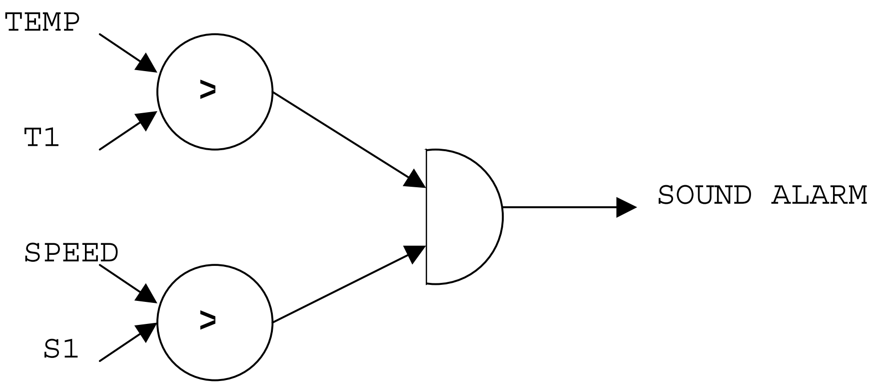

Figure 3.

An Intuitive System

Both of the specification methods, described above, depend on the user translating a physical concept as described in figure 1 to a particular implementation as presented in figures 2 or 3.

Currently a system is in development[1] at Teledyne Controls, that eliminates need for the user to describe a condition by specifying its software or hardware implementation. The system allows the user to specify the basic physical concept in a notation very similar to figure 1.

The system under development is a graphical Computer-Human Interface (CHI) that, among other functions, allows the user to define events by specifying and shading regions similar to figure 1, associating restrictions and attributes to the regions, combining regions and defining actions to be performed when the events are detected.

The system also translates the entered definitions to a format suitable for processing by an airborne computer.

Bounded and Unbounded Regions

After associating parameters to the axis of the plane where a region is to be defined, the user must assign limits to the parameters or equivalently, sizes to the region. One or two limits (upper and or lower) may be assigned to each parameter (on each axis). Figure 1 represented a region that was bound on two sides but unbound on top and right. Figure 4 shows some other variations, which are based on how the parameter limits are defined.

Figure 4a is a region unbound on the left and the bottom and might be a simple “stall” type event detection, whereas 4b is bound on all four sides, and might be part of stability detection.

[1] Some of the methods presented in this article are part of a system for which a patent application has been filed.

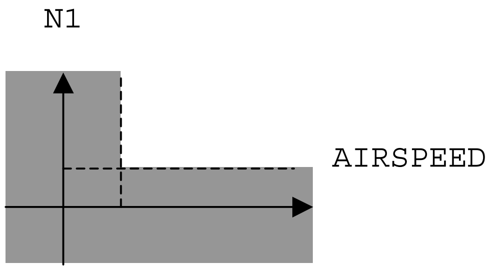

Figure 4a.

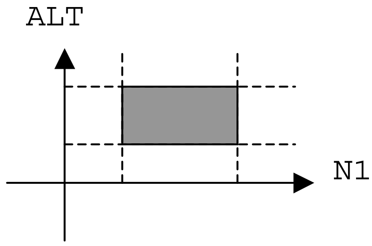

Figure 4b.

Entry and Exit Criteria

The regions described above allow the user to specify basic conditions. By adding additional restrictions on the regions, more sophisticated conditions can be defined.

One example of such restrictions is entry/exit criteria. In some cases entry into a shaded region might not necessarily be an indication of an event. The direction of entry or exit might also be relevant. For example consider the shaded region in figure 5, which imposes the restriction that an event is announced only if the region is entered from below and exited from the top.

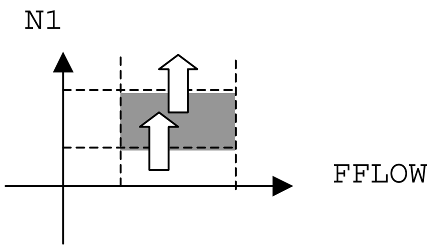

Figure 5.

The above definition would cover an engine start but exclude engine shutdown or idle.

Relative and Absolute Limits

Another variation on the definition of a region is to define one or both of the axes to represent Relative values of parameters, rather than Absolute. In the example of figure 6, an event is defined when there is an excessive altitude drop during a roll. The axis for the roll angle is defined as absolute, whereas the axis for altitude is defined as relative (note the arrowhead and the fact that the axis does not extend downwards).

In this case as the roll angle passes over the limit r1, the altitude parameter is baselined (entry from left to the region). If at any time the altitude falls below a2-a1 (exit from below) then an event is announced.

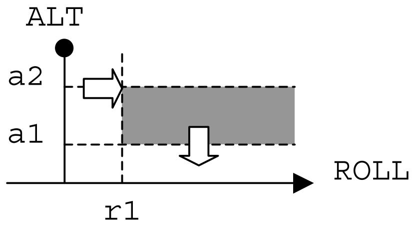

Figure 6.

Time

In airborne systems, Time, in one form or another, is always available as a parameter to be monitored. The absolute value of time, however, generally is not of interest in an event detection system, unless there are some flight restrictions at certain times of the day or one need to sound an alarm at lunch-time!

Time as a parameter on a Relative axis, on the other hand, represents "duration" and allows for the definition of regions that can be used for de-bouncing or time-delay considerations.

Figure 7 represents an event where the parameter EGT rises and remains above e1 for a duration of t1 seconds. If the parameter were to return below e1 before t1 seconds had elapsed, the exit from the region would be from the bottom, not the right, as specified and therefore the condition of the region would not have been satisfied and the event would not be announced.

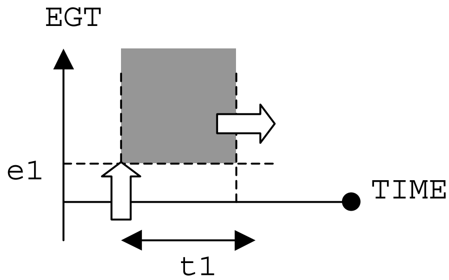

Figure 7.

Complex Conditions

While the above mentioned trigger condition specifications allow great flexibility in defining a single condition, it is often a combination of conditions that are required to trigger an event. Complex conditions can be defined as a network of logical "AND" and "OR" conditions. The Teledyne AGS implements this network as a mosaic of tiles.

Condition Tiles

A condition tile may contain a simple trigger condition, as defined above, or it may itself contain a previously defined mosaic. The latter approach provides a “drill down” method to the user and facilitates modular, reusable and manageable mosaics. A condition tile may also contain the result of an AMTAL evaluation. AMTAL is a language previously developed by Teledyne and specifically intended for trigger specifications. The AMTAL language is not detailed in this paper, but the inclusion of an AMTAL trigger provides a migration path for airlines with existing Teledyne applications.

“AND” Conditions

In the simplest form of an “AND” condition the user is looking for the concurrence of two events. In the Teledyne AGS, this would be represented by horizontally adjacent tiles each specifying a condition. An example is shown in figure 8.

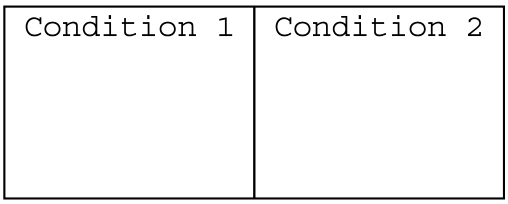

Figure 8.

“OR” Conditions

In the simplest form of an “OR” condition the user is looking for the occurrence of at least one of two events. In the Teledyne AGS, this would be represented by vertically adjacent tiles each specifying a condition. An example is shown in figure 9.

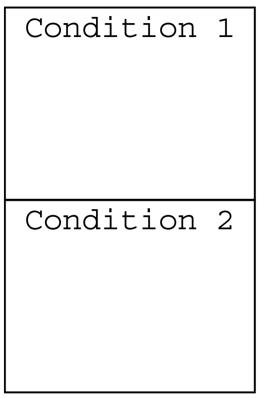

Figure 9.

Operator Precedence

Where “AND” conditions and “OR” conditions intersect, precedence may be indicated by a dashed line between the tiles, as opposed to the usual solid line. The dashed line indicates a stronger association. A simple example is shown in figure 10. The example represents the equation:

(Condition 1) AND ((Condition 2) OR (Condition 3))

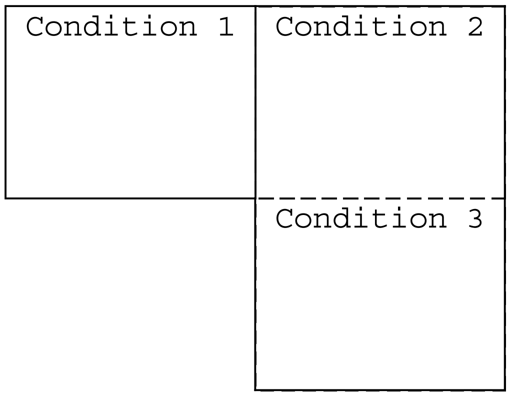

Figure 10.

Complex Condition CHI

The user interface for the specification of complex conditions allows the user to move or copy one or more tiles of the mosaic. The tiles can be moved or copied to another position in the mosaic or to another mosaic. The copy or move operations may be carried out via menu items or via click and drag operations. Double-clicking on a given tile will perform the “drill-down” operation which would open either a nested mosaic or a simple condition specification. Right clicking on a tile would allow for modification of the tiles properties. This would include specifying operator precedence or inversion of a condition.

Conclusion

The system in development allows the user of the AGS to specify simple conditions based on parameter values, using an intuitive, graphical CHI. The system also allows the user to combine such conditions by “AND”-ing, “OR”-ing or nesting the conditions, using drag and drop techniques.

Flight Data Interface and Management Unit (FDIMU) for the A340-500/600

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Abstract:

For the A340-500/600 Airbus develops the FDIMU which integrates the Flight Data Interface Unit and the Data Management Unit on the basis of the A330/340 to one single box. The objective is to save weight, electrical power and maintenance costs. We use the occasion to take into account experiences from previous programs and to implement improvements and to add additional functions.

The FDIMU is divided in a mandatory part which supplies the Flight Data Recorder with parameters and an Aircraft Condition Monitoring part on a second processor board which includes all Airbus defined standard functions as reports for engine condition monitoring, Auxiliary Power Unit health monitoring and others.

A software package which runs on a commercial PC on ground will allow the operator to customize the system and to add user defined functions for trouble shooting to the Aircraft Condition Monitoring part. Uploading of the user defined software will be done by disk or by a PC memory card.

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1. Introduction

The Flight Data Recording System of the a/c records parameters on the Flight Data Recorder to fulfill requirements from the airworthiness authorities. The A/c Condition Monitoring System acquires and processes parameters from various systems for trending and maintenance related tasks and for troubleshooting and for special investigations.

New rules from airworthiness authorities require to modify the parameter frame and the speed for the Flight Data Recording System for the A340-500/600. The new a/c type and new engine type requires the modification of the A/c Condition Monitoring System. Airbus takes the occasion to develop a new computer, i.e. combine the two units of the two systems, the Flight Data Interface Unit and the Data Management Unit to one box: the Flight Data Interface and Management Unit (FDIMU). It will work with a new processor and more memory capacity for internal parameter recording and will provide improved and additional functions.

The FDIMU is a step forward on the way of avionics integration. Advantages which can be achieved with the new controller are less weight, less power consumption, less space, less maintenance cost and state of the art technology. We take profit of experiences which we gained from the existing A310, A320 and A330/340 family and take into account the inputs which we received from various operators.

Besides this we recognize importance to a high flexibility which shall be achieved by means of an user friendly Ground Support Equipment which allows the operator to program and configure the system and to read out and display results.

2. Role of the Aircraft Condition Monitoring System and the Flight Data Recording System in Airbus A/c

The upper part of figure 1 shows the Flight Data Recording System, i.e. the mandatory part. Parameters are recorded by this system on the Flight Data Recorder and are used for incident and accident investigation.

The Aircraft Condition Monitoring System provides software which collects and processes data from up to 64 digital data busses. There are standard functions implemented which are defined, tested and certified by the Airbus Industrie and there is space for customer defined functions.
The system provides reports and recording frames for Engine Condition Monitoring, APU Health Monitoring, hard landing and inflight loads detection and Environmental Condition System trouble shooting and others.

The data from both systems the Flight Data Recording System and the Aircraft Condition Monitoring System are analyzed on ground. For example assessment of engine data for the purpose of Engine Condition Monitoring is performed with a software package provided by the engine manufacturers. The results and used to initiate maintenance actions and to define operational recommendations.

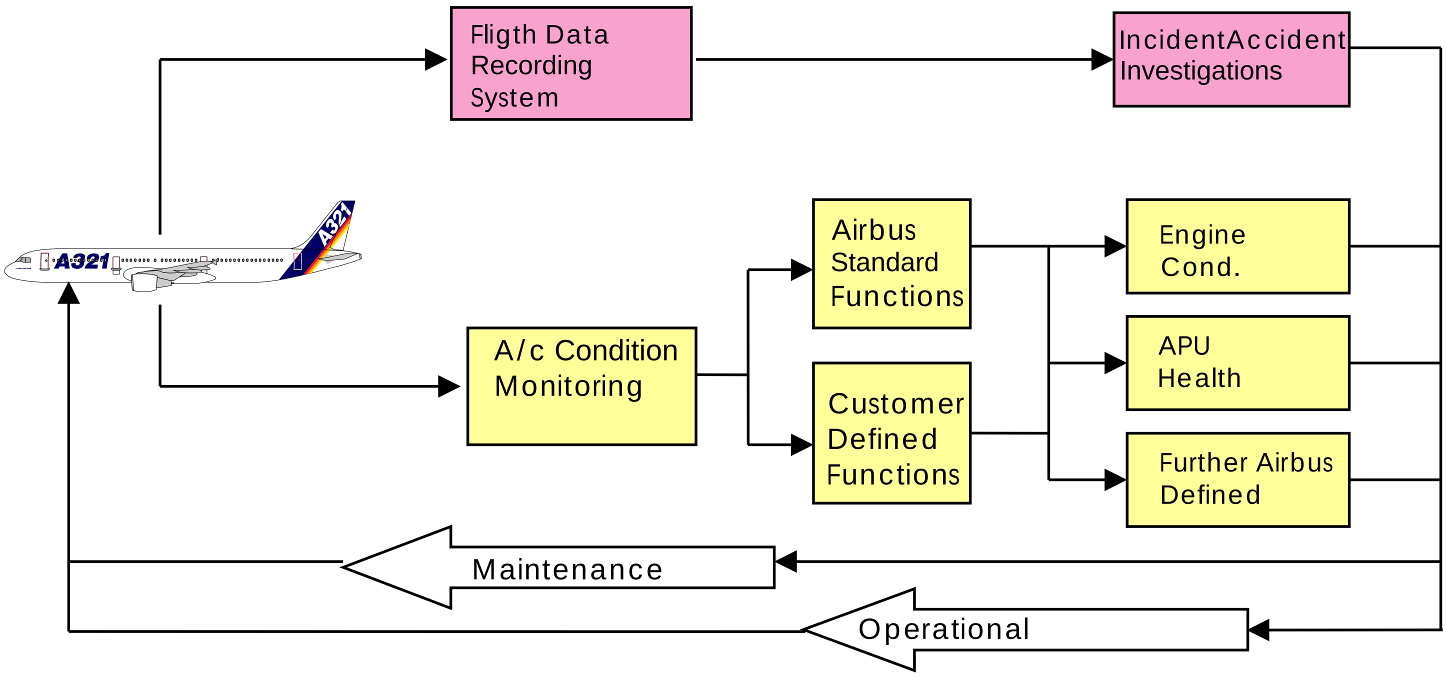


Figure 1 : Aircraft Condition Monitoring System and Flight Data Recording System in Airbus A/c

3. Actual A330/340 Architecture

Today the two systems, the mandatory Flight Data Recording System and the Aircraft Condition Monitoring System are fully separated and separately tested and certified (with different airworthiness categories).

The Flight Data Interface Unit sends data to the Flight Data Recorder and in parallel to an optional Quick Access Recorder.

The Data Management Unit sends data to an optional Digital ACMS Recorder or stores data in an internal SAR memory and generates reports which can be printed, downlinked or dumped on disk.

The Ground Support Equipment is used to configure the system and to perform customer defined programming and for display generated reports and recording files.

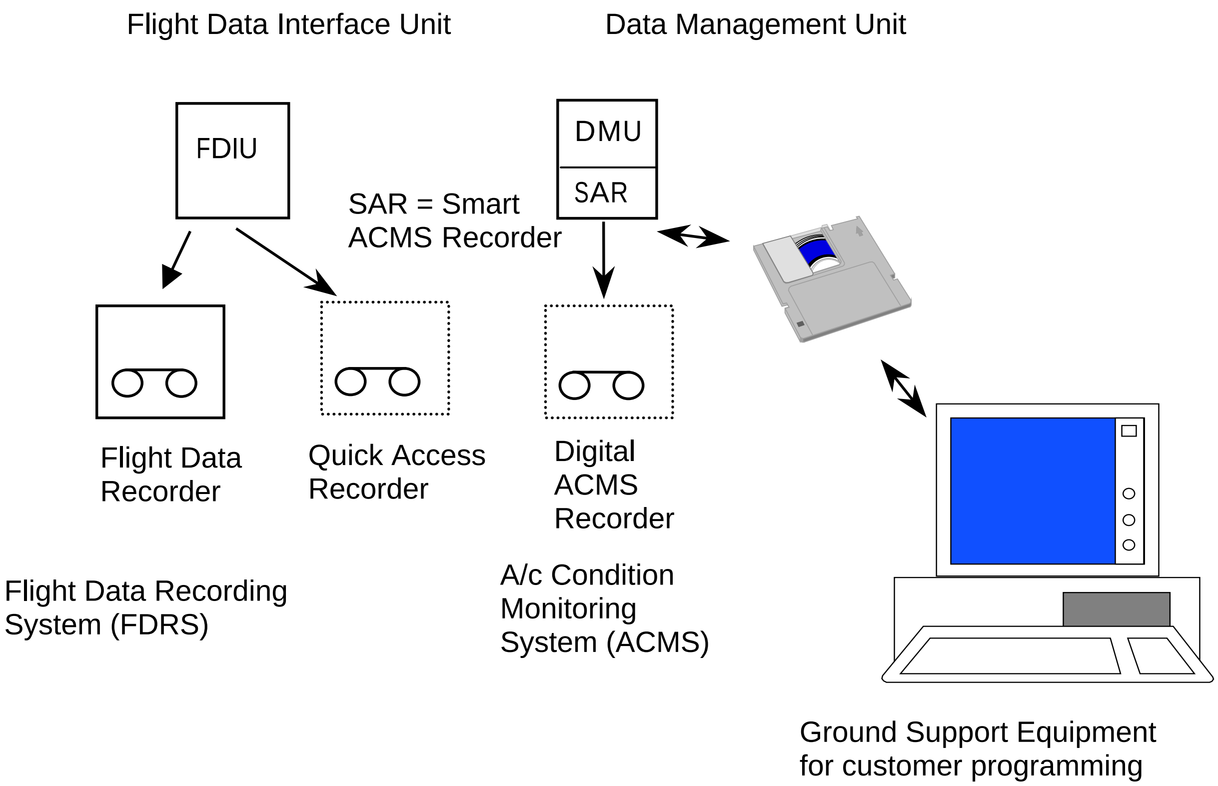


Figure 2: Actual A330/340 Architecture

4. New Architecture for the A340-500/600

For the A340-500/600 the two units, the Flight Data Interface Unit (FDIU) and the Data Management Unit (DMU) are combined to one box: the Flight Data Interface and Management Unit. The unit has one common power supply and two separate boards, one for the FDIU part, the 2nd for the DMU part. Within the DMU part there is a PC Memory Card installed which expands the DMU's internal storage capacity.

Besides the possibility to use the Quick Access Recorder or the Digital ACMS Recorder the parameters frames can be directed to the PC Memory Card. The PC Memory Card can be used for up- and downloading software and data.

It will be possible to update the mandatory parameter frame by loading new frames by disk in case that airworthiness authorities' regulations change.

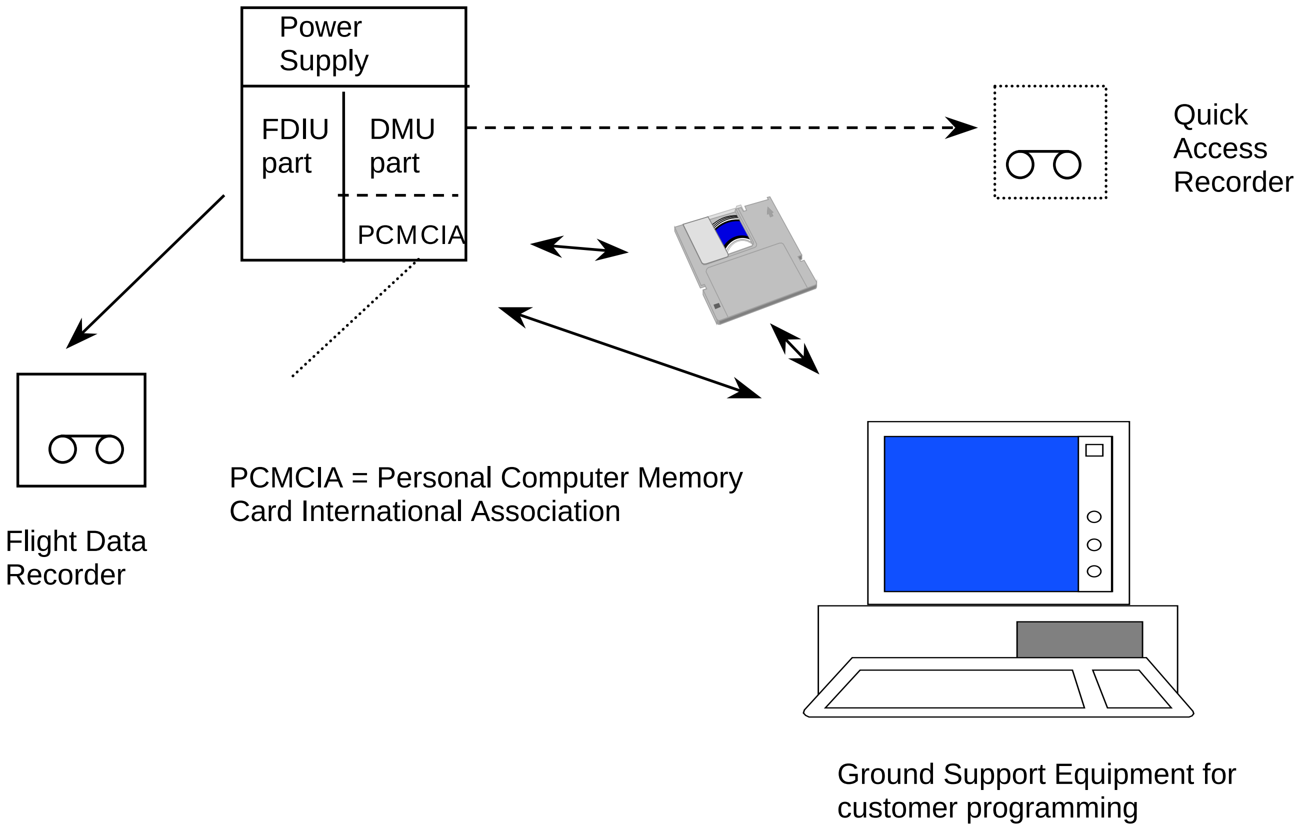


Figure 3: New Architecture for A340-500/600

5. FDIMU Interconnections and Locations

The FDIMU is located in the electronic bay of the a/c. It is connected to the Control Display Units in the cockpit pedestal which are used for display of parameters or set of parameters in real time. And it is connected to the cockpit printer which is mainly used for print out of reports. Besides this it is connected to the Air Traffic Service Unit which provides an air to ground wireless data link and which is used to uplink programmable items – mainly for temporary change of the configuration - and for downlink of reports and of information about generated reports and recording files. Upload and download of software and data can be done by the cockpit mounted Data loader or by the PC Memory Card

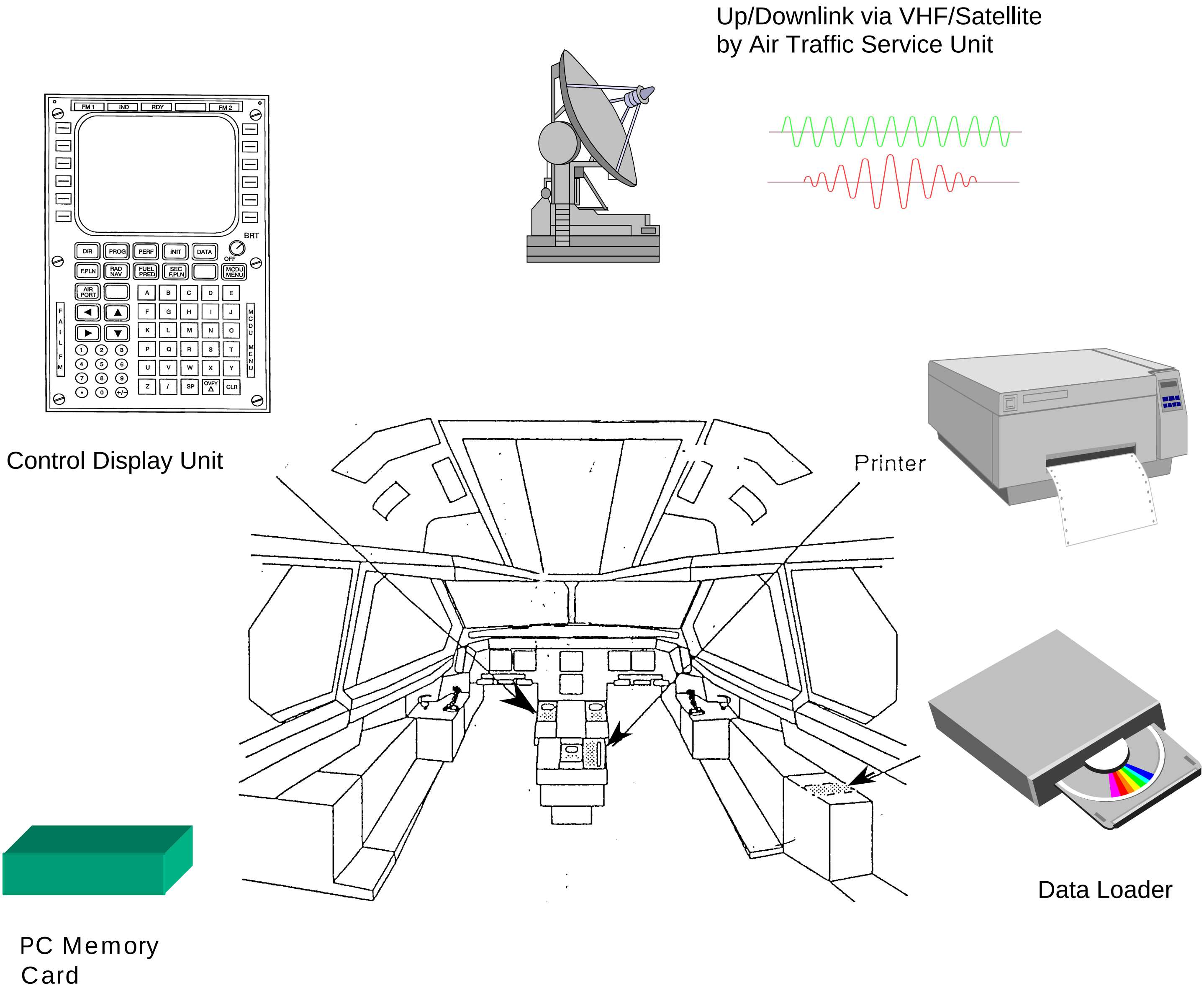


Figure 4: FDIMU Connections and Locations

6. FDIMU Tasks

The Flight Data Interface part of the FDIMU acquires and processes parameters and generates the mandatory parameter frame for the Flight Data Recorder.

The A/c Condition Monitoring part of the FDIMU acquires and processes parameters for the tasks as mentioned below:

- engine condition monitoring (for trending, performance and in case of limit exceedance of gas path or mechanical parameters)
- A/c performance monitoring
- Auxiliary Power Unit Health monitoring (trending and exceedance)
  cabin monitoring (Environmental Condition System), monitoring of pressure and temperature
- detection of inflight loads and hard landings
- data collection for Flight Operation Quality Assurance
- customer defined parameter recording for special investigations/trouble shooting

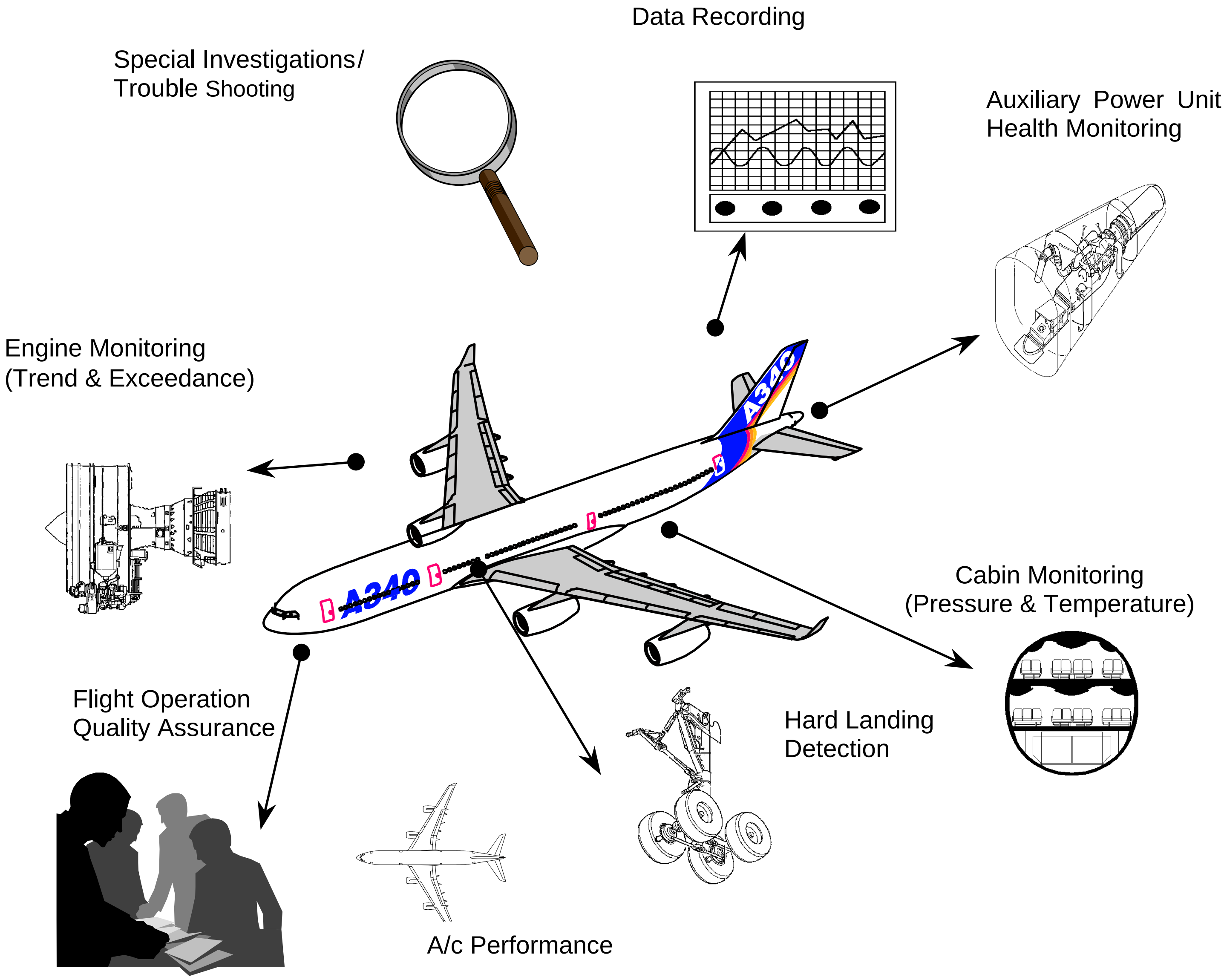


Figure 5: FDIMU Tasks

7. ACMS Processing Results

7.1 Reports

The FDIMU generates various reports for trending purposes, at exceedance of predefined limits and for trouble shooting.

The example below shows a “Cruise Report” which was generated during an A330 test flight at 41000 ft altitude during the a/c was in cruise flight mode. It was generated under “stable conditions”, i.e. a set of engine and a/c related parameters was found stable during a certain time period.

For each of this type of reports the FDIMU calculates a “stable frame quality number”, i.e. a number which describes the stability of this time period. The FDIMU keeps on looking for the time period with the “best” quality number for the entire flight and at the end of the flight it stores only the report with the best quality number.

```
          A330 ENGINE CRUISE REPORT <01>    PAGE 01 OF 01

   ACID    DATE      UTC      FROM TO     FLT                CODE
CNT

C1 F-ABCD  99MAY15 02.51.11 LFBO VVNB 20ABCDE123      5000
097 8F

 PRV PH   TIEBCK     DMU IDENTIFICATION       MOD  AP1 AP2

C2 002 06.0 000000  SE6T04 VT6004 C00050   C00    052 052 2E

 TAT      ALT     MN    SYS       (.... BLEED  STATUS ....)       APU

C3 N28.8  41005  0.818 111    0.97 111 10 0 01 111 0.97   -  A1
C4 N28.8  41003  0.818 111   F

  ESN      EHRS   ERT      ECYC   ECW1     EVM   QE

C5 733304 00051  00009 00022  01010  06064 12 AA
C6 733305 00048  00036 00023  10101  06064 30

   EPR    EPRC   N1       N2      EGT    P5       FF      P2       P25

N1 1.331 1.331   081.98 084.9 0374  05.406 02281 04.047 12.527 3A
N2 1.331 1.331   081.85 085.2 0373  05.434 02268 04.049 12.503 32

 T25       T295  P3      T3      TCA      TCC    LPTC     SVA      B25

S1  072.5 113.4 118.0 436.2 090.5  077.3  090.5     089.43  N01.2 E8
S2  072.3 142.7 118.1 438.8 087.1  070.8  087.1     089.61  N00.0 F4

  GLE     PD  TN   OIQH    OIP OIT     OC   VF    VC     VH     VL
```

7. ACMS Processing Results (continued)

7.2 System Data Pages

A new function of the FDIMU will be the Systems Data Pages. This is a collection of data for some systems which can be easily displayed on the cockpit's Control Display Units and allow an overview – mainly for maintenance purposes – over a set of significant parameters.

The figure below shows the initial menu which allows to select System Data Pages for the Environmental Control System or for the Auxiliary power Unit or for the Proximity System Control Unit (Doors).

	01	**ACMS : SYSTEM DATA**	1 / X	
	02		STORED	
-1L	03	**<ATA21-ECS**	**DATA>**	1R-
	04		STORED	
-2L	05	**<ATA49-APU**	**DATA>**	2R-
	06		STORED	
-3L	07	**<ATA52-PSCU**	**DATA>**	3R-
	08		STORED	
-4L	09	**<ATAXX-XXX**	**DATA>**	4R-
	10		STORED	
-5L	11	**<ATAYY-YYY**	**DATA>**	5R-
	12			
-6L	13	**<RETURN**	**PRINT** ★	6R-
	14	↑↓ SCROLL		

Figure 6: System Data Pages Initial Menu

7.2 System Data Pages (continued)

The example below shows the 1st part of the System Data Page for the Auxiliary Power Unit. It allows to display and print parameter values with a one second rate. It is also possible to store this information on disk or to downlink it to ground.

The Ground Support Equipment allows to program start and stop logics for those System Data Pages later on during flight without crew action.

	01	**ACMS : ATA49 - APU START** 1/2	
	02	← SEND/DUMP/STORE →	
-1L	03	★START PRINT TIME-SERIES	1R-
	04	APU SPEED [%RPM] 120	
-2L	05	EGT MAX T5 [DEGC] -1800	2R-
	06	EGT T5 [DEGC] -1200	
-3L	07	TIT T4 [DEGC] -2400	3R-
	08	FUEL TM CMD [MA] 8000	
-4L	09	FUEL FLO CMD [LB/h] 1200	4R-
	10	FUEL FLO [LB/h] 1200	
-5L	11	FUEL FLO MAX [LB/h] 1200	5R-
	12	LABEL 37 111111111111	
-6L	13	**<RETURN PRINT**★	6R-
	14	↑↓ SCROLL ←→ SELECT	

Figure 7: System Data Pages for Auxiliary Power Unit parameters

7. ACMS Processing Results (continued)
7.3 Recording Files

For larger amounts of data there is a possibility to store parameters in the FDIMU's internal 4 MB memory in compressed form. In parallel the same information can be stored in the PC memory card.

The parameters which shall be recorded and the events when the recording shall start and stop can be defined by the operator by using the Ground Support Equipment.

After the flight the recorded parameters can be displayed with a read out software package.

The example below shows the presentation of engine data which were generated during an A330 test flight (for this figure the read out tool of the actual A330 DMU was used).

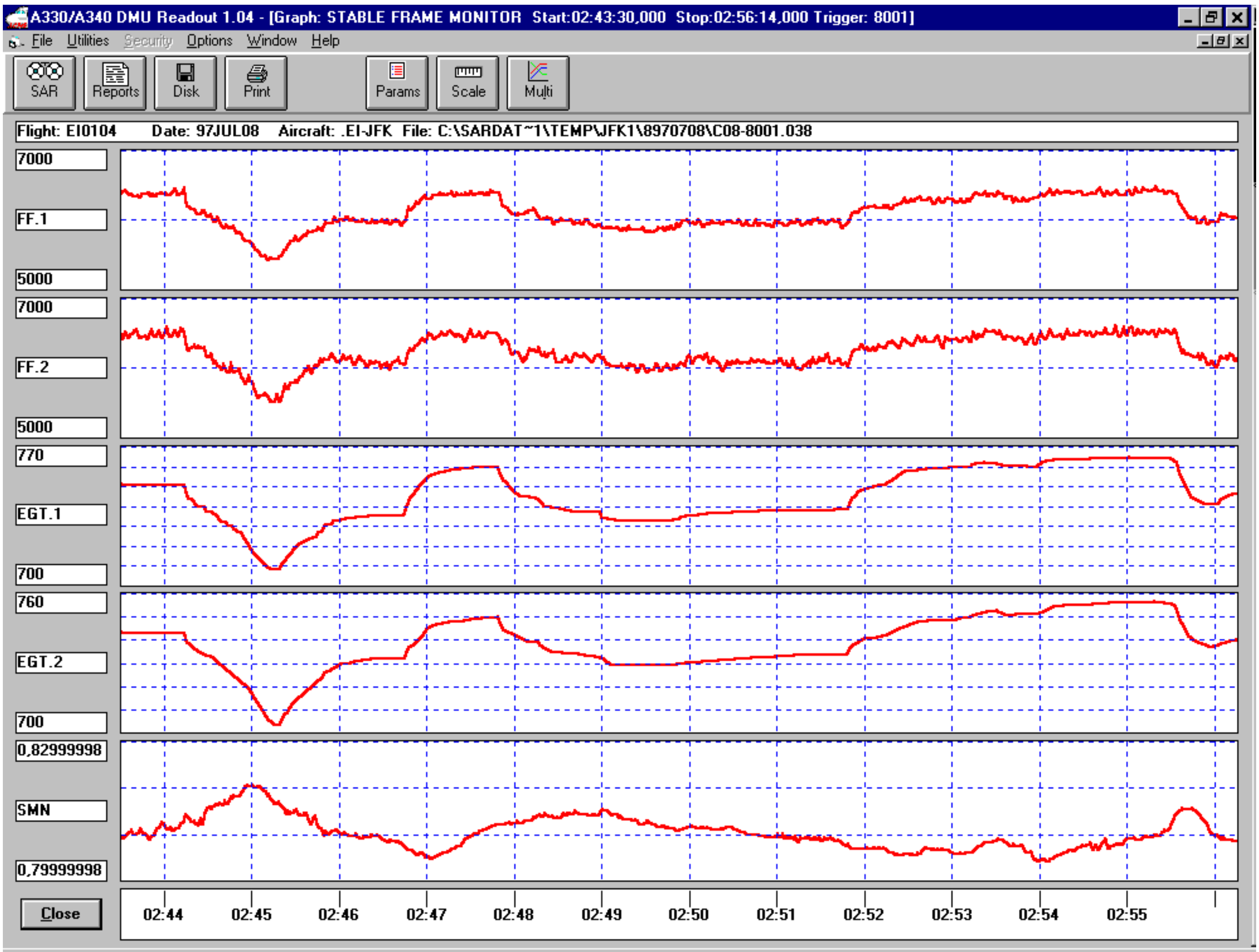


Figure 8: Read Out of A330 engine data

7. ACMS Processing Results (continued)

7.4 Recording on Quick Access Recorder (QAR) and Digital ACMS Recorder (DAR)

The FDIMU provides interfaces which allow to connect an optional QAR for recording of the same parameter frame which is recorded on the Flight Data Recorder. And it allows to connect an optional DAR. The parameters for the DAR and the conditions when the recording shall start and stop can be programmed by the operator with help of the ground Support Equipment. Recording speed can be 64, 128, 256, 512, 1024 words per second.

The QAR and DAR recording frames can also be directed to the PC memory card within the FDIMU.

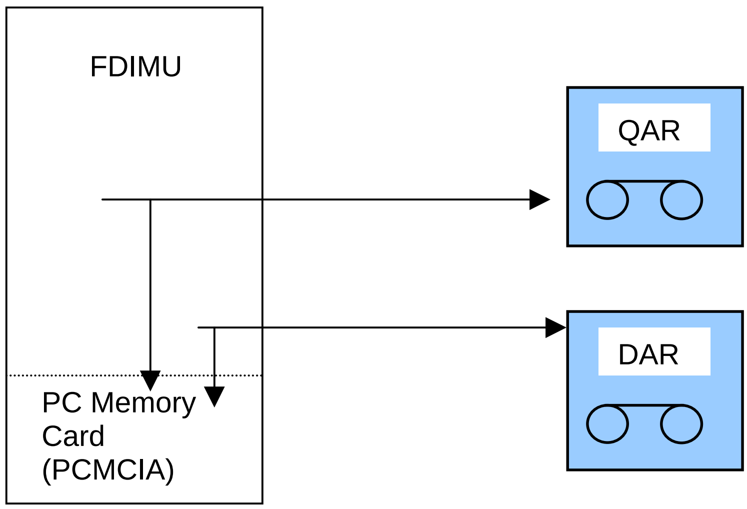


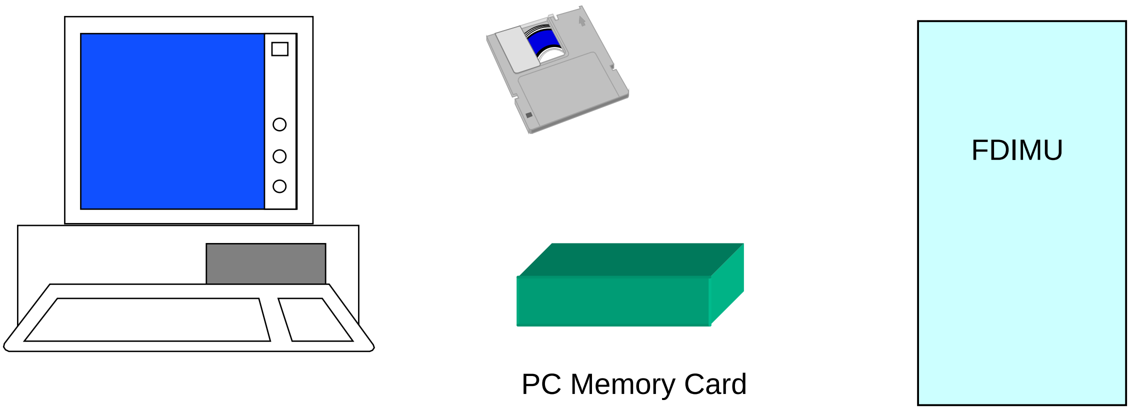
Figure 9: FDIMU connection to QAR and DAR

8. Ground Support Equipment

The Ground Support Equipment is a software package which runs on a PC on ground. It enables to program and to configure the system. Items which are programmable are as follows:

- report programming, parameters and start/stop logic
- recording frames programming, parameters and start/stop logic
- memory partitioning, amout of memory which shall be reserved for reports, report types and recording files
- output rules, print, downlink, storing rules
- new menus, e.g. Systems Data Pages

Besides programming of the FDIMU the Ground Support Equipment shall also be used for read out and presentation of the reports and recorded parameters.



9. Onboard Server (under Investigation)

Currently we investigate the integration of an Onboard Server to which the FDIMU shall be connected. The Onboard Server shall be used for mass storage of reports and recording files, repository of loadable software and databases, post processing of FDIMU data and improved human-machine interface. The figure below describes the possible architecture. The FDIMU with it's various tasks as described above is separated from the network by a firewall. The network – to which other systems are connected – allows to program and read out FDIMU data by a Maintenance Access Terminal (portable laptop computer) via an A/c Wireless Local Area Network Unit. The Onboard Server will also allow to read and dump FDIMU data to the airlines maintenance station on ground via a Terminal Wireless Local Area Network Unit.

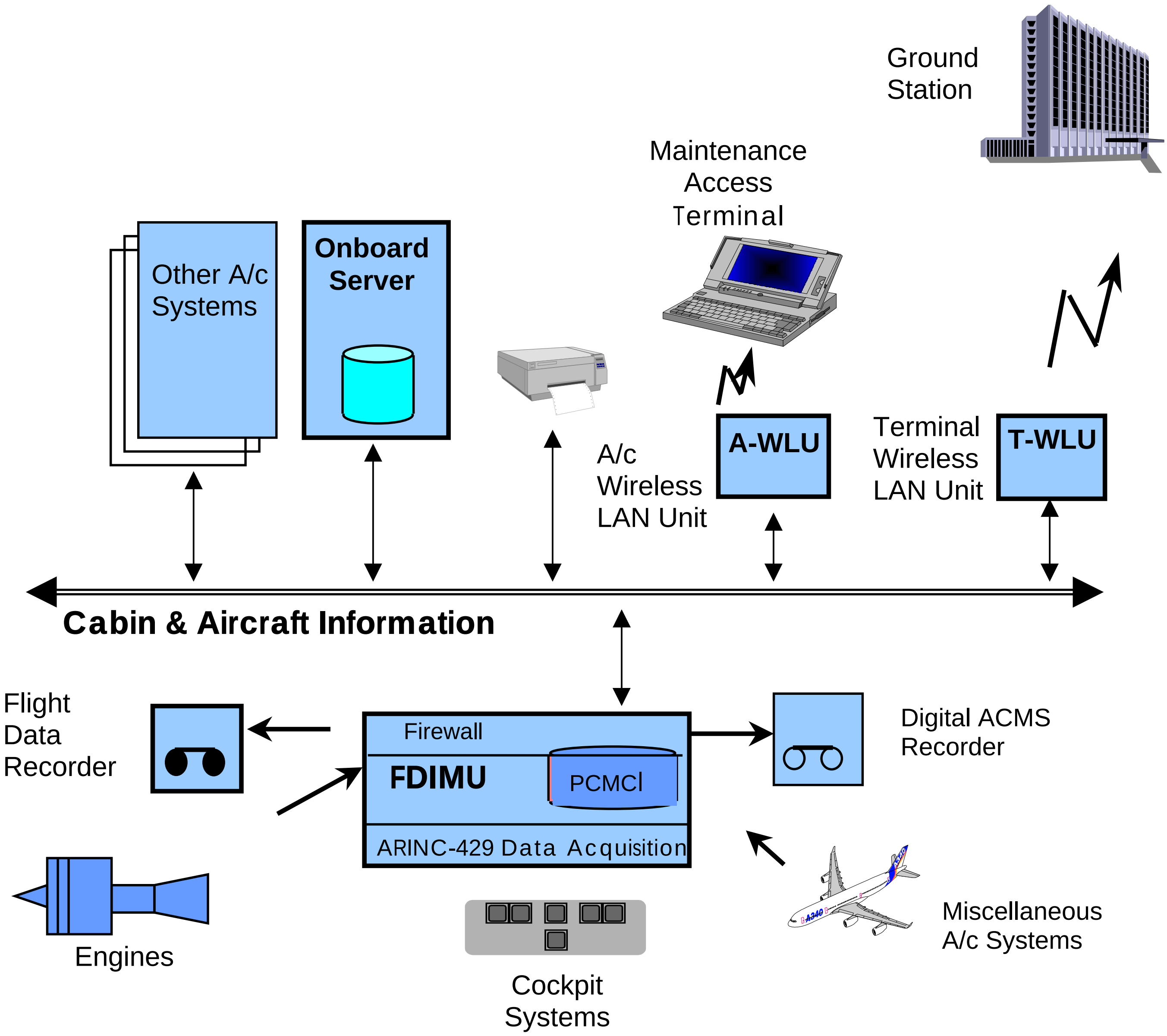


Figure 11: FDIMU Connection to Onboard Server

10. Conclusion

With the new approach we assure that state of the art hardware and software for the A340-500/600 is used. For the basic criteria, customer friendly handling, weight, costs and flexibility the FDIMU represent a good solution.

Fatigue and Operational Loads Monitoring of the Red Arrows

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Squadron Leader Steve Reed (Defence Evaluation Research Agency, LAS, Farnborough, UK)

Abstract

The Royal Air Force Aerobatics Team (RAFAT), the Red Arrows, has operated the British Aerospace Hawk since 1980. Understandably, the Red Arrows' service usage differs significantly from the RAF's more common role for the Hawk as a basic flying and weapons fast jet trainer. However, the methods of accounting for fatigue accrual and airframe lifing are essentially the same.

An Operational Loads Monitoring (OLM) programme was undertaken between 1997 and 1999. This assessed the adequacy of the structural monitors in conjunction with the clearances derived from the Hawk Full Scale Fatigue Test (FSFT), reviewed the airframe dynamic response, particularly that of the empennage, and identified especially damaging manoeuvres specific to Red Arrows flying. This paper describes the principal findings of the OLM programme and some of the follow-on activities.

Introduction

The Royal Air Force Aerobatics Team (RAFAT), better known as the Red Arrows, was formed in 1965 originally flying the Folland Gnat and has displayed over 3300 times in fifty countries. In 1980 the British Aerospace (BAe) Hawk, a tandem seat, single engined advanced trainer and operational aircraft, replaced the Folland Gnat. The Hawk standard operated by the Red Arrows is essentially that introduced into RAF service in 1976 with the exception of the uprated Rolls-Royce Adour 861A turbofan engine and a centreline diesel pod for smoke trail generation.

An Operational Loads Measurement (OLM) programme had been undertaken with two RAF Hawks between 1985 and 1988 but this excluded Red Arrows flying. However, the method of accounting for fatigue accrual and lifing of the airframe for operation of the aircraft in an aerobatic display team role is essentially the same as for the more common fast jet trainer role. Given that the loading applied to the Hawk Full Scale Fatigue Test (FSFT) had been amended as a result of findings from the original Hawk OLM exercise of the mid-1980s the importance of performing a Red Arrows OLM programme had been acknowledged by all parties concerned. When the RAF initiated the Hawk Life Extension Programme (LEP) [1,2] to extend the aircraft's Out of Service Date (OSD) to 2010 it was deemed imperative to perform a further OLM programme with two aircraft, one of which, XX260, would be based with the Red Arrows.

OLM Programme Aims

The aim of the programme was to gather in-flight direct strain data and associated flight parameters to quantify the effects of Red Arrows flying on the structure of the Hawk aircraft, and specifically to:

i. investigate the adequacy of the fatigue meter formulae,

ii. determine the adequacy of the loads applied to the Hawk Life Extension Fatigue Test (LEFT),

iii. investigate the effect of dynamic loading on the aircraft, particularly with respect to the empennage,

iv. identify flight regimes, specific to Red Arrows flying, which are particularly damaging.

One of the main concerns with Red Arrows flying was the effect of load mechanisms which are not present during normal in-service usage, such as use of the airbrake as a flying control during the display and close formation flying, Figure 1. The tailplane, fin and rear fuselage are subject to additional loading due to the rear formation aircraft flying through the jet efflux of the front aircraft. The Red Arrows aircraft are known to use flying in this state as a cue for maintaining formation position. Prior to this OLM the structural consequences of this type of flying had not been quantified for the Hawk TMk1/1A aircraft.

The Red Arrows 'Diamond Nine' formation is annotated in Figure 2, although this configuration forms only a small element of their display routine.

OLM System Configuration

The recording system comprised an Aydin Vector Data Acquisition Unit (DAU) for converting the analogue strain gauge and aircraft parameter signals to digital which were transmitted to the on-board NAC video recorder via a Heim PCM encoder. The system powered up automatically at 45% maximum engine RPM.

The aircraft was fitted with 36 strain gauges (Figure 3), 14 of which duplicated the positions used in the original OLM programme with minor changes to account for a modified wing. The OLM strain gauge fit was replicated on the LEFT. The empennage strain gauges were recorded at 2048 samples per second (s/s) on each channel in order to account for the high frequency vibration that was identified during a dedicated dynamic response flight trial programme in 1988. The tailplane was known to have significant structural responses at approximately 20Hz, 70Hz and 90Hz. All other strain gauges were recorded at 256s/s per channel.

Additionally, 19 flight parameters were recorded to enable aircraft accelerations and control surfaces to be monitored, and to identify aircraft manoeuvres and flight conditions at which significant events occur (TIME defined as single parameter). These are detailed in Table 1.

Ref	Parameter	Sampling Rate (s/s)	Range
AAP	Aileron Angle Port	16	± 12 degrees
ALT	Pressure Altitude	8	0 - 48,000 ft
BAA	Bank Angle	16	±180 degrees
FLA	Flap Angle	16	Up - Mid - Down
FTP	Fuel Tank Pressure	16	0 - 30 psi differential
IAS	Indicated Airspeed	8	0 - 500 knots
LAC	Lateral Acceleration	32	±2g
NAC	Normal Acceleration	32	-6g to +12g
P6	Static Exhaust Pressure	16	0 - 15 psi
RRR	Roll Rate	32	±300 degrees/second
RUD	Rudder Angle	16	±20 degrees
THR	Throttle Angle	16	Closed - Idle - Open
TPA	Tailplane Angle	16	-15 to +6 degrees
VAS	Vertical Acceleration	32	-6g to +12g
ABP	Airbrake Position	4	Event (0 or 1)
CWP	Central Warning Panel	4	Event (0 or 1)
UND	Undercarriage Down	4	Event (0 or 1)
UNU	Undercarriage Up	4	Event (0 or 1)
SEC	Time - Second Count	4	0 to 59
MIN	Time - Minute Count	4	0 to 59
HR	Time - Hour Count	4	0 to 23
DAY	Time - Day Count	4	1 to 365

Table 1 - OLM Aircraft Flight Parameters

Structural Clearance & Fatigue Monitors

The Hawk was originally designed for a 6000 hour fatigue life under a Flight Training School (FTS) load spectrum [3]. The design was largely substantiated by the Hawk Full Scale Fatigue Test (FSFT), operated by BAe Kingston between 1979 and 1985, and was subject to a spectrum that was principally based on the RAF's utilisation of the aircraft in its original role as a basic fast jet trainer.

It was evident from the original Hawk OLM programme that FSFT tailplane and rear fuselage loading was inadequate to provide sufficient structural clearance for the aircraft operated in a tactical weapons role. A phase of dedicated tailplane loading, based on OLM information, was therefore applied to the FSFT. At the time the fuselage was lifed only in flying hours and the OLM had also identified that aircraft operated in the Tactical Weapons Unit (TWU) were accruing fatigue damage at a greater rate than the fatigue test. By reacting the additional FSFT tailplane loads through the fuselage the FSFT fuselage damage rate was increased at the same time.

A military aircraft is generally operated within many varying roles which place significantly differing demands on the airframe and hence affect the rate of fatigue damage accrual. To account for this, fatigue formulae are derived which track the life used of a component in a more precise manner than by simply using flying hours. The safe life of a component is derived either from a full scale or coupon fatigue test or by calculation. All RAF Hawk aircraft operate three fatigue formulae: wing, fuselage and fin. The wing and fuselage formulae are based on the aircraft's normal acceleration spectrum which is monitored by a counting accelerometer with six positive 'g' windows (up to +8g) and three negative 'g' windows relative to 1g (down to -1.5g). The fin fatigue formula is based on sortie specific coefficients derived from the original Hawk OLM and sortie duration. As the original Hawk OLM programme did not include Red Arrows flying a single coefficient, calculated from TWU data, covered all Red Arrows operations and all formation positions.

Flight Programme

The OLM flight programme consisted of three principal phases :

i) display training in Cyprus

ii) main display season

iii) synchro pair work up training.

A total of 194 Red Arrows flights were recorded: 61 from display training in Cyprus (including transit to and from Cyprus); 106 from the display season and 27 from the synchro pair work-up flying. After data verification checks 151 flights were selected for damage analysis. The distribution of these flights with respect to Sortie Profile Code (SPC) and formation position are given in Table 2.

	Formation Position										
Sortie Profile Code (SPC)	**1**	**2**	**3**	**4**	**5**	**6**	**7**	**8**	**9**	**0****	**Total**
21 - General Handling & Training	7	4	2	2	4	7	8	6	8	9	**57**
22 - Transit, Display, Transit		3		1		6	5	3	3	2	**23**
23 - High Level Transit		1				1			2		**4**
24 - Display, Take-off @ Display Site		2		1	1	2	1	1			**8**
25 - Low Level Transit	1	3		1	2	7	5	5	5	4	**33**
26* - Synchro Pair Work-up						6	20				**26**
Total	**8**	**13**	**2**	**5**	**7**	**29**	**39**	**15**	**18**	**15**	**151**

* SPC 26 was introduced as a result of the Red Arrows OLM
**Formation position 0 refers to any flight not designated position 1 to 9

Table 2 - OLM Flight Programme

Data Analysis Process

The raw decommutated video cassette data were stored on CD in binary Digital Data Units (DDU). Data integrity checks were performed to remove spikes, replace poor minor or major data frames and identify signal discontinuities based on bit synchronisation and internal data stream time stamps. If more than 1% of the data was replaced or reconstructed the flight was rejected. It became clear early in the OLM programme that the build up of dirt on the VCR recorder heads accounted for the steady increase in poor data over time. As a result the VCR recording head was cleaned at three monthly intervals.

In addition to recording the 36 strain gauge channels the 18 strain gauge bridge excitation channels were also monitored at 4 samples per second. This proved to be a valuable exercise as three flights exhibited smooth but extreme tailplane gauge response which suggested severe loading. However, this was not supported by the flight parameters. The gauge response was attributed to the bridge excitation transgressing acceptable limits.

Strain gauge fatigue cycle counting was performed using the rainflow technique. Fatigue analyses were based on traditional Miner's S-N methodology using S-N curves appropriate to each feature (aluminium alloy notch, low load transfer bolted joint, high load transfer bolted joint or lug). As was explained previously Hawk structural clearances are principally derived from the achievement of the Hawk FSFT, Interim FSFT and LEFT (Figure 4) so S-N curves were 'set' using fatigue test stress histories to either actual feature failures (e.g. cracking of the fuselage fuel tank frame 15 beam at 29250 test hours) or the projected end of test life (e.g. the wing to 36000 test hours). This provided a relative life for each feature compared to the fatigue test loading history.

In addition to the fatigue analyses using the 'mean S-N' curve approach the 'safe S-N' curve method was also considered, this now being the recommended methodology of the British Defence Standard 00-970. The use of the 'safe S-N' approach was for comparison only in order to assess the effects of using a later recommended design standard on structure designed using earlier methods. The aircraft's clearance remains via the mean-life approach.

The frequency response characteristics of each component were investigated using standard Fourier techniques, and of particular interest was the empennage region. Consequently, the tailplane and fin strain gauge responses from the particularly high activity synchro pair work-up flights were low-pass filtered in order to assess the relative contributions to the total damage from the manoeuvre and buffet regimes.

Analysis Results

The analysis of the data was reviewed by component with respect to the corresponding fatigue monitor and furthermore by considering the impact of both sortie type and formation position.

Wing

As an indication of the relative severity of Red Arrows flying Figure 5 shows g-exceedence curves for representative SPC usage (for 1000 hours) using OLM recorded normal acceleration data and compares these to the original Hawk g-exceedence design curve. This alone makes clear the importance of performing regular operational measurement programmes on aircraft whose utilisation is different from that originally anticipated, a point which has been emphasised before [4].

Figure 6 compares the calculated OLM strain gauge damage from the port wing channels to that derived from the RAF Hawk wing Fatigue Meter Formula (FMF) for all sorties. Values on or below the solid line indicate that the wing FMF is monitoring the location adequately. Compared to typical RAF Hawk usage the damage rates are high, particularly for the synchro pair work-up flights. However, it can be seen that, with the exception of the inboard rear spar location, the wing FMF is generally offering protection for RAFAT usage.

Points lying above the line at the outboard front spar location are mostly from the synchro pair work-up exercise. This phase of flying involves significant and repeated negative 'g' manoeuvring, often exceeding the lowest g-meter window. This, combined with the high proportion of asymmetric manoeuvres which the g-meter cannot fully monitor (acceleration effects increase outboard spanwise with aircraft roll), results in the FMF underpredicting the wing damage. However, the synchro pair work-up exercise forms a relatively small proportion of Red 6 and 7's annual flying and overall the wing outboard region is protected by the existing wing fatigue meter formula.

The failure of the wing fatigue meter formula to monitor adequately the inboard rear spar region was noted during the original Hawk OLM programme. This result is most probably due to the wing centre of pressure being further aft on the flying aircraft than that of the fatigue test loading. Although there is not believed to be a problem with this region due to the low rear spar stress levels a further detailed assessment is currently in progress.

Fuselage

Figure 7 presents the results of the fuselage upper longeron strain gauge analyses with respect to the fuselage fatigue meter formula. The longeron gauges were located at, or close to, the main fuselage manufacturing joints at frames 12, 20 and 28.

The diagram indicates that the fuselage fatigue meter formula, which like the wing FMF uses normal acceleration g-exceedences and aircraft mid-sortie mass as inputs, is offering adequate protection for the monitored locations. The apparent over prediction of fatigue damage by the formula is the result of fuselage life being measured in flying hours only at the time of the original Hawk OLM programme. The in-service longeron damage rate from tactical weapons usage had been found to be greater than that being applied to the fatigue test and so the longeron fatigue test loading was increased to similar levels. However, the fuselage fatigue formula was derived using the original fatigue test loading and hence the damage rates obtained using S-N curves set with actual fatigue test loading are low compared to those predicted by the formula.

In contrast, the damage rates from the strain gauges located at frame 33 are very high. This is due to tailplane vibration loading being transmitted into the fuselage at the frame 33 connection, a load mechanism which is not fully represented on the LEFT. Although the Hawk is a safe-life aircraft, the structural integrity of this area is assured by inspections, the intervals of which are being reviewed for Red Arrows flying.

Wing /Fuselage Links

The low wing configuration of the Hawk principally places the wing to fuselage attachment links in compression for positive 'g' manoeuvres. As Figure 5 illustrates though the rear of the formation, particularly the synchro pair, have a significant negative 'g' content as part of their display, which puts the links in tension. Figure 8 plots the front spar and auxiliary spar wing to fuselage link damage rates, derived using a high load transfer bolted joint S-N curve, against wing FI. Two points are interesting to note: firstly the auxiliary spar wing link is loaded through wing torque during engine run-up against brakes before take-off and by undercarriage drag loads during touchdown which explains the apparent failure of the wing fatigue meter formula to monitor damage at very low wing FI values; and secondly the fatigue formula does not monitor adequately all of the high wing FI synchro pair work-up flights due to the aircraft's negative g-spectrum exceeding that embedded in the formula. As with the wing, however, when assessed against a representative annual usage spectrum instead of individual flights the wing fatigue meter formula offers adequate protection.

The impact of negative 'g' on the wing to fuselage links is revealed in Figure 9 which shows the analysis of the wing link lug features by considering only the tensile load spectrum and a lug specific S-N curve. From this it appears that the wing to fuselage link damage increases disproportionately at high wing FI levels, particularly for the synchro pair. To ensure structural integrity of the wing/fuselage links and other lug features in the vicinity of the links a separate detailed analysis of this region was undertaken for Red Arrows spectra. This was important as the negative 'g' spectra of Reds 6 to 9 exceeds that monitored by the aircraft g-meter and tested by the fatigue test. All features analysed demonstrated adequate fatigue life by calculation using appropriate safe-life factors.

Fuselage Fuel Tank

Coincident with the start of the OLM programme a dedicated block of fuel tank pressurisations was applied to the Hawk LEFT to rectify a shortfall in the fatigue test loading when it was designated the FSFT. During a routine inspection, before the completion of the pressurisation catch-up block, multi-site damage (MSD) fatigue cracking was identified in the fuselage fuel tank side skins.

Operational fuel tank strain gauge and system pressure data from both OLM aircraft (the Trials Directive for the In-Service OLM Hawk was amended to capture representative data from the full spectrum of RAF usage) was used to assess when RAF aircraft would be expected to experience similar cracking and which aircraft were most at risk. At first, Red Arrows aircraft were categorised as high risk due to their severe g-spectrum. However, OLM data made it apparent that contrary to initial speculation, because the Red Arrows aircraft almost always take-off with a half full fuselage fuel tank the fuel tank inertia loads were relatively low. As a result of this analysis the Red Arrows aircraft were removed as candidates for the RAF's Hawk Fuselage Replacement Programme (FRP) which will extend the life of the In-Service fleet to 2010 and probably beyond.

Fin

The fin fatigue formula was derived from original Hawk OLM data and accounts for variations in aircraft usage with sortie profile code (SPC) specific coefficients. A single coefficient, 9.43, was used for all Red Arrows SPCs which meant that Red Arrows fin FI varied only by flight duration. This is clearly anomolous as in practice the most damaging Red Arrows flights for the fin are the shortest (synchro pair work-up) and the least damaging the longest (high level transit) which was in direct contradiction to the fin fatigue formula. However, no better data were available and, pending Red Arrows OLM data, it was hoped that the coefficient used would provide adequate protection on average.

Figure 10a shows the damage per hour from the fin rear spar gauge for the non-transit flights (excluding synchro pair work-up) with respect to diamond nine formation position. Each column represents a flight and the data for Reds 6 and 7 is shown last for clarity only. The diagram clearly illustrates the significant damage rate variation throughout the display positions but also within nominally similar flights, i.e. all display or practice.

Emphasising again the severity of synchro pair flying Figure 10b compares the fin damage per hour rates for Reds 6 and 7 from the main display season (same data as in Figure 10a) to that of the work-up exercise.

Due to the vast range of damage rates it was concluded that the only reliable method of monitoring fin damage was through an active monitoring system (strain gauge and parameter based). As an interim measure the fin fatigue formula for Red Arrows usage was reviewed and revised SPC and formation position specific coefficients derived. To reduce the management burden to a minimum for RAF personnel the Red Arrows aircraft were considered in four groups based on the similarity of the display routine and damage results. The updated fin fatigue formula coefficients, detailed in Table 3, were retrospectively applied to all fins that had seen Red Arrows service resulting in maximum fin FI adjustments of +49.6% and -32.9%.

SPC	**Formation Position**			
	1, 2 & 3	**4 & 5**	**6 & 7**	**8 & 9**
21	5.2	54.0	114.4	40.0
22	2.1	15.7	161.7	30.2
23	2.0	2.0	2.0	2.0
24	2.9	15.7	161.7	30.2
25	2.0	2.0	2.0	2.0
26	-	-	490.1	-

Table 3 - Updated Red Arrows Fin Fatigue Formula Coefficients
(Original Red Arrows Coefficient : 9.43)

A spectral analysis of fin data identified the principal fin response frequencies as 12Hz, 15Hz, 21Hz and 69Hz. The 21Hz and 69Hz responses are driven by asymmetric tailplane vibration. To assess the contribution of high frequency fin response to total damage the fin strain gauge time histories from the synchro pair work-up flights were filtered using a digital 3-pole Butterworth low pass filter with a 10Hz -3dB cut-off point. Approximately 15% of fin damage from these flights could be attributed to high frequency response indicating that fin damage is principally a function of manoeuvre loading.

Tailplane

Figures 11a and 11b show the equivalent damage rate information for the inboard front spar tailplane strain gauge S02, as Figures 10a and 10b did for the fin. As with the fin, the tailplane damage rates vary significantly depending on the formation position. It is only the front three aircraft that exhibit damage rates similar in magnitude to standard RAF Hawk Tactical Weapons usage which in itself is more severe than the design spectrum, the other six positions being far in excess of this. Again, nominally similar flights produced large variations in fatigue damage with Red 7 during the synchro pair work-up being the worst.

The Hawk tailplane has neither a reasonable monitor, such as the g-meter for the wing, or an operationally derived fatigue formula as for the fin. The structural integrity of this component is maintained by analysis based inspection. The structural clearance derived from the Hawk Full Scale Fatigue Test, even with the additional phase of tailplane loading initiated by the original OLM programme, is of limited value due to the unrepresentative tailplane loading of the FSFT. Loads were applied with a wiffle tree system via a single jack on each side of the tailplane in order to match damage rates recorded by the original Hawk OLM at two strain gauge locations, S01 & S02. Therefore, the high frequency buffet effects which are so significant for tailplanes operated in a Red Arrows role are not fully represented.

Figure 12 shows typical tailplane strain gauge responses for Red 6 during a 6g formation break manoeuvre. Inboard the dominant modal response frequency is 21Hz, with 69Hz and 93Hz modes increasing in intensity outboard. Analysis of the synchro pair work-up flights showed that the high frequency buffet response accounted for approximately 75% of the total inboard tailplane damage and almost 100% of the damage outboard. Further investigation suggested that the rate at which control surfaces are operated to initiate and recover from manoeuvres has a considerable impact on the intensity of the buffet and hence the fatigue damage. Continuous and rapid correction of position during the display to maintain the high standards of precision flying that the Red Arrows are renowned for therefore incurs large variations in tailplane damage rates.

Until the OLM had measured the true impact of Red Arrows operations the tailplane inspection intervals for these aircraft were more penalising than for Tactical Weapons usage but by a factor based on engineering judgement rather than analytical science. In light of the findings from the OLM it was recommended that a continuous, active Fatigue Monitoring System (FMS) was installed on Red Arrows aircraft to track more accurately the accrual of tailplane fatigue damage in order to optimise the inspection regime. This is dealt with in more detail later in this paper. As an interim measure tailplane inspection intervals were reviewed to account for sortie type and formation position by equating Red Arrows flying hours to equivalent Tactical Weapons Unit (TWU) flying hours. In one example a tailplane's usage was increased from 1058 Red Arrows flying hours to 4613 equivalent flying hours of TWU.

Red Arrows Fatigue Monitoring System (FMS) & Rolling OLM Programme

Pilots become familiar with the particular handling qualities of a given aircraft and are reluctant to change during the display season. However, the need for careful fatigue management of the Red Arrows fleet is apparent and annual rotation of aircraft about the display formation was recommended. Aircraft rotation had been performed before the OLM programme but as a result of the programme findings formal methods have been developed, principally based on rear fuselage/empennage fatigue damage rates.

The introduction of an additional Sortie Profile Code (SPC) for use by Reds 6 and 7 during the annual synchro pair work-up was also recommended to identify this flying which had previously been logged under SPC 21 (Continuation Training). The RAF accepted this advice and SPC 26 is now in use.

The principal recommendation from the flight trial was for the introduction of an aircraft dedicated, active Fatigue Monitoring System (FMS) to track more accurately rear fuselage/empennage damage. As has been highlighted previously, the structural integrity of Red Arrows tailplanes is maintained by a series of inspections. Due to the significant variations in fatigue damage recorded by both fin and tailplane strain gauges, particularly the latter because of vibration loading, the only reliable method of ensuring safe inspection intervals was considered to be by monitoring.

A subset of the OLM empennage strain gauge fit was defined for the FMS comprising two tailplane gauges and a single fin gauge. The proposal recommended that the strain gauge data should be recorded on airbourne solid state memory modules in full time history format. The memory modules hold up to 5 hours of data before download to the Analysis Groundstation (AGS) is required, equating to one download per day during periods of intense training and display. To aid the process of verifying the quality of the strain gauge data during AGS checks a subset of the OLM aircraft parameter fit was also defined. This included normal acceleration which enables the monitoring of g-manoeuvring outside the bounds of the aircraft's g-meter to be made. It also included airbrake deployment which is significantly higher in the Red Arrows than for other RAF Hawk usage and is currently unrecorded.

It had been intended to fit the FMS as a Trial Installation (TI) on a single aircraft (one of the synchro pair) mid-2000 with the remainder of the fleet, or as a minimum the rear of the formation, receiving the FMS from the end of 2000. However, the RAF recognised that for existing tailplanes the FMS could only optimise the existing inspection intervals and would not extend the life of the tailplane. Therefore, to ensure structural integrity the RAF is considering replacing the existing tailplane standard operated by the Red Arrows with the new standard Hawk tailplane.

The new standard tailplane addresses the known problems of the original standard and was designed to account for the effects of buffet loading. To demonstrate the structural capability of this tailplane a Tailplane Full Scale Fatigue Test was designed and is currently in progress. This applies 'manaoeuvre' and 'buffet' load blocks independently via hydraulic jacks and vibration shakers respectively. The tailplane is mounted on a rear fuselage section with stub fin to ensure the test is fully representative. Figure 13 shows the TFSFT during the application of a block of 'buffet' loading.

The introduction to the Red Arrows fleet of the new standard tailplane, even though it is considered 'inspection-free', does not negate the need to account for the accrual of tailplane fatigue damage. The most accurate method of achieving this remains with an aircraft dedicated Fatigue Monitoring System similar to that described above. This would be in line with the RAF's intended move towards more sophisticated fleet wide fatigue monitoring systems [5] and is also similar in concept to the Health Usage Monitoring System (HUMS) fitted to the latest Hawk standard, the Lead-In Fighter (LIF) [6].

To support the introduction of the new tailplane the Red Arrows OLM aircraft, XX260, will be fitted with a fully instrumented new standard tailplane during the first half of 2000. It is anticipated that a flight programme comprising approximately 150 flights will be completed during the second half of the year and this will enable a thorough assessment of structural life derived from the TFSFT for the tailplane operated in a Red Arrows role. This will include the synchro pair work-up for the 2001 display season.

Finally, it is noted that the RAF intend to maintain the OLM system on XX260 with the intention of performing a rolling OLM programme every other year. This will enable changes in the display to be assessed and will increase the database of information with which to support the structural integrity of the Red Arrows Hawks.

Conclusions

This paper has outlined the principal aims and findings of the Red Arrows Hawk Operational Loads Measurement programme conducted between 1997 and 1999. The OLM information forms an important database for maintaining the structural integrity of the RAF's aerobatic team.

The programme demonstrated the adequacy of the wing and fuselage fatigue meter formulae, but highlighted shortcomings of the existing fin fatigue formula which have now been addressed. Significant rear fuselage and tailplane damage rates, particularly for the rear formation aircraft, prompted the proposal for an active Fatigue Monitoring System. This continues to be reviewed in light of recent RAF decisions regarding tailplane standard and rear fuselage modifications.

The OLM programme highlighted the importance of Red Arrows fleet fatigue management to ensure even lifing across the fleet and this is best achieved through regular aircraft rotation about the formation. This is particularly relevant for the synchro pair aircraft whose usage is most demanding.

The Red Arrows OLM programme has been a proven success and the continued instrumented operation of the aircraft in a rolling OLM role enables both BAE SYSTEMS and DERA to offer the RAF comprehensive support to the Red Arrows.

Acknowledgement

The authors wish to express their thanks to all those who have contributed to the success of the Red Arrows Hawk Operational Loads Measurement programme and for the support that has been received during the preparation of this paper.

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References

1. Wg Cdr M.J. Kilshaw, "The Hawk Life Extension Programme - Part One", *Air Clues,* pp.467-472, December 1996.

2. R. Shooter, "Hawk - Structural Life Extension", *Proceedings of CEAS Forum Life Extension - Aerospace Technology Opportunities*, Cambridge, UK, March 1999.

3. J. O'Hara, "The Evolution of the BAe Hawk and its Structural Clearance", *17th Symposium of the International Committee on Aeronautical Fatigue,* Stockholm, Sweden, June 1993.

4. D.M. Holford, "Operational Load Measurement on Military Aircraft", *Keynote Address for Institute of Mechanical Engineers Seminar 'Operational Load Measurement',* Bristol, UK, March 1993.

5. Sqn Ldr S.R. Armitage & D.M. Holford, "Future Fatigue Monitoring Systems", *Proceedings of the 19th Symposium Aircraft Integrated Monitoring Systems,* Garmisch-Partenkirchen, Germany, pp.495-510, May 1998.

6. N. McCusker & N. Penistone, "The Hawk Fatigue Monitoring System", *26th Meeting of the International Committee on Aeronautical Fatigue – Structural Integrity for the Next Millennium,* Bellevue, USA, July 1999.

Figure 1 - The Synchro Pair (Reds 6 & 7) in Close Formation

Figure 2 - Red Arrows Diamond Nine Formation

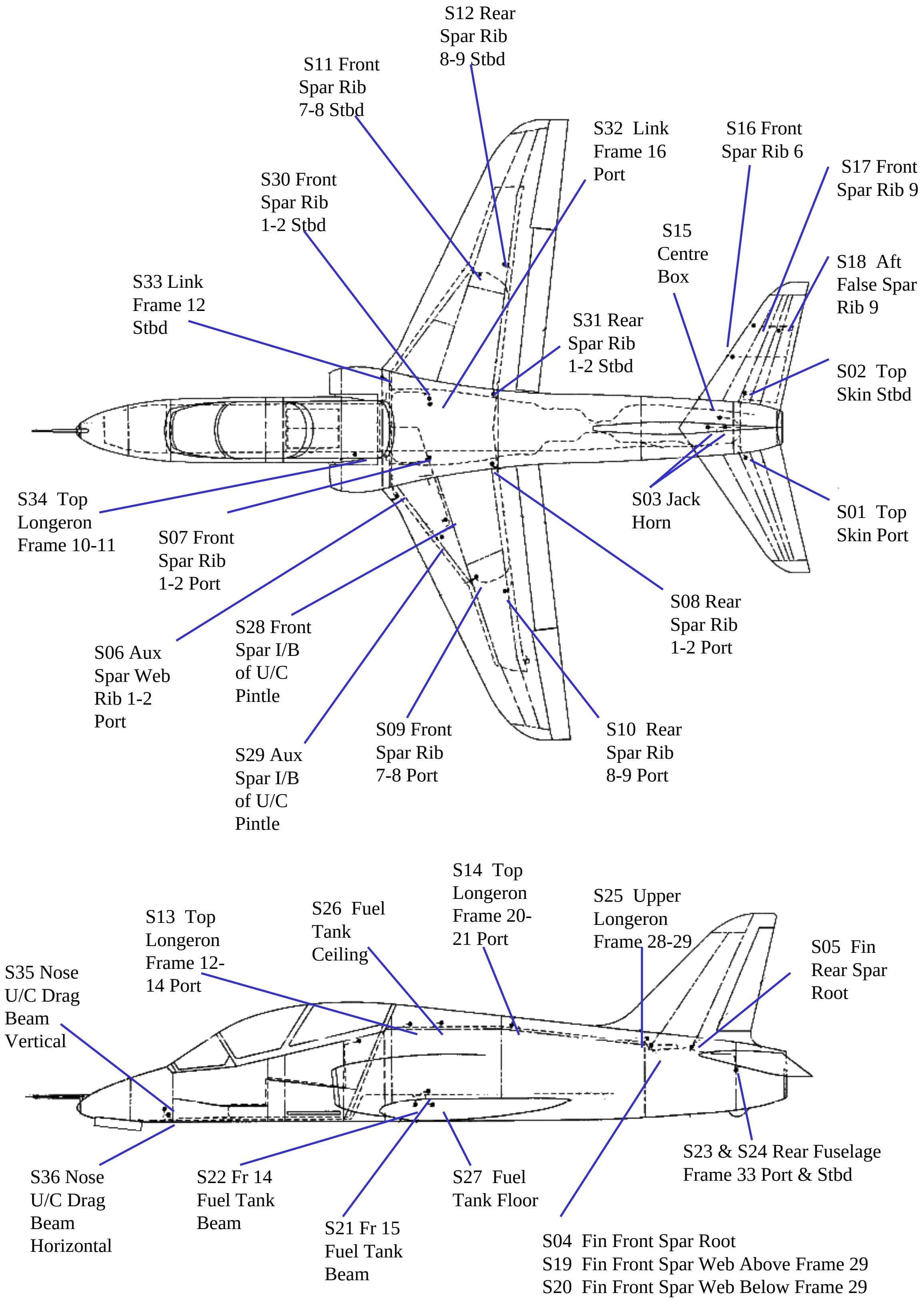

Figure 3 - OLM Strain Gauge Fit

Figure 4 - The Hawk Life Extension Fatigue Test (LEFT)

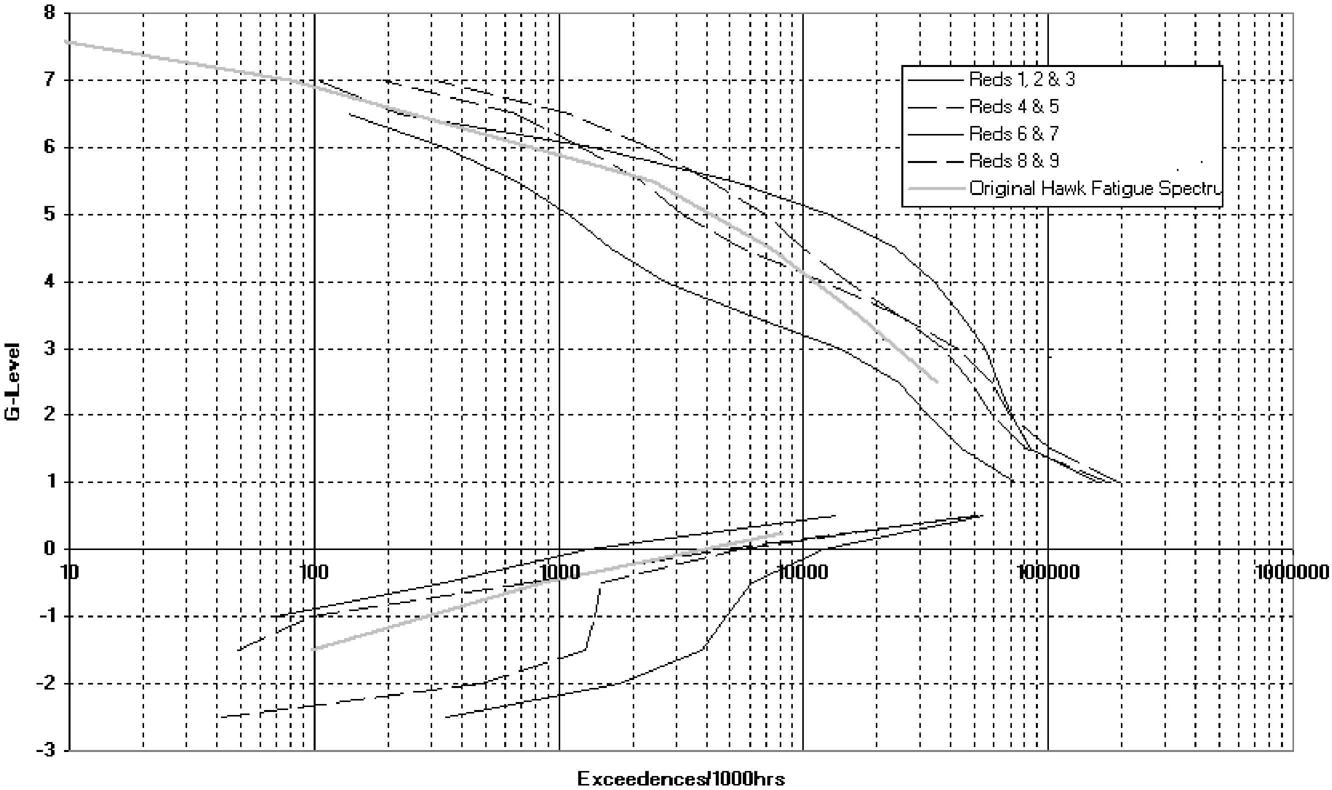

Figure 5 - Red Arrows OLM G-Exceedence Data Compared to the Original Hawk Design Requirements

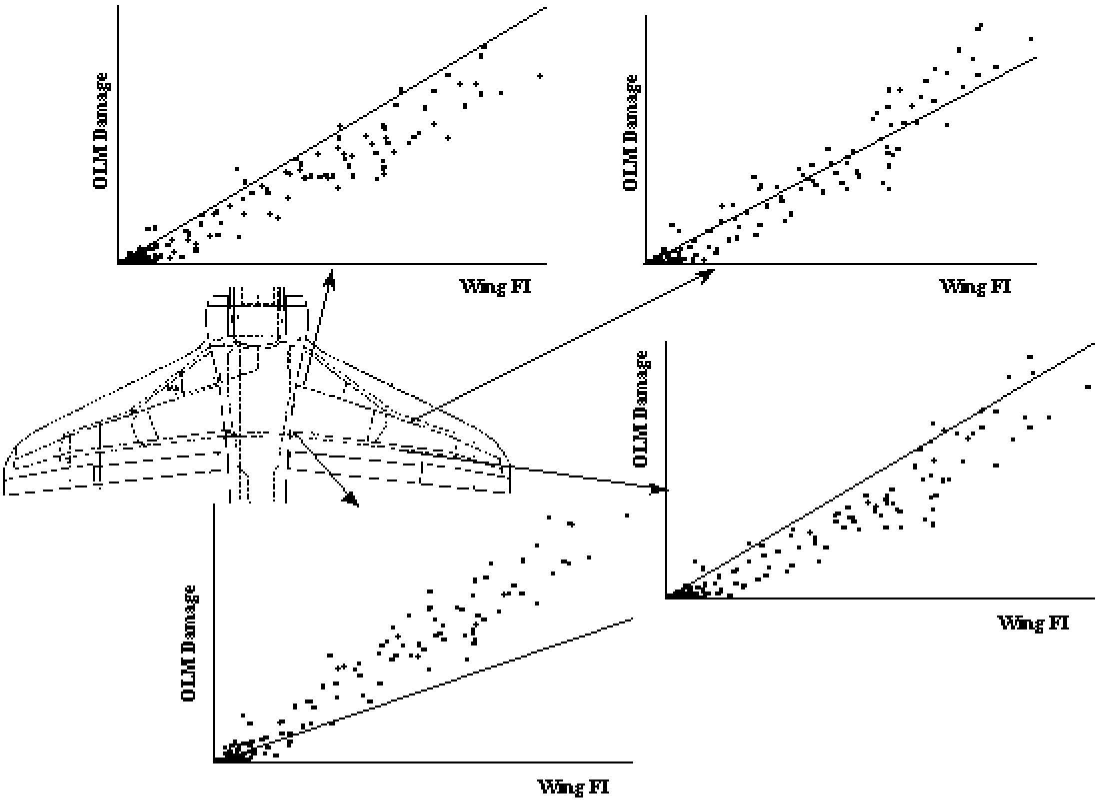

Figure 6 - Wing : OLM Damage v Wing Fatigue Meter Formula FI

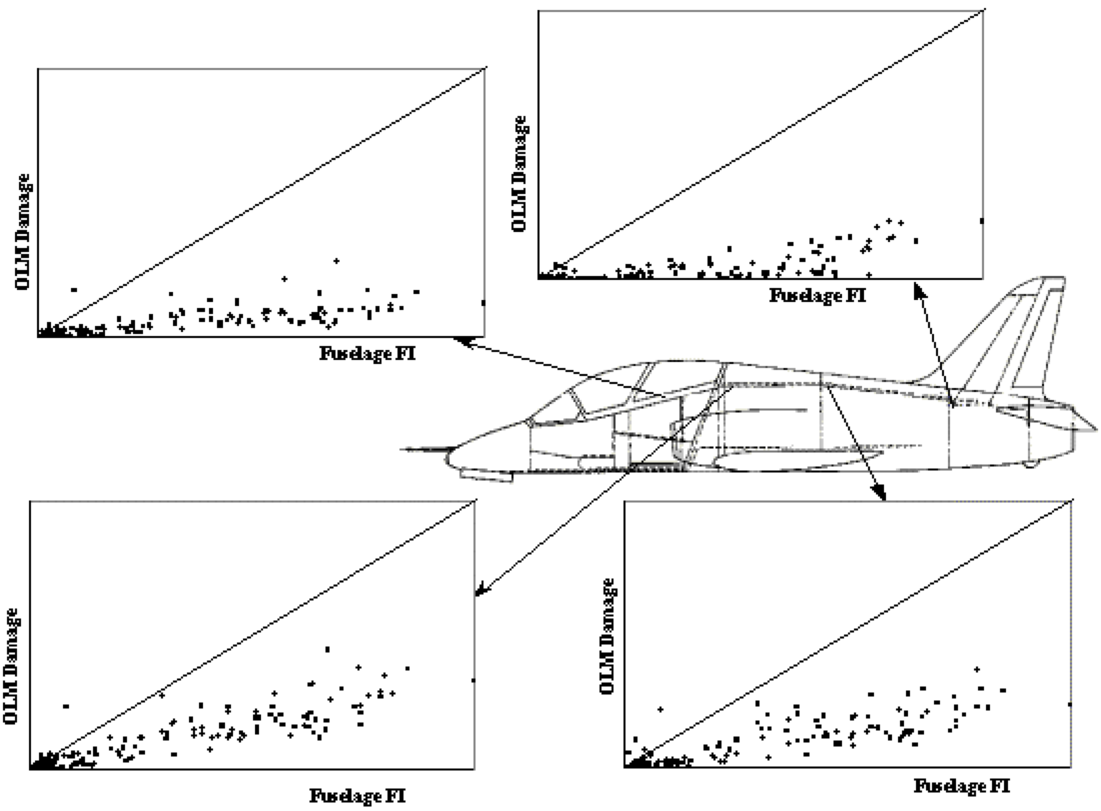

Figure 7 - Fuselage (Upper Longeron) : OLM Damage v Fuselage Fatigue Meter Formula FI

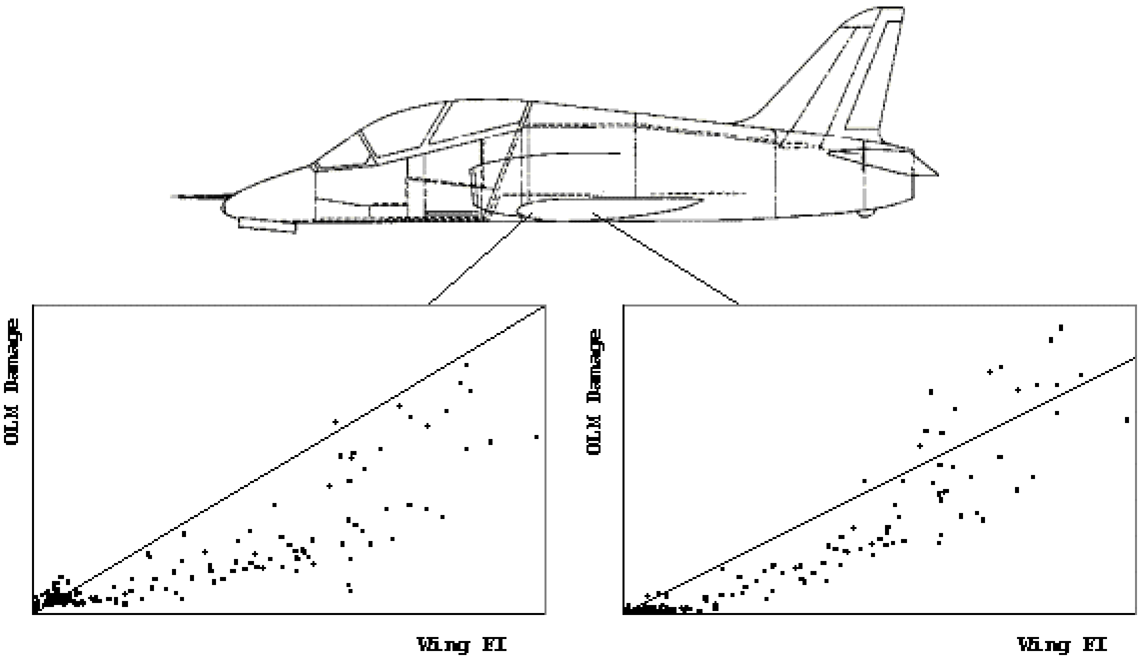

Figure 8 - Wing/Fuselage Links
OLM Damage (High Load Transfer S-N Curve) v Wing Fatigue Meter Formula FI

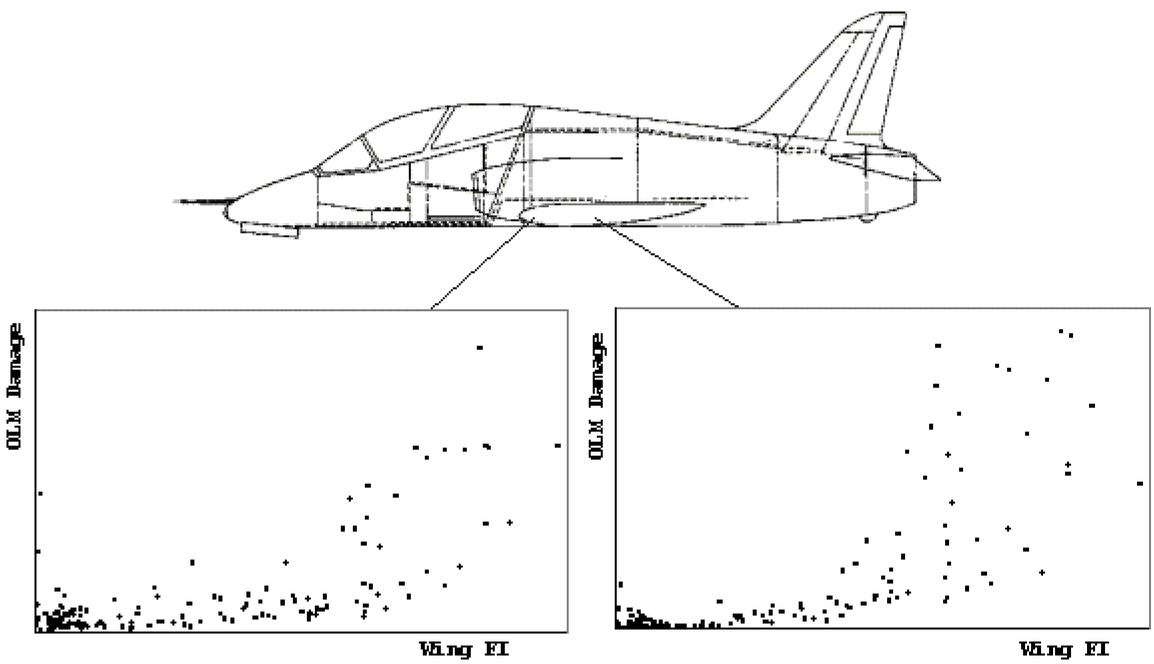

Figure 9 - Wing/Fuselage Links
OLM Damage (Lug Analysis) v Wing Fatigue Meter Formula FI

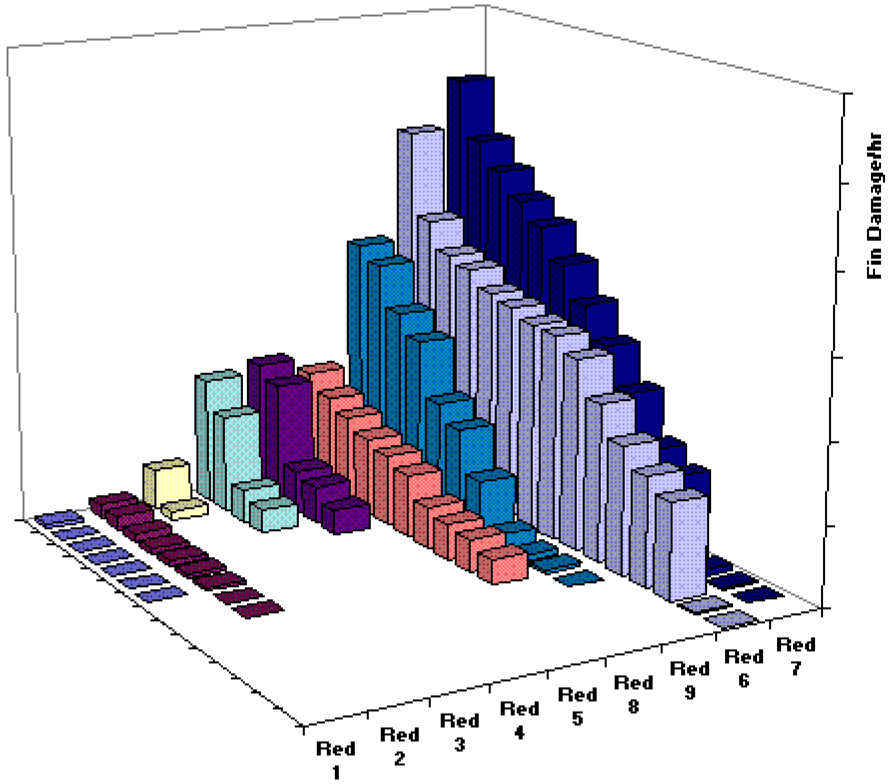

Figure 10a - Fin Damage/Hr by Formation Position for Main Display Season (No Transit Flights)

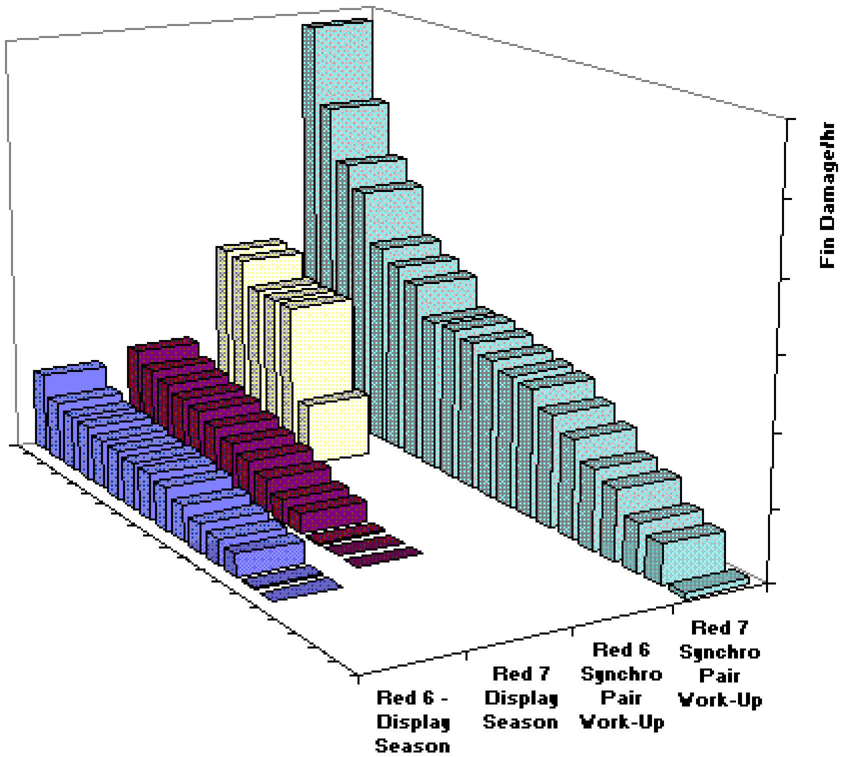

Figure 10b - Fin Damage/Hr by Formation Position for Main Synchro Pair Work-Up (Main Display Season Data for Reds 6 & 7 Shown from Figure 10a for Comparison)

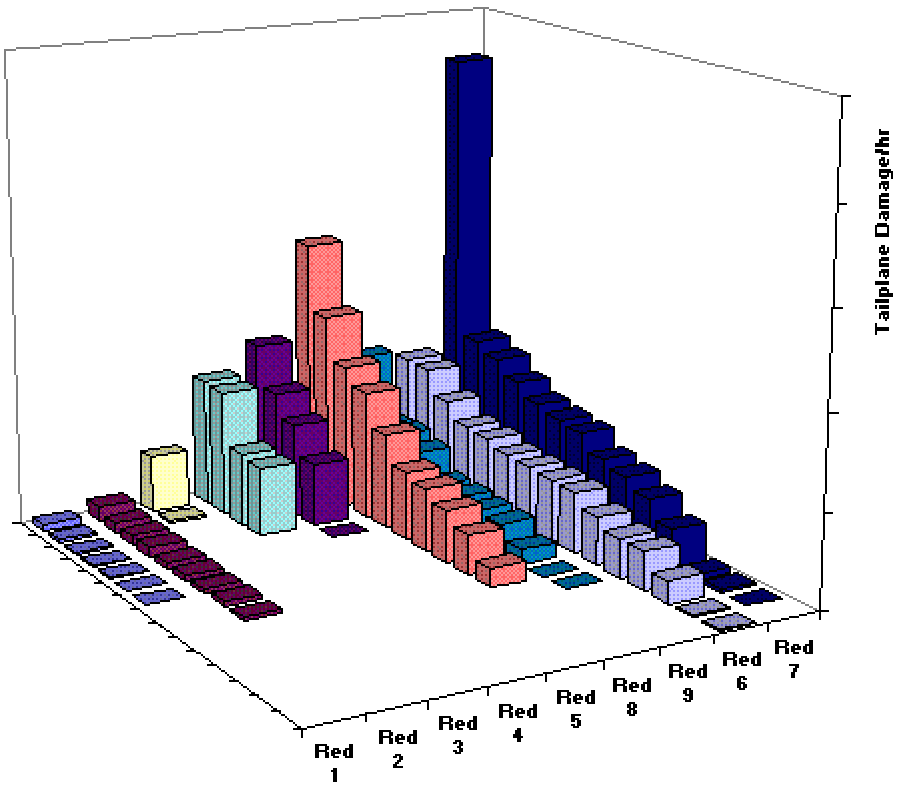

Figure 11a - Tailplane Damage/Hr by Formation Position for Main Display Season (No Transit Flights)

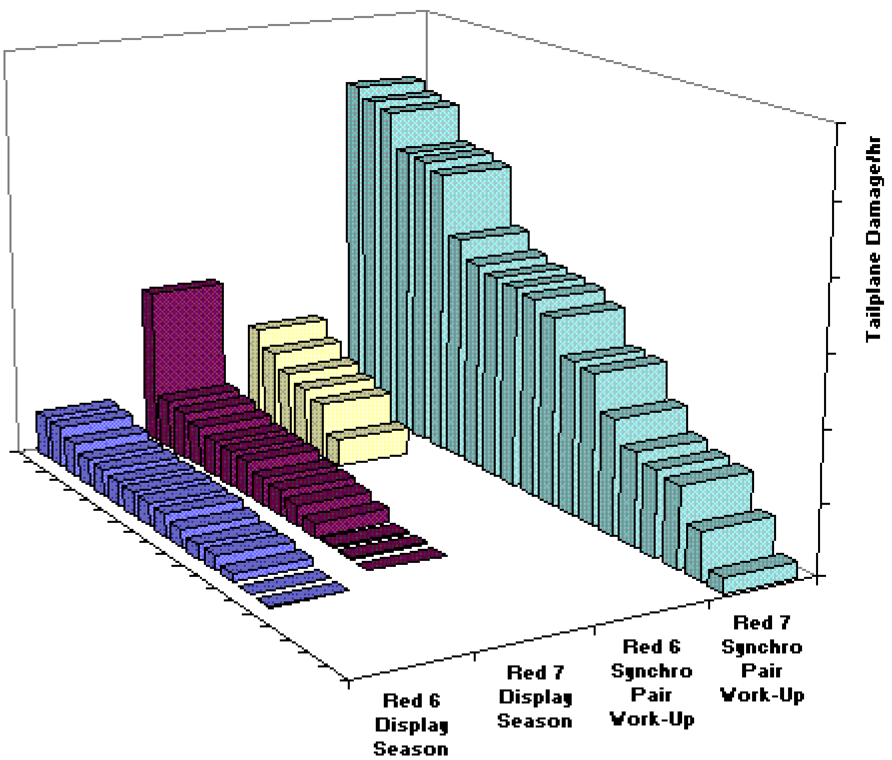

Figure 11b - Tailplane Damage/Hr by Formation Position for Main Synchro Pair Work-Up (Main Display Season Data for Reds 6 & 7 Shown from Figure 11a for Comparison)

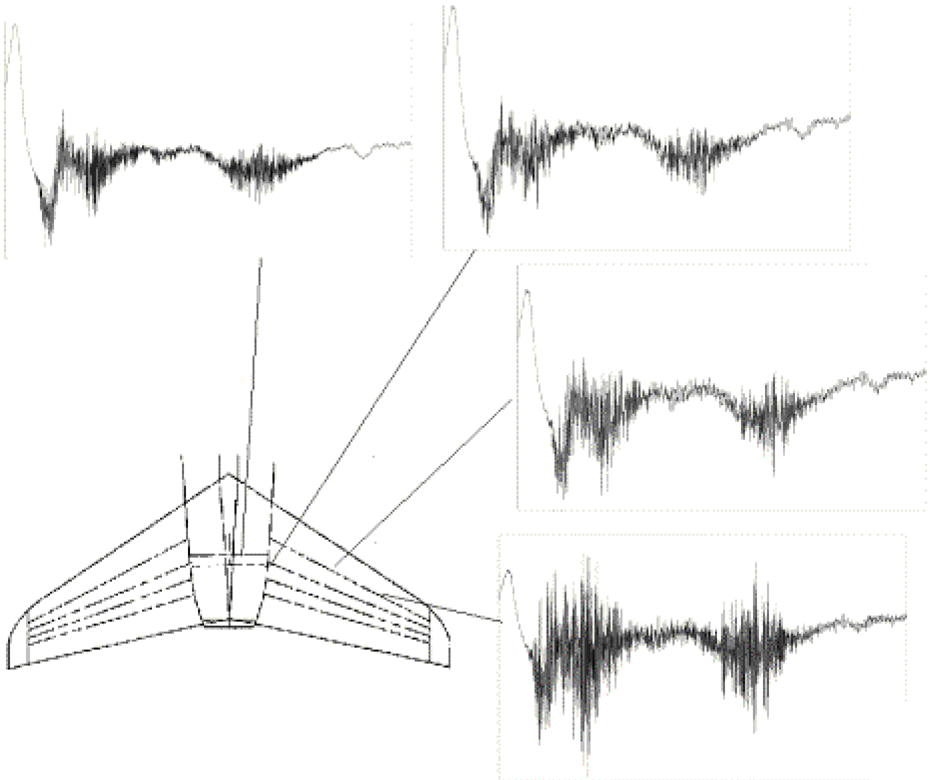

Figure 12 - Red 6 Tailplane Strain Gauge Response from 6g Formation Break Manoeuvre

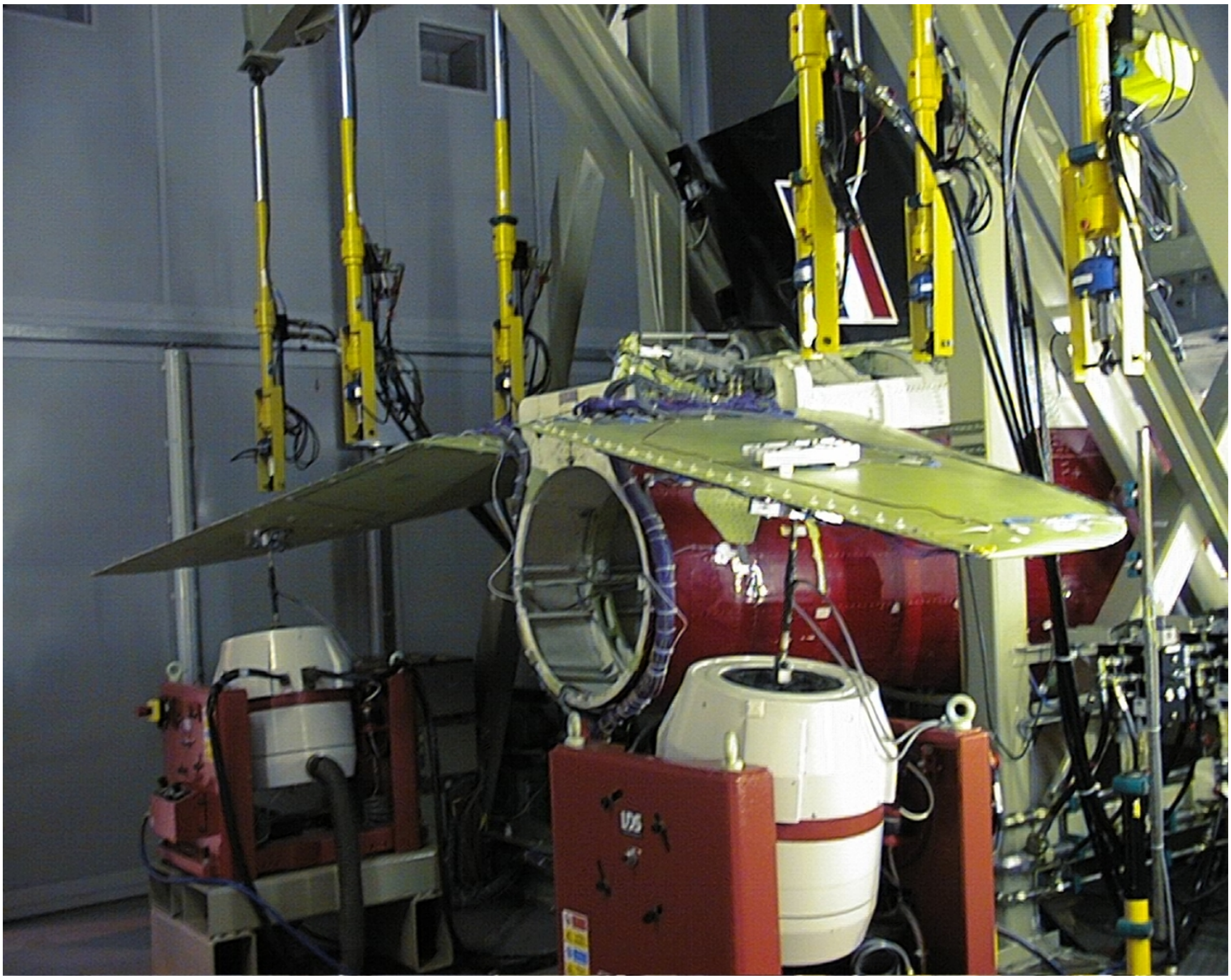

Figure 13 - The Hawk Tailplane Full Scale Fatigue Test

THE AIRFRAME FATIGUE LIFE MONITORING CONCEPT FOR THE WS TORNADO IN THE GERMAN AIR FORCE

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Table of Contents

List of Abbreviations

A/C	Aircraft
FCA	Fatigue Critical Area
GAF	German Air Force
IAT	Individual Aircraft Tracking
OLMOS	Onboard Life Monitoring System
SAT	Selected Aircraft Tracking
SHMS	Structural Health Monitoring System
TAG	Touch And Go
TAT	Temporary Aircraft Tracking
W	Weight
WS	Weapon System

1. INTRODUCTION

The general aim of a Structural Health Monitoring System (SHMS) is the prediction of the remaining fatigue life of each individual A/C of a fleet. Since especially military aircraft are exposed to a wide range of loads, the structural health is one of the substantially limiting factors for the in-service time.

Generally there are different levels of effort imaginable to realise a SHMS. Low effort on SHMS results in low costs but a high loss of fatigue life. In this case high safety factors have to be applied or high in-service loads have to be assumed so that structural failure can be excluded. The benefit of high effort is the maximum exploitation of the fatigue life but at the same time the costs for SHMS are higher.

Increasing investments for developing and producing a new WS have promoted the tendency to extend the in-serve time of existing WS. The initially planned in-service time is ensured by the experiences of a fatigue test, which is initiated in the design phase.
When extending the in-service time the experience of operational usage, which might differ from the assumptions made in the design phase, becomes more and more important. Changing boundary conditions, such as modified parameters of operational usage and modifications of the WS itself, require high accuracy in evaluating individual operational loads.

The TORNADO WS was introduced to the GAF in 1980. Right from the beginning of the in-service time the TORNADO was equipped with fatigue–monitoring devices to control the remaining fatigue life of critical fatigue areas detected during the fatigue test. Since that time the concept of fatigue monitoring has been modified going along with in-service experience and technical innovations. Considering the fact that also for the TORNADO WS a extension of the originally scheduled maximum flight hours in the GAF is in discussion, the importance of a reliable fatigue monitoring concept becomes evident.

2. THE APPLIED FATIQUE LIFE MONITORING CONCEPT

Generally a fatigue-life monitoring concept in the GAF consists of three major tasks. These are

- In-Service Measurement
- Mathematical Modelling / Evaluation of Damage Calculation
- Maintenance Planning

These major tasks are applied to all WS in the GAF, e.g. TORNADO, C-160 TRANSALL and F-4F. The expended effort differs, mainly dependent on the strutural complexity of a WS and the degree of the in-service loads.

The technical progress led to fatigue monitoring concepts with increasing accuracy in predicting the remaining fatigue life. As a result of decreasing costs and weight of data-storage capacity the amount of measured data to represent in-service flight conditions could be enhanced. The corresponding data-evaluation algorithms have gained in increased accuracy. This has resulted in a subsequent improvement of maintenance planning or, in other words, in a maintenance planning, which goes along with the actually occuring stresses of individual fatigue critical areas.

2.1 In-Service Measurement

The ‚In-Service Measurement' task for the TORNADO WS can be described as an integrated concept, consisting of three inter-related components. These are:

IAT (Individual Aircraft Tracking)
TAT (Temporary Aircraft Tracking)
SAT (Selected Aircraft Tracking)

All three components are essentially based on the acquisition of flight parameters, representative for external loads. The components differ with respect to the extent of the measured data.

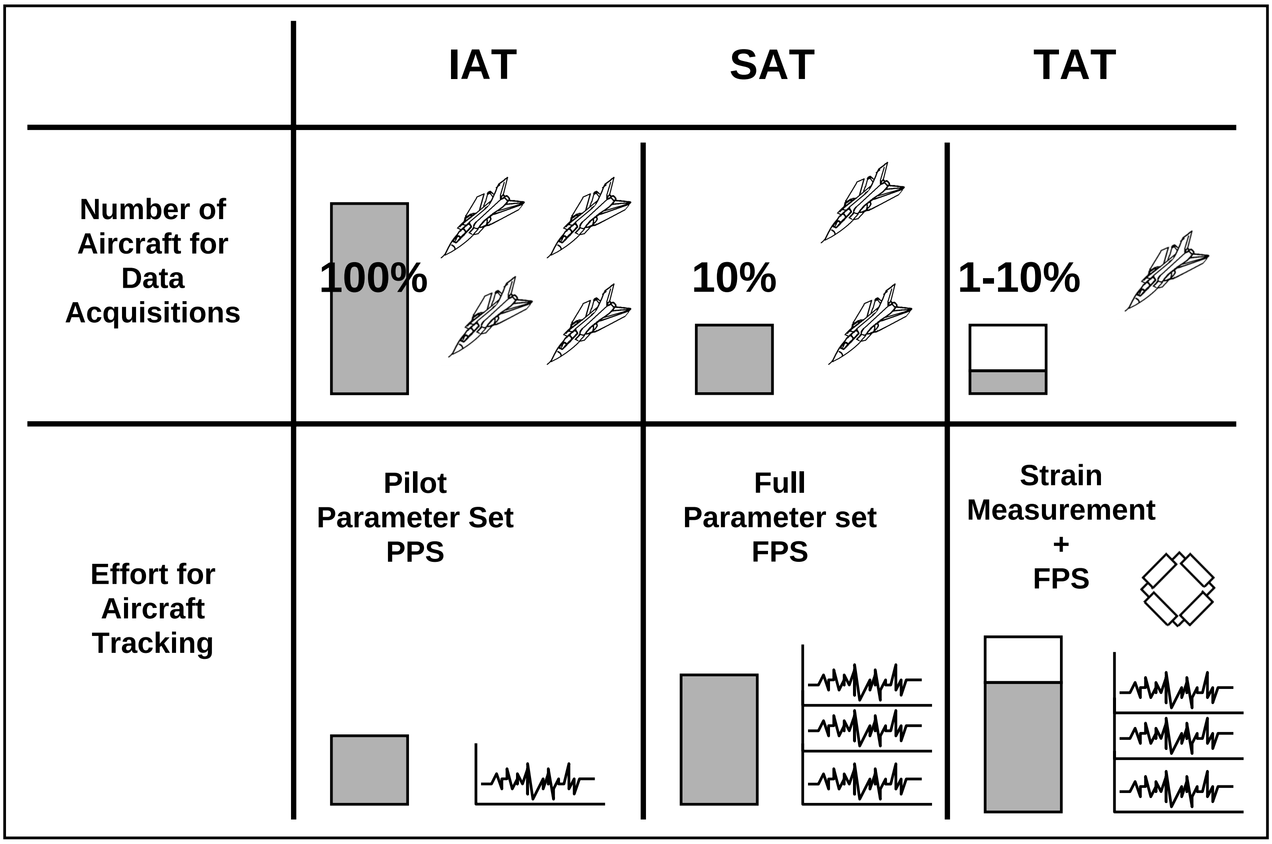

Fig. 1

As Figure 1 shows, in the scope of TAT the most comprehensive acquisition is performed whereas IAT records only a restricted amount of flight parameters.

Each A/C of the fleet is at least subjected to the Individual Aircraft Tracking (IAT). Selected A/C are additionally subjected to TAT or SAT. The general idea behind the concept in data acquisition is that the A/C, which belong to the restricted data acquisition component (IAT), benefit from the higher accuracy of the other components (SAT and TAT). This is an approach, which is cost-efficient and achieves simultaneously an overall high accuracy.

- Individual Aircraft Tracking (IAT)

  The IAT covers a measurement concept, which is applied to all A/C. A so called Pilot Parameter Set (PPS) is measured continuously and stored during operational usage.
  The PPS consists of the parameters N_z, weight, wing sweep angle and stores.
  The IAT contributes the data-basis for the calculation of the individual damage index of the FCAs.

- Temporary Aircraft Tracking (TAT)

  The TAT is applied to selected A/C. The measured data set is most comprehensive compared to the data-sets of the other components. It covers the measurement of the so called Full Parameter Set (FPS) and the Pilot Parameter Set (PPS), which is an inherent part of the FPS.
  In addition to the flight parameter sets FPS/PPS, in the scope of TAT also strain is measured. Strain gauges can be placed at various FCAs.
  The flight parameters and the simultaneously measured strain allow to adjust the coefficients of a flight-parameter-dependent mathematical set-up by matching with the actually measured strain.

- Selected Aircraft Tracking (SAT)

  The SAT is also applied to selected A/C. It covers the measurement of the Full Parameter Set (FPS) and the Pilot Parameter Set (PPS). The SAT is the connecting link between IAT and TAT.
  The PPS is measured for all A/C. Starting from a relative small parameter set the calculated strain is comparatively inaccurate. The FPS supplies more accurate results regarding the strain, but it is measured only for selected A/C.
  To transfer the benefit of a higher accuracy of FPS-based calculated strain a so-called 'transfer-function' is created. It arises from the comparison of the FPS-based calculated strain with the PPS-based calculated strain, which is carried out with the data obtained from the SAT.

As outlined above the SAT and TAT concept is realised only for a restricted number of A/C. The flight recorders used with SAT and TAT are distributed on a statistically representative basis throughout the individual squadrons. The simultaneously performed strain/flight-parameter measurements of TAT are cyclically repeated for the same FCA on several A/Cs.

2.2 Mathematical Modelling

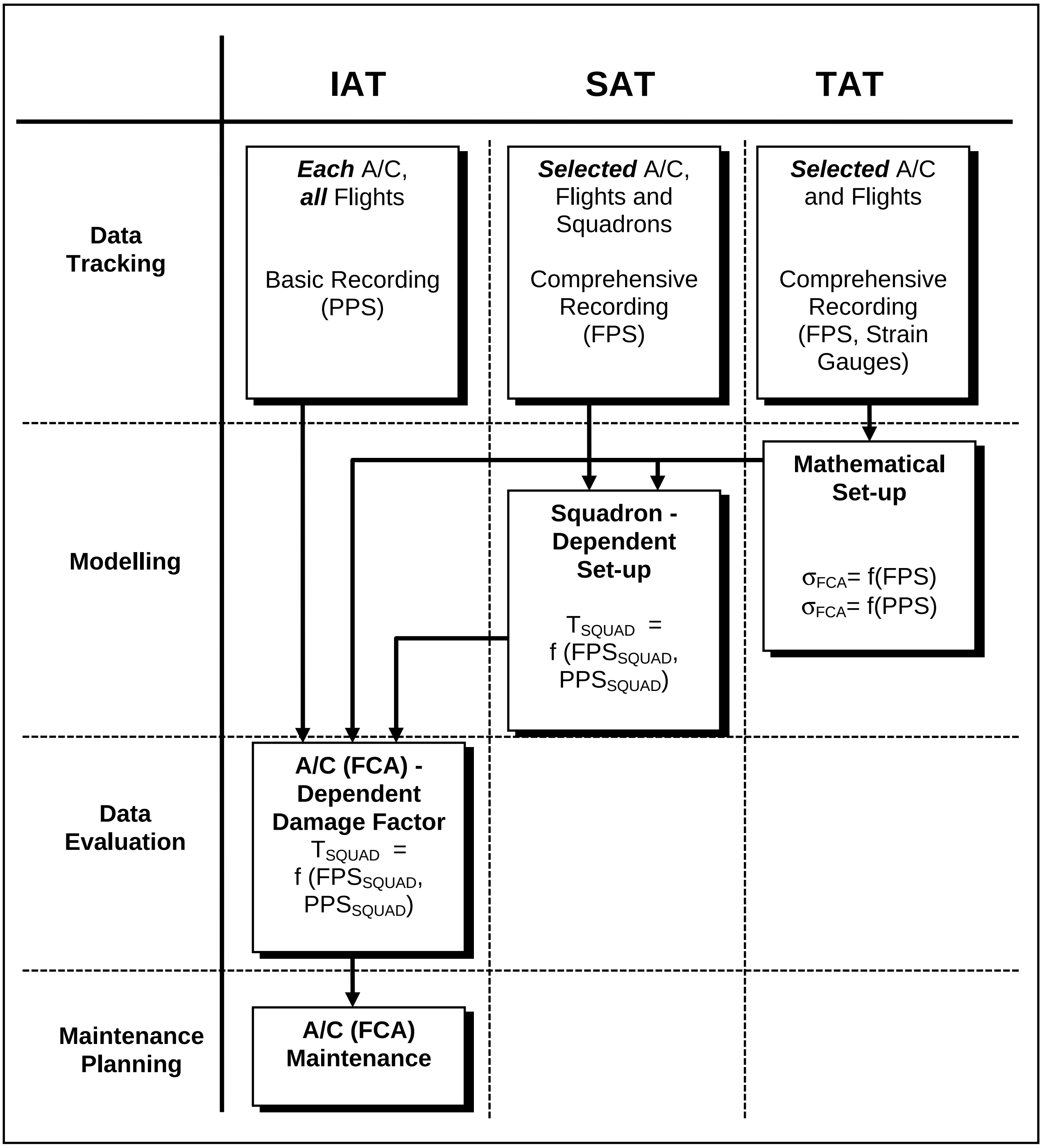

Fig. 2

As shown in Fig. 2, the precedence of mathematical modelling is coordinated with the concept of in-service data-tracking.
Results obtained from TAT- and SAT-derived data sets drop into the evaluation of the restricted parameter set of IAT. Because all A/C are at least subjected to the basic data tracking this is the bases for the fleet-embracing calculation of the damage factors and therefore this is also the basis for the maintenance planning.

The mathematical model in general should supply the occuring stresses and strains of each A/C at each FCA by evaluating the incoming flight parameters. The concept of the mathematical modelling has to be adapted to the ‘physical reality‘ by considering the system inherent characteristics and boundary conditions.
Since the comprehensive recording, which is more accurate with respect to the ‘physical reality‘, is restricted only to selected A/Cs the concept of modelling incorporates also statistical evaluation methods.

2.2.1 TAT-Based Mathematical Modelling

The data obtained from the TAT serve for the derivation of the functional relation between the measured flight parameters and the strain at individual FCAs. To choose an appropriate functional set-up, first the ‘physical reality‘ has to be considered.

The strains respectively stresses at particular FCAs depend on the deformation of the A/C structure. The deformation has to be discussed considering two main aspects.

First, the external forces and forces resulting from the inertia of masses are the origin of the occuring deformations. The forces can have various points of application depending on the particular maneuver the A/C is performing. Besides there are forces having a deterministic origin as well as forces occuring stochastically. E.g. forces resulting from TAGs are induced by pilot action and therefore of an deterministic origin, whereas forces resulting from gusts have a stochastic origin. Different ‚classes‘ of forces might have various influences on functional relations of the strain at a particular FCA.

The second aspect is that the A/C itself is not a fixed mechanical set-up. Especially the TORNADO is a WS, which can have various mechanical set-ups. The most conspicuous fact in this sense is that the TORNADO WS makes use of the wing sweep technology. Another aspect regarding the mechanical set-up of the TORNADO is the variable mass distribution. The TORNADO can operate carrying various external loads at different stations. Thus the TORNADO might carry the MW1 weapon, weighing more than 2 tons, under its fuselage or - during another mission - external tanks, weighing 1.5 tons maximum each, at the wings. It is obvious that various mechanical set-ups will have an influence on the occuring strain at particular FCAs.

When creating a mathematical set-up to describe strain as a function of flight parameters, the question arises how to implement the existence of different A/C mechanical set-ups or different classes of external loads resulting from clearly distinguishable sequences of operational A/C usage. In the concept for the mathematical modelling this is done by classification. That means that various classes of operational usage are introduced and for each class a set of functions

$$\sigma_{FCA} = f(FPS)$$
$$\sigma_{FCA} = f(PPS)$$

has to be derived.

Fig. 3 shows a selection of defined classes regarding the mechanical set-up of the TORNADO WS.

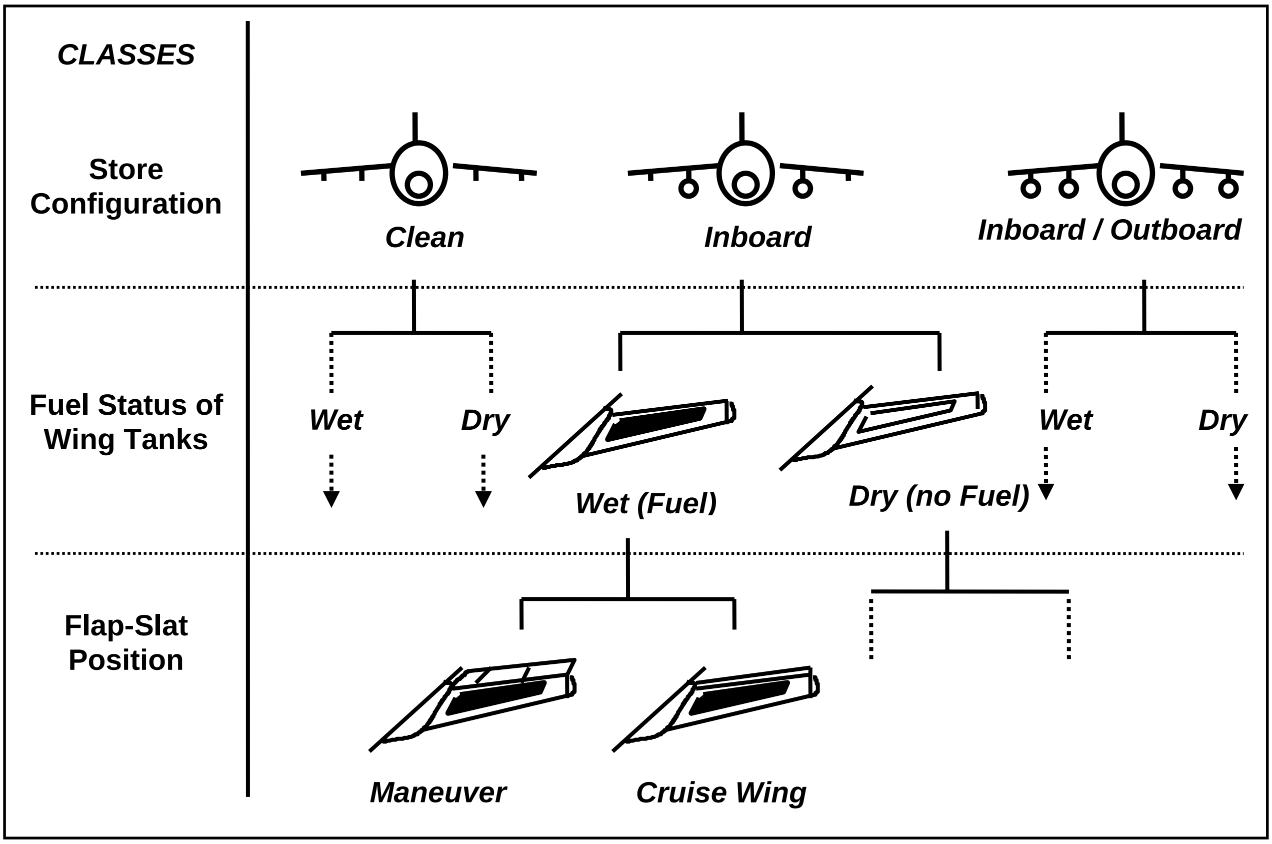

Fig. 3

The second step in formulating the TAT-according mathematical model is to find a mathematical set-up to adjust the coefficient by performing a regression analysis. The accuracy of the derived set-up can be proven by deriving the correlation coefficients between the measured and calculated strain values.

The dependency of strain on a flight parameter can be evaluated by performing correlation analysis. The vertical acceleration N_z is the most influencing flight parameter regarding the strain of an A/C structure. This is the reason why N_z in combination with the A/C weight is the essential flight parameter of the PPS.

Other flight parameters also correlate with strain. The extent of influence depends on the particular flight parameter as well as the location of the FCA. A mathematical set-up, which considers various flight parameters, is based on a selection of flight parameters of the FPS obtained with TAT. Table 1 outlines the parameter set, which is recorded in the scope of TAT and SAT.

No.	**Parameter**	**Sampling Rate / sec**	**No.**	**Parameter**	**Sampling Rate / sec**
1	Pressure Altitude	0.5	11	Inboard Spoiler Right	1.0
2	Calibrated Airspeed	0.5	12	Rudder Position	2.0
3	Normal Acceleration	16.0	13	Wing Sweep Angle	0.5
4	True Angle of Attack	2.0	14	Primary Strain Gauge	16.0
5	Roll Rate	8.0	15	Secondary Strain Gauge	4.0
6	Pitch Rate	4.0	16	Flap Position	1.0
7	Yaw Rate	2.0	17	Slat Postion	1.0
8	Taileron Pos. Left	4.0	18	Fuel Remaining	1.0
9	Taileron Pos. Right	4.0	19	Stores Configuration	4.0
10	Outboard Spoiler Left	1.0	20	Oleo Switch	0.5
			21	Identification Data	1.0

Table 1

2.2.2 SAT-Based Mathematical Modelling

The Mathematical Modelling in the range of SAT is based on the functional set-ups obtained from TAT.

The functional set-ups which rely on the FPS are of a relative high accuracy, whereas the other set-ups based on PPS do not fully comprise the influences of maneuvers which are not or not substantially N_zW - determined. These maneuvers are not considered in the results of the PPS-based function but they are considered in the FPS-based function. Therefore the PPS-based results will always differ from the FPS-based results depending on the occurence of not N_zW-determined maneuvers.

The TORNADO SHMS assume that particular maneuvers are statistically linked to specific tasks of different squadrons. Therefore, the deflection of the FPS-based results to the PPS-based results should be typical for a particular squadron.
To benefit from the higher accuracy of a FPS-based result a squadron-specific transfer function is derived so that less accurate results of a PPS data-set can be transformed to a higher accuracy considering the characteristics of squadron-specific operation usage.

The target of SAT is to derive squadron-specific transfer functions which determines the relation of FPS-based results to PPS-based results.

SAT- and TAT-based mathematical modelling supplies functions which allow to derive strain at particular FCAs proceeding from the PPS data. TAT contributes the general functional set-up whereas SAT contributes the squadron-specific influences of the non N_zW-determined maneuvers.

2.3 Maintenance Planning

First experiences regarding fatigue critical areas (FCA) were obtained from the Major Airframe Fatigue Test (MAFT). For the design of this test a common load spectrum was defined, which covered the most stringent requirements of the participating nations. The results of the fatigue test provided the basis for fatigue life assessment in service. The test program revealed the maximum number of safe-life test hours of various fatigue critical areas.

Design improvements, to modify the FCAs, had been derived from the fatigue test results and - as far as feasible - implemented during manufacturing of production of the A/C. Due to the overlap of fatigue testing and A/C manufacturing, this was not always possible. Besides, it had to be understood that not all fatigue critical areas could be identified by ground testing. Consequently, some retrofit modifications have to be embodied during in-service time for extension of the fatigue life.

The time of implementing a structural modification is identified by fatigue life monitoring. In case of the FCAs, identified by the fatigue test, the fatigue consumption depends on the usage spectrum of the individual A/C. Test hours of the fatigue test do not follow a simple correlation with the flight hours of the fatigue test. Therefore, it is necessary to convert the fatigue damage in individual flight hours for the derivation of the point of embodiment for the modification.

Figure 4 shows schematically the major structural modification packages applicable to the GAF TORNADO fleet and the standardised time of embodiment related to the Design Life Limit. The Design Life Limit (FH) represents the number of admissible test hours of the fatigue test.

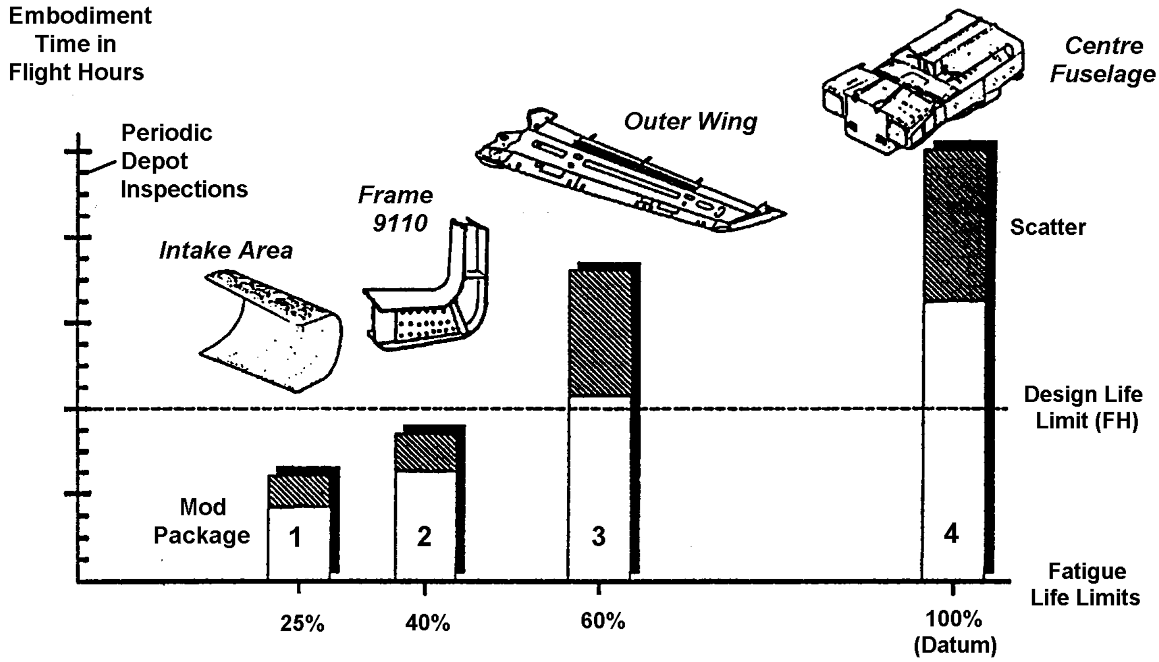

Fig. 4

The respective flight hours are represented by the bars of the chart. E.g. the modification of the centre fuselage has to be embodied, when its fatigue life, which shall correspond with 100% of the fatigue life according the Design Life Limit, is consumed. As it is seen a higher number of flight hours compared to the test hours can be obtained before modification becomes necessary. This means the consumption of fatigue life during operational usage is not as high as the fatigue consumption during testing. The hatched area of the bar indicates the difference between the A/C with the hardest and the A/C with the softest operational spectrum of the fleet.

3. FATIQUE LIFE MONITORING – A DYNAMICAL PROCESS

3.1 Modification of the TORNADO WS

The preceding explanations have described the actual concept of fatigue monitoring for the TORNADO WS. Since WS – and especially A/C – have a long lasting operational time, they are subjected to varying boundary conditions. The technical development is preceding while economical and political boundary conditions are changing. All these aspects might influence the subsequent development of a WS. The SHMS is also involved in that process. On the one hand it supplies data for the estimation of the options of future engagement. One the other hand the SHMS itself is subjected to the implementation of innovations for improvement and adaptation.

The development of the TORNADO WS started in early 1970. The maiden flight of the first prototype was in 1974. The first production aircraft was delivered in 1980. Thus, the oldest A/C have been in operational usage for about twenty years.

Figure 5 shows the demands of a 'developing' WS to a Fatigue Life Monitoring System in general. At the beginning of a WS life there is the design stage. The expected fatigue life determines the conditions for fatigue testing. During testing the fatigue critical areas (FCAs) have to be defined. Unexpected structural failure may occur, which results in the definition of structural modification.

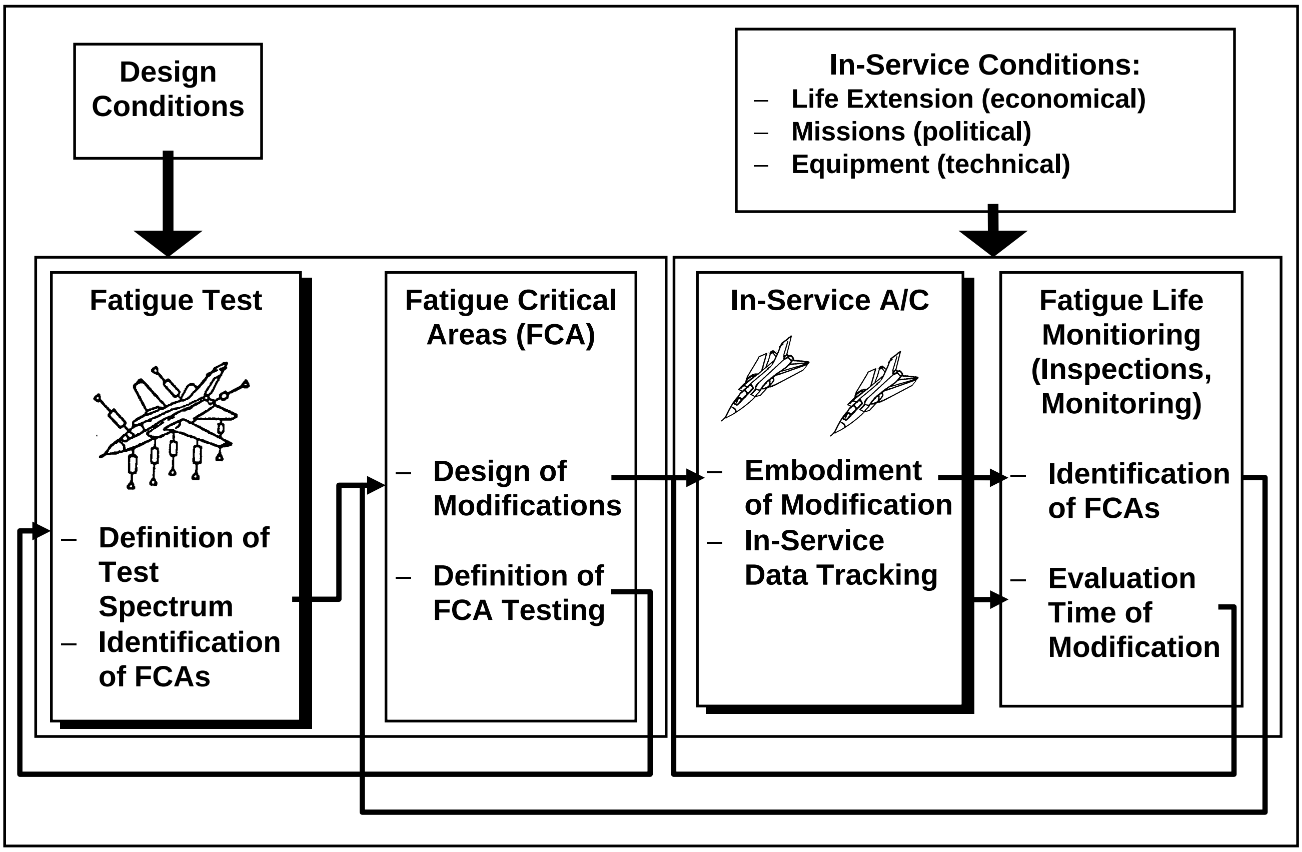

Fig. 5

If the production of the A/C already has started, the modification cannot be implemented to all A/C during the production stage and will be implemented in the in-serves stage when fatigue life monitoring indicates that the fatigue life of the concerning FCA has been consumed. This reveals what Figure 5 shows in general. Fatigue Life Monitoring is not a matter of a steady state procedure applied to similar TORNADOs lasting the whole operational time of the weapon system. It is rather a recursive process, where the different A/C have a different structural status. Modified former FCAs may not have to be monitored any longer. During in-service life new FCAs may be detected, tested and included in the monitoring procedure. Even modifications which were introduced during operational time could lead to modified diffusion of stress which results in a new FCA at different locations.

3.2 Innovation in Fatigue Life Monitoring

The approach with the highest accuracy to determine the consumed fatigue life would be to attach strain gauges at each FCA of each A/C just evaluating the occuring stresses directly. But - as indicated above - this was not a practicable solution when the WS TORNADO was introduced.

At the beginning of the fatigue life monitoring of the WS TORNADO not all FCAs had been detected. The knowledge about critical fatigue areas has increased with increasing in-service time. Besides, the status of a fatigue life monitoring system is influenced by economical and technical boundary conditions of its present time. E.g. when the WS TORNADO was introduced the data storage capacity had been restricted due to costs and weight aspects. Nowadays storage capacity is available at a reasonable price and storage devices have assembly dimensions and weight which do not cause problems to application in military A/C.

When developing a SHMS of an existing WS the following general aspects have an influence:

The SHMS development of an existing WS

- is based on results obtained from its own in-service experience.
- has to embrace new technical facilities while considering also cost aspects.
- has to be incorporated into the existing SHMS which was initially configured years ago.

The SHMS which is realised for the TORNADO WS in the GAF is a flight-parameter-based concept. As outlined above, the occuring stress respectively strain is derived from flight parameters. This concept is open to a gradual refinement during in-service time.

Figure 6 outlines the different data storage devices which were introduced to the TORNADO WS since its first delivery in 1980.

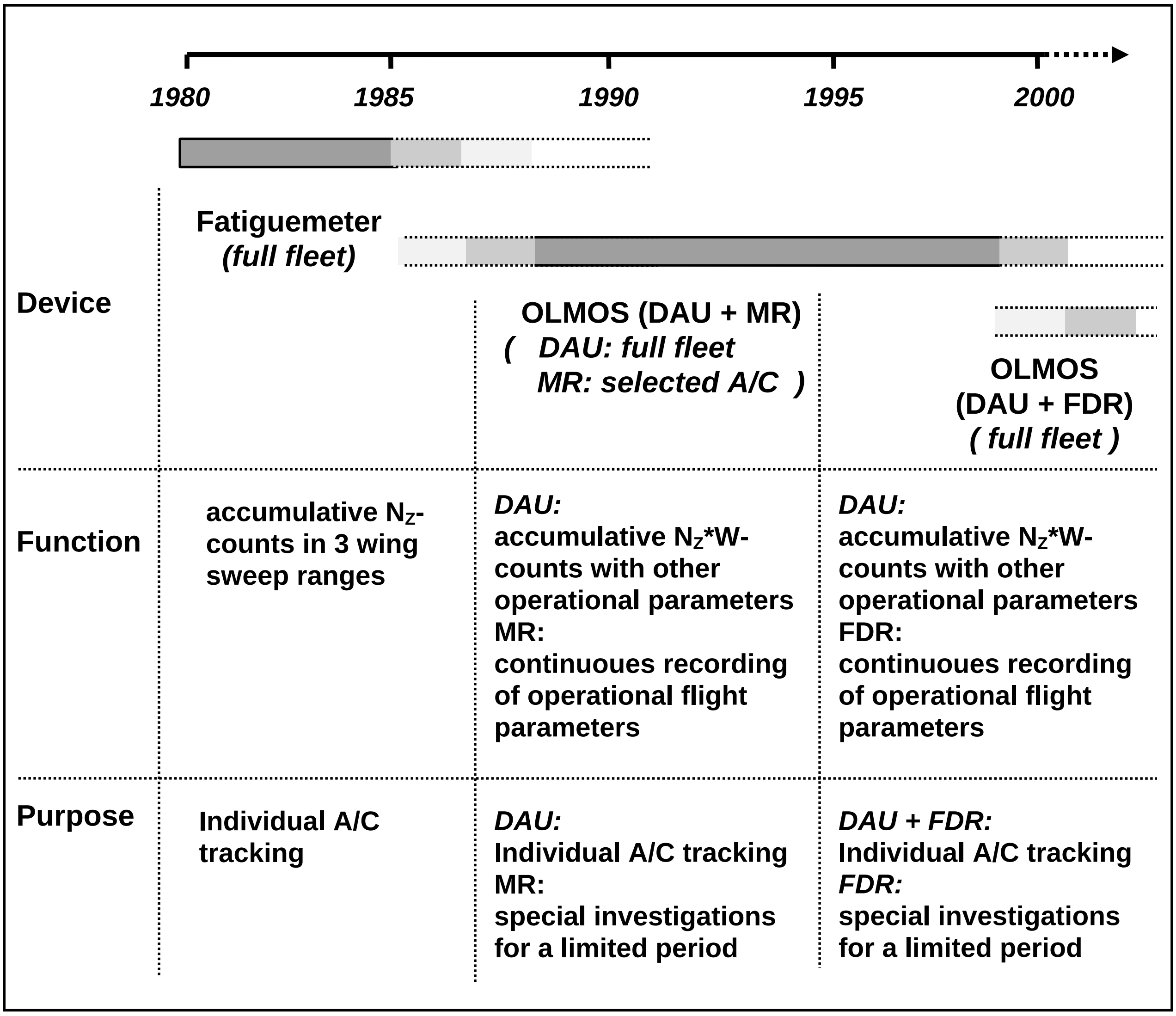

Fig. 6

The early GAF TORNADOs were equipped with the so-called ‚Fatiguemeter‘, which records the cumulative N_z classes within three different wing sweep ranges. As outlined above, this ‚basic‘ classification meets the requirement of a subdivided evaluation for various ‚mechanical set-ups‘ of the TORNADO.

Since the TORNADO is a WS with a wide spectrum of operational usage, the refined concept of fatigue monitoring considered also additional parameters concerning the mass distribution among the A/C structure as well as various aerodynamic configurations.
Thus, the concept using the fatiguemeter was replaced by the so-called OLMOS concept (Onboard Life Monitoring System). OLMOS and its attached technical devices are still involved in the present evaluations.
The data for IAT are stored in the DAU (Data Acquisition Unit). The device stores the cumulative Nz*W-classes depending on the different A/C-set-ups. These ‘set-up-classes‘ consider the wing sweep angle, the flap and slat position, the external store configuration and the fuel-level in the wing-tanks.

Within the scope of TAT a comprehensive recording of flight parameter is performed. Data are stored by a special device, the Maintenance Recorder (MR). The data-storage capacity is sufficient to store time sequences of various flight parameters (ref. Tab. 1).

The most recent development in data evaluation aims at a comprehensive data recording for each A/C. The new data storage device, the Flight Data Recorder (FDR), was introduced to the first A/C in the GAF in 1999 and substitutes the Maintenance Recorder. Since the FDR shall be installed in each A/C this opens new prospects to a further development of the data evaluation algorithms.

3.3 Refinement of Data Modelling

The data modelling aims at the calculation of the remaining fatigue life of the FCAs performed for each individual A/C in the GAF. As outlined above the mathematical model makes use of analytical and statistical methods. The degree of implementation of statistical methods depends on the extent of the in-service measurements. This means that the more data are measured the more one can dispense with statistical probability.

When relying on statistical results the mathematical modelling has to guarantee that the evaluated fatigue parameters meet the reality with a high probability. Thus, when statistical results are implemented the mathematical model relies on a ‘save side‘ philosophy potentially leading to a loss of not consumed fatigue life.
The benefit of a larger extent and accuracy of measured data is to identify the truly consumed life so that loss of fatigue life can be minimised.

The type and the accuracy of the chosen mathematical set-up is influenced by the kind and extent of the input data. As indicated above, in the beginning of fatigue life monitoring some physically determinative parameters could not be measured continuously during operational usage. E.g. the A/C weight had been derived from the take-off and landing weight and was implemented in the algorithm as a respresentative, constant flying weight. Nowadays a flight-time dependent weight can be derived since the remaining fuel and the store configuration of the A/C is continuously recorded. Thus, an increasing number of data supports the accuracy of the results of mathematical modelling and allows to dispense with methods of approximation.

Classification of data (ref. 2.2.1) is one of the applied methods to minimise scattering of the calculated results and obtain high correlation coefficients when comparing the measured with the calculated strain. Thus, refinement of the classification regarding the input data can be seen as a starting-point to obtain a higher accuracy.

Finally, the mathematical set-up itself can be refined to obtain more accurate results. As explained in section 2.2.1, the TAT-based mathematical modelling can dispose of an extendend parameter set (FPS), which allows to perform multiple regression analysis. The derived mathematical set-up supplies results with a higher accuracy as can be obtained from the PPS-based modelling. In this sense, the introduction of the new Flight Data Recorder, which records the full parameter set (FPS) for each A/C of the German TORNADO fleet, opens new prospects for the refinement of data modelling.

4. REFERENCES

[1] R. Neunaber: 'Aircraft Tracking for Structural Fatigue'
presented at the 72nd AGARD Structures and Materials Panel, Specialists 'Meeting on Fatigue Management', 29. April - 1. Mai 1991 in Bath, UK

[2] P. Fraas and A. Göllner : 'TORNADO Structural Fatigue Life Assessment of the German Airforce'
presented at the 72nd AGARD Structures and Materials Panel, Specialists 'Meeting on Fatigue Management', 29. April - 1. Mai 1991 in Bath, UK

Special Applications

Development and Validation of Inflight Diagnose Methodologies using Deterministic Simulation of Subsystems

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1 Abstract

In cooperation with DaimlerChrysler Aerospace, Military Aircraft Division, an integrated CAE software has been developed for design and simulation of aircraft subsystems, particularly fuel and hydraulics. The original intention was to increase the efficiency of the subsystem design processes, however, the simulation-related part of the software can also be applied for development or as an integral part of inflight or offline diagnose methodologies. The simulation algorithm allows highly efficient treatment of the two-way coupling of continuous and discrete state changes - the latter ones being modeled with petri nets - and makes it possible to analyze whole missions by retrieving mission profiles from a database.

2 Modeling and Simulation

2.1 Simulation Environment

The usage of the simulation algorithm has been demonstrated with two applications, one being the analysis of the aerodynamical stability of rockets under influence of atmospheric turbulences [Hübner 96] and the other one being the fluid power simulation tool HydroNet [Hübner et al. 99]. The latter one will be demonstrated in the following section.

2.1.1 Boundary conditions

The boundary conditions for design and development of HydroNet were on the one hand governed by the familiar notation for fluid power systems and the technical model behind it and on the other hand the rather limited amount of information about the components of these systems - in most cases, only the head-capacity curve* is known. However, HydroNet is required to integrate design and simulation in a single software tool that uses the *engagement strategy* described in [Hübner 96]. This simulation methodology makes it possible to synchronously simulate discrete and continuous state changes and has proven to be superior compared to conventional simulation tools. In a concrete application case at DaimlerChrysler Aerospace, guided missiles division, it could be demonstrated, that without the possiblity to simulate a mixed discrete and continuous state model, a single simulation run for a rocket trajectory of

*the curve that shows pressure change over mass flow a few seconds could last between several hours and a whole day, making it impossible for the system engineer to interactively optimize his configuration and design.

2.1.2 Understanding the problem

The performance advantage of a fast simulation algorithm would be completely eliminated if the user was required to find the state equations by himself. Therefore, the software tool must enable the engineer to design his system using a familiar notation and then *automatically* generate the state equations. What we need is a drawing tool that needs only a minimum learning effort and which must be able to link simulation-relevant properties of the elements that constitute the system to the graphical objects. From these properties and the graphical layout, HydroNet must be finally able to simulate the dynamical behaviour of the system design. To make this possible, we have to go through the process which is described in [Zeigler 76] of converting a theoretically perfect but not simulatable *base model* (in this case the Euler equations for incompressible flow) into a *lumped model* that has the following properties:

1. The logical arrangement of the elements in the system design and the manufacturer specifications of the elements must be sufficient to complete the state equations.

2. It must be possible to simulate the state equations using the *engagement strategy*.

3. The precision of the simulation results must be sufficiently high to replace as many expensive and time-consuming hardware experiments as possible and to validate the system design concerning the original requirements.

This results in a systems engineering approach as described in [Igenbergs 93]. The roles which are played by the different problems that drive the development of HydroNet are summarized in fig. 1.

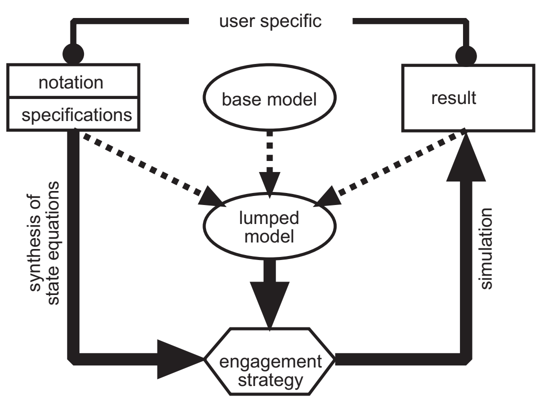


Fig. 1: The development of HydroNet

2.1.3 System design

The user interface of HydroNet has been designed for a minimum learning effort for its potential users and therefore resembles that of more famous CAD applications. According to the revised ISO 1219 standard, the symbol set is no more fixed but enables the user to assemble his own symbols from components supplied by the standard. HydroNet reflects this philosophy by allowing the user to define as many symbol palettes as he wants and to fill them with standard or user-defined symbols. After selecting a symbol in a palette, the user can place it in his system design (fig. 2) and define attributes like name, switch state or device type. These elements can be linked with pipes, tubes and similar connecting elements which have a different graphical representation but which are similarly treated, including the assignability of device types. From the user's perspective, his manipulations only result in visual differences. However, each non-trivial manipulation instantly changes the physical model and is made persistent in the database, resulting in a *guaranteed, automated synchronization of physical and visual model.* Any further workload caused by the need to manually define the system equations, fill them in with the element attributes or to persistently store them together with the system design is completely eliminated. The visual representation of the system design or simulation results can be copied into standard desktop applications like drawing tools or word processors.

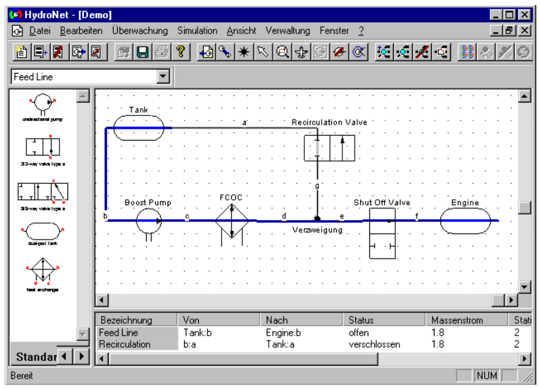


Fig. 2: The graphical user interface

2.1.4 Dynamical simulation

The static system analysis may be insufficient if parallel open paths have to be analyzed, cycles are present or the time-dependent behaviour becomes important. Then, HydroNet offers dynamical simulation and uses the methodology described in section 2.3.1 to calculate currents, pressures, tank masses and temperature over a user-defined mission profile. At each port of each element, a *virtual sensor* can be installed for these variables and the time-dependency of their values can be displayed the after simulation. If the user desires, the results can also be stored persistently. As opposed to other fluid mechanical simulation programms, HydroNet can also calculate the *instationary* current acceleration/deceleration phenomena. Elements may switch their states, if certain sensor outputs exceed user-defined boundaries or if user-defined events (which are stored in the mission profile) are raised. HydroNet therefore goes far beyond calculating the stationary system state.

2.1.5 Discrete Controllers

Many control algorithms can be described by means of discrete events that are triggered by particular values crossing a threshold. In order to manage the complexity of such controllers, HydroNet can model them using so-called Petri nets. This modeling language is well established for discrete-event systems with concurrent state changes and ideally suited for examples like the fuel management computer (fig. 3). Here, the following rules are modeled:

- Refill central collector tank if level drops below 80%.
- Use left and right wing tanks for alternating refill.

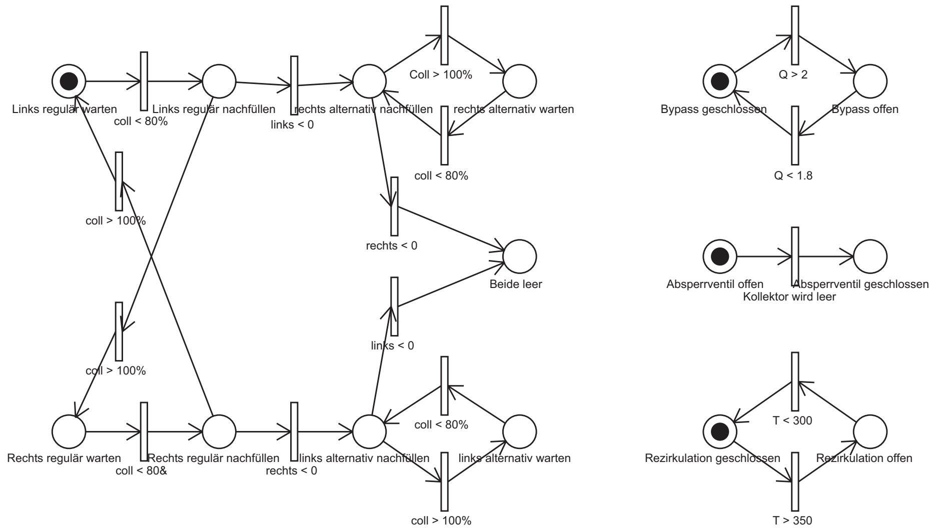


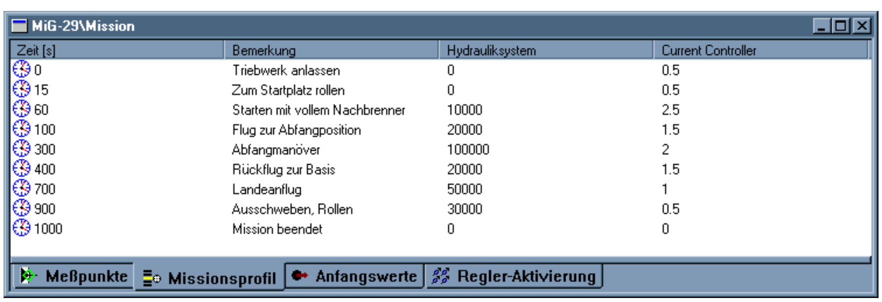
Fig. 3: fuel management computer

- Open bypass of fule-cooled oil cooler if current exceeds a certain limit
- Open recirculation valve if temperature exceeds a certain limit

Currently, only the most basic type of petri nets using boolean tokens can be modeled but this is already sufficient for a lot of applications. The user can define an arbitrary number of petri net controllers for each system and selectively activate or deactivate them together with PID controllers in each simulation configuration.

2.1.6 Mission Profiles

In order to describe non-conditional events that occur at a given time, the *mission profile* is used as a kind of event list. This mission profile reflects external influences or user input and makes it possible to evaluate different operation scenarios. A typical event list is shown in fig. 4.



Zeit [s]	Bemerkung	Hydrauliksystem	Current Controller
0	Triebwerk anlassen	0	0.5
15	Zum Startplatz rollen	0	0.5
60	Starten mit vollem Nachbrenner	10000	2.5
100	Flug zur Abfangposition	20000	1.5
300	Abfangmanöver	100000	2
400	Rückflug zur Basis	20000	1.5
700	Landeanflug	50000	1
900	Ausschweben, Rollen	30000	0.5
1000	Mission beendet	0	0

Fig. 4: event list

2.2 Simulation Methodology

2.2.1 The roles of modeling and simulation

According to [Zeigler 76], the process that leads from the problem to a simulation result can be partitioned into three major steps (fig. 5): the problem is expressed in terms of the *real system* which contains all experimentally gatherable informations. The successful attempt to formally describe the behaviour of the real system results in the *model*, which is a collection of rules that can reproduce or predict the behaviour of the real system. The translation of this model into a computer language results in the simulation program. These two processes have each a main quality criterion: the *validity*, which describes how well the rules specified in a model can reproduce the behaviour of the real system and the *correctness*, which tells us how good the simulation program can execute the rules stated in the model. To optimally satisfy these

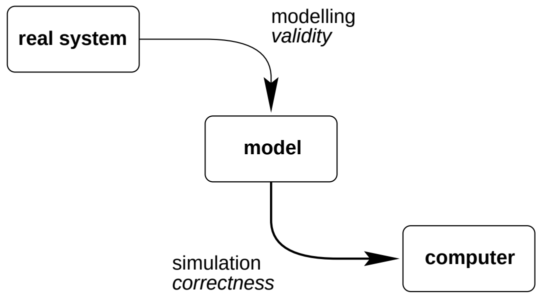


Fig. 5: modeling and simulation

criteria, the model becomes the central asset on which all efforts have to be concentrated. The correctness does not only depend on the quality of the software design process but also - and critically - on the *implementability* of the model. A model which cannot be implemented on a digital computer can never result in a correct simulation program. Therefore, the process of modeling must be analyzed more carefully.

2.3 From the base model to the lumped model

A well defined model is constituted by the *state variables* and the *state transition rules* [Zeigler 76]. Basically, there are two kinds of state variables - *discrete* and *continuous* ones. Continuous state variables mostly represent physical quantities like energy, position, momentum while discrete ones were originally only present in quantum physics. Their surprisingly frequent apperarance comes from the base model having an unmanageable complexity of the theoretically exact state equations. Therefore, the base model has to be simplified (see 2.1.2) appropriately to get satisfactory precision on the one hand and the implementability for a correct simulation program on the other hand. This process of obtaining the *lumped model* may turn several continuous state variables into discrete ones. Even if there are no quantum mechanical processes, discrete state variables are now present. For example, in most cases it is unimportant to know what exactly happens if a switch is activated or a valve opens. Only the final state is relevant so a discrete state variable is introduced that describes the state of the switch or valve (open/closed). Even a capacitance can have discrete states as the state equation changes as soon as a capacitance in a cycle becomes empty or full. Normally, it is not of interest how the very last few drops leave a tank - one only wants to know wheter fluid may be ordinarily withdrawn from it or not. If we transfer this *lumped model* on a digital computer, we get one further discretization: as digital computers are finite-state machines, they cannot represent real numbers†. Finally, the base model which was completely continuous in classical physics has become a completely discrete simulation program (fig. 6). If a model has only continuous state variables, they evolve steadily and have steady derivatives. Discrete state variables however may change infinitely fast by definition. In case of mixed sets of state variables which may even be

†The type REAL e.g. in FORTRAN is just a large integer with a different scale.

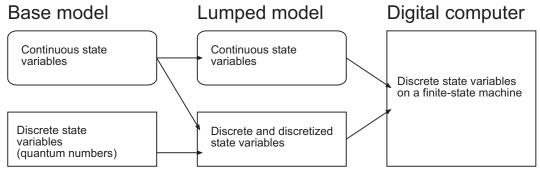


Fig. 6: stepwise discretization

coupled to each other, the derivatives of the continuous state variables may become unsteady (fig. 7). For example, the acceleration of the mass current increases infinitely fast, if a valve is being opened (of course, only in the lumped mode where there are only the states "open" and "closed".) These couplings become very difficult to handle if discrete state changes occur

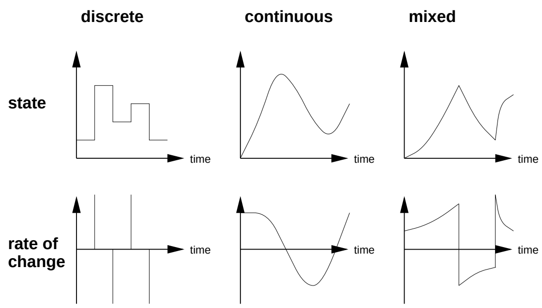


Fig. 7: state variables

at certain continuous states and not only at certain times. The situation of a tank becoming empty or full is a typical example - the time is not known, only the condition, that the stored fluid mass has exceeded certain limits. Thermostat- or pressure-controlled valves are similar examples. Conventional simulation algorithms must always use very small integration steps to hit the conditions where the discrete state changes are supposed to occur with sufficient precision. This results in simulation runs becoming unacceptably long and in a dramatic loss of precision due to the truncation of real numbers with the floating point types. To simulate systems witch coupled discrete and continuous state equations more efficiently, a new methodology which is called *engagement strategy* has been developed.

2.3.1 The engagement strategy

The engagement strategy is based on the following postulates according to [Hübner 96]:

1. The discrete state is represented by the activity modes of the engagements. At least one engagement must be active. A single engagement is a boolean part of the overall discrete state of the system. The discrete state may influence the rates of change of the continuous state variables.

2. An event is a simultaneous change of activity modes of one or more engagements.

3. There are two types of events:

   (a) *Scheduled events* are triggered at a previously known time while the continuous system state is unknown.

   (b) *Conditional events* are triggered by certain continuous states that fulfill particular conditions. The time is unknown.

4. Conditions for the triggering of conditional events are expressed in terms of so-called *hypersurface functions*. A hypersurface function is defined as a function which takes the continuous state vector as an argument and returns a real number that changes its sign exactly when the event is supposed to occur.

The scheduled events can be simulated easily with conventional methos - the integration of the differential equations that govern the rates of change of the continuous state variables is simply stopped in case of a scheduled event at the defined time of this event. The event is then handled, resulting in the discrete state changes. Then the integration is continued with the modified discrete state. The conditional events are detected with the hypersurface functions - during each step of the integration, they are permanently evaluated. If one or more of them have changed their sign, the last step is being rolled back and the integrator switches to a modified stepsize control schema that makes it possible to sufficiently approximate the roots of the hypersurface functions (fig. 8). Consequently, the integration stepsize must *only* be reduced

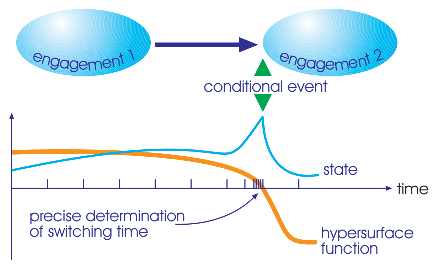


Fig. 8: root of a hypersurface function

in the proximity of a conditional event but apart from this, the integrator can still operate with its optimum stepsize. Normally, the usage of hypersurface functions may accelerate the integration by factors between 10 and 100. The term hypersurface function comes originally from the simulation program Rocksim ([Hübner 96]) as the conditional event of the impact of the rocket on the surface can be specified by a function that describes a spherical surface (of the planet on which the rocket impacts). If we refer to the often more than three-dimensional state space, we get the general term of a hypersurface.

3 Diagnose

The current subsystem design process was based on a faultless functioning of the simulated subsystem. The goal of simulation was just to evaluate whether the design proves to be viable for a given set of mission profiles. For the purpose of diagnose, the simulation tools have to be extended to include component malfunctions and also pilot reactions. The following sections will use the fuel system as a reference example.

3.1 Problem description

In currently operational aircrafts, the pilot is usually informed about malfunctions by the *central warning panel*. Based on the combination of active warning lights, the pilot has to find appropriate pages in a printed manual and go through certain procedures to take useful emergency measures. This method has the following disadvantages:

- If the pilot is under extreme stress, which is likely in case of malfunctions, this mehtod is error-prone
- Under high g-loads, no one can firmly hold a manual and read it
- The long reaction times until the emergency measures can be taken may make them obsolete due to propagation of malfunctions to other subsystems

3.2 The Role of Simulation

The simulation of aircraft subsystems can be used in various ways to support inflight diagnose or the optimization of emergency procedures. However, in each case the simulation has to be linked to some kind of evaluation method for the effects and quality of the selected procedure. Both the translation of the system design into a simulatable model and the simulation and evaluation should be used with automated, computer-based tools (fig. 9).

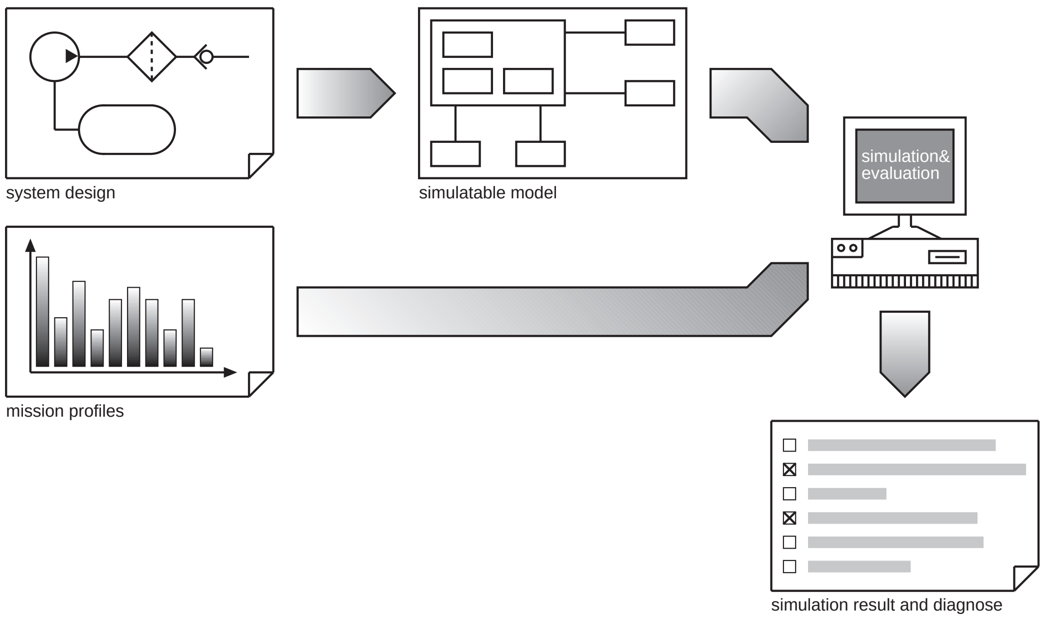


Fig. 9: simulation and diagnose

3.3 Options

3.3.1 Passive simulation

This method only uses a fixed set of mission profiles and scenarios to evaluate and then optimize emergengy procedures. The model used for simulation of the subsystem under fault-free conditions has to be extended to monitor the system state in a way that is analogous to the central warning panel and to allow failure injection and pilot reactions based on emergency procedures. This makes it possible to optimize the emergency procedures and to develop a refined version that is available as a more detailed online manual instead of the printed one.

3.3.2 Active simulation

Concept As opposed to the emergency procedures which can be considered as a kind of pre-fabricated diagnose the active simulation is aimed at providing a flexbile means of reaction to system malfunctions. Instead of using emergency procedures which are linked to particular malfunction patterns, the diagnose software will first try to determine the cause of a malfunction as precisely as possible This enables the pilot to determine the chance of survival and eases the decision for a continuation or an abort of the mission. There are several methodologies for active simulation available:

Anomaly detection Using the available simulation software, mission profiles with and without injected failures will be compared. The deviation patterns between states with and without failure must be categorized to enable their assignment to possible causes. These deviation patterns and their causes will be stored in a database that can be used for inflight-diagnose as the simulation itself probably does not provide sufficient real-time capability.

Failure prediction In addtion or as an alternative to failure prediction or the other methods, the propagation of malfunctions to related subsystems may be predicted, based on the currently determined diagnose result. This means that during the flight, a simulation process is permanently running that is provided with injected failures as soon as a certain diagnose algorithm has determined the cause of a subsystem malfunction. This requires the software and the computer to operate in real-time, however, the benefit is that the information about survivability is much more reliable.

Trial & error In this case, the simulation software will be used directly in order to retrieve a set of failures from a database that it can inject into the simulation of a previously faultless subsystem. By conducting a sufficiently high number of simulation runs, it becomse increasingly likely that the cause of the subsystem malfunction will be determined. The anomaly detection algorithm can be used as a predecessor in order to reduce the amount of causes that have to be injected as failures into the simulation process.

Fuzzy logic The main drawback of the anomaly detection algorithm is that based on continuous states and values, decisions have to be made about the cause of a malfunction that is of a discrete struture. Apart from this, a malfunction may have several simultaneous causes that contribute with varying significance to the failure. This impedance mismatch between continuous values and discrete causes leads to the introduction of diagnose methods that are based on the theory of fuzzy sets. The basic idea behind such a diagnose methodology is to use a database of fuzzyfied expert knowledge or simulation results like the ones used in the anomaly detection algorithm. This would make it possible to provide an incremental diagnose result - the most significant causes will be listed first, with the less significant ones following later, as their number is greater and their combination has to be evaluated.

Neural networks This method is described in detail in [Seitz et al. 2000]. The different sets of data that are used for diagnose are pre-processed and then fed into a neural network that generates an *error list* of possible causes of a subsystem malfunction. This neural network has however to be trained in order to provide reliable results. This can be accomplished using the simulation software to go through a huge number of mission profiles and failure scenarios. As ths procedure can be automated, it can significantly increase the variety of causes and effects and therefore improve the efficiency of this method.

Risk evaluation Besides the cause of a subsystem malfunction, the potential danger is also an important information for the pilot, as he is not always able to keep track of the proper functioning of all subsystems, especially under high workload conditions. Therefore, it makes sense to boost the selected diagnose methodology by an stochastic tool that calculates the probability of an aircraft loss or a similarly fatal incident from the sensor data and from simulation results. For civil aviation, such a tool has already been developed ([Kreichgauer 94]).

References

[Hübner 96] Gerhard Hübner: *Advanced stragegies for simulation of rocket performance, stability and control* 47th international astronautical congress, Beijing, China, 1996

[Hübner et al. 99] Gerhard Hübner, Eduard Igenbergs, Peter Hurst: *Facilitating integrated Aircraft Development with dynamical Simulation of Subsystems* Proceedings of the INCOSE conference, Brighton, UK, 1999

[Igenbergs 93] Eduard Igenbergs: *Grundlagen der Systemtechnik.* Vorlesungsskript, Lehrstuhl für Raumfahrttechnik, TU München, Germany, 1993

[Kreichgauer 94] Olaf Kreichgauer: *Quantitatives dynamisches Modell zur Simulation von Systembelastungen in Luftverkehrsabläufen* Dissertation am Fachgebiet Raumfahrttechnik, TU München, Germany, 1994

[Seitz et al. 2000] R. Seitz, G. Kunerth, W. Mansel, P.Hurst: *Diagnostic system with neuronal networks for aircraft* Proceedings of the AIMS conference, Garmisch-Partenkirchen, Germany, 2000

[Zeigler 76] Bernard P. Zeigler: *Theory of Modelling and Simulation.* John Wiley&Sons, London, UK, 1976.

Potential and limits of using structural modal data for health monitoring

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Abstract. Structural modal data and to some extent also dynamic response data are by nature integrating over a region, a component and eventually also the full structure. So any change in these data might be used for global health monitoring including damage localization, followed then by local damage identification techniques. The procedure and its advantages as well as drawbacks are discussed. It is shown that limits might exist due to measurement noise, which then requires the measurement of quite different modal data and the use of pattern recognition methods.

1. Introduction and problem statement

Health monitoring means monitoring the change from a normal (healthy) state to a state of degradation and damage due to degradation in system parameters which adversely effect performance or safety. In structures such parameters are material properties, geometry (including its integrity) and boundary conditions (e.g. related to fastening)Structural health monitory (SHM) and damage detection is an extension to more traditional nondestructive inspection (NDI) and measurement methods in the sense that NDS uses ad hoc data, while for SHM the history of data is crucial. Hence, NDI relies more on sensors and equipment, while SHM in addition is significantly dependent upon the interpretation software. Moreover and equally important, NDI is closer to the „damage of the material“, while SHM looks to the component and full structural system (with eventually leaving local detection to NDI methods, or even accepting local damage). Such a monitoring is not only related to properties or response quantities, but also to environmental conditions and loads. This can be extented to avoiding undesirable external effects, which is the case for example in maneuver control of aircraft.

2. General overview on structural health monitoring techniques

There is currently a large effort underway to efficiently and economically detect damage – or preferably even relevant degradation – in structures such as aircraft, machines, bridges and buildings. Historically, inspections have been conducted manually which has typically resulted in downtime, labor costs, and human oversight. While some forms of structural damage are fairly obvious upon inspection, other forms are not. For example, delamination or debonding inside laminated or hybrid structures are difficult to detect. Though different methods such as ultrasound, x-ray radiation, electromagnetics, acoustic emission, thermal imaging, fiber-optics or strain sensing (including piezoelectrics) are available, none of these can be considered as a silver bullet for this task. These techniques either require skilled human interaction and judgment, or are very difficult to apply in an in-situ manner. Moreover, most of these methods are local methods, i.e. sensing and observation has to take place at the right position within the structure or component. Some of these methods, such as those based on acoustic waves within the solid under consideration, may be extented to a more global approach. But then complex sensing and evaluation networks are required, and due to considerable but undesirable back-ground noise, in practice they often so far work under clean (quiet) lab conditions, only. Techniques which use known (actuator) inputs increase complexity, but often lead to more reliable results. A considerable amount of work has been also done using dynamic system response measurement. This per se is a global method as far as fundamental modes are concerned, and becomes more local for higher modes. These techniques are based on the observation that the modal properties (resonant frequencies modes and damping) of a structure will change with damage. Some form of modeling is required in order to quantify the natural characteristics of the structure before and after damage. Good examples are mass and stiffness matrices from finite element models, transfer function parameters, or analysis of the structural impulse response. Many of these methods are model-based and often require some computational effort, for example to solve an eigenvalue problem. The use of piezoelectric transducers has served to simplify the hardware requirements of these techniques as well as preclude mass or stiffness loading in non-destructive evaluation methods. These techniques, their pros and cons, are discussed in more detail in the following.

3. Basic concepts of dynamics and modal system response techniques

System modal properties and dynamic response in most cases „integrates“ the system properties over the observed modes and points of dynamic response determination. So they more aim to global health monitoring, including rough localisation of defects which then are to be identified in more detail by more conventional methods..

In the following, modal approaches are described first, followed by a discussion of SHM techniques based on dynamic response.

3.1. Modal techniques

Modal techniques aim at matching actual experimental(e) modal data to desired / theoretical (t) or ideal ones. Eventual mismatch then leads to supposed degradation or damage. In principle, mathematically this leads to a nonlinear least squares problem of type

Minimize $$I(\{x\}) = I_1\ (\{x\}) + I_2(\{x\}) + I_3(\{x\}) \tag{1}$$

with $$I_1 = \sum_i w_i \left[\left(f_i^{\,e} - f_i^{\,t}\right) / \, f_i^{\,t}\right]^2 \tag{2}$$

as weighted eigenfrequencies deviation

and $$I_2 = \sum_{i,k} W_i \left[\left(\varphi_{ik}{}^{e} - \varphi_{ik}{}^{t} \right) / \varphi_{ik} \right]^2 \tag{3}$$

as weighted mode deviations

and $$I_3 = \sum_{j} w_j x_j{}^2 \tag{4}$$

used to constrain deviations of system parameters $x_j = 1,...n$, which are to be monitored. Eventually, further constraints on x_j are to be taken into account as well. It is the hypothesis of this approach, that by solving (1) and obtaining system parameters x_j which are not nominal, degradation or damage has to be assumed. This then also allows some localization of damage. Before discussing this further also with some practical consequences, the mathematical-numerical implication of (1) is briefly resumed.

Solution of (1) as a nonlinear least squares problem requires

- a theoretical or ideal model „t" (which could be also determined experimentally for the ideal/undamaged system)
- a solution algorithm for nonlinear and constrained least squares problems with
- preferably also sensitivity information of the eigenfreqencies and modes with respect to the system parameters x_j (these can be obtained with proper finite element codes,)
- and further algorithmic details such as selection of weighting factors w_i, scaling of data, or pairing the proper ideal with observed data e.g. via modal assurance criteria (parallelism of ideal and observed mode vectors)

For simplicity of presentation, these points are assumed to be given. In fact, a lot of literature and software tools can be found. Moreover, these point are considered not to be the bottlenecks of such dynamic and modal health monitoring techniques.

Application of this nonlinear least squares technique to different test examples shows its robustness also against noisy measurement data. Essential is that the damage has an effect on the measured data, which could require a mix of classical modal data together with static measurements. Of course local defects in general do not significantly effect lower modes and frequencies, and hardly can be identified within the measurement noise which typically is in the order of 0,1-1 % for (resonance) frequencies and even larger for modes. If for example in a simply supported beam the stiffness at quarter length is reduced by 10 % due to some degradation, then the fundamental frequency shifts from e.g. 8.08 Hz to 8.076 Hz, a difference which is difficult to identify, let even assess, in practice. Methods that use mode shape vectors as a feature generally analyze differences between the measured modal vectors before and after damage. Mode shape vectors are spatially distributed quantities; therefore, they provide information that can be required to accurately characterize mode shape vectors and to provide sufficient resolution for determining the damage location. An alternative to using mode shapes to obtain spatially distributed features sensitive to damage is to use mode shape derivatives, such as curvature. A comparison of the relative statistical uncertainty associated with estimates of mode shape curvature, mode shape vectors and resonant frequencies showed that the largest variability is associated with estimates of mode shape curvature followed by estimates of the mode shape vector. Resonant frequencies could be estimated/measured with least uncertainty.

3.2 Dynamic response quantities

The discussion related to modal quantities is also valid to a considerable extent to the use of response quantities ranging from transfer functions over response data in time domain until the measurement of strain waves. An advantage in practice is that data to be measured can be excited either by “natural” loads or by inherent micro-shakers such as piezoceramic plates. The latter can be used also highly sensitive strain sensors. Disadvantages are the relatively complex data processing electronics, or such details as the relatively small rupture strain of classical piezoceramic plates.

4. Conclusions

Modal and dynamic response data offer the possibility of global structural health monitoring. Techniques based on mathematical structural models similar to those used in model updating allow to determine the region of probable damage with high probability Conditions of significant measured data which overrule measurement noise have to be satisfied. This then leads to a multiple measurement net, eventually to be combined with pattern recognition methods. There also seems to be no single health monitoring technique, but proper combinations of different methods allow damage identification with high reliability.

<u>Reference</u>

C.R. Farrar et al, A statistical pattern recognition paradigm for vibration based structural health monitoring, in: Fu-Kuo Chang (ed): Structural Health Monitoring, Technomic Publ, 1999, pp 764-773

Monitoring Rolling Stability during Start and Landing

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ABSTRACT

Fuselage, wings, landing gears and tires are elastic structures and carry a big dead load. Ground roughness and sidewind, braking and other suddenly acting forces disturb the straight run of the elastic and hysteric damped airplane structure. Computing stability of the run needs many degrees of freedom (DOF) to simulate the energy-flux penetrating through the contact areas of the tires into the system. Monitoring of dynamic tire behaviour can be done by intelligent sensors built into the wheels.

Measurements of tire dynamics are compared with computations of runs using three types of nonlinear tire models:
A) simple model (2 DOF), B) circular ring model (50 DOF), C) membran-shell model (1500 DOF).
This different dynamic behaviour is shown for the three types of sensors:
a) tire side wall magnetic sensor, b) tire tread Hall sensor, c) acoustic tetraeder reflector.

INTRODUCTION

The airplane consists of several substructures with different frequencies. As shown in [1] the contacting medias do not produce conservative forces and leads to self excited oscillations. To get a slowly changing or quasi stationary state of the system it is necessary after I**. Prigogine** to have enough hysteretic energy dissipation in this system[2]. Not only the tires, also the fuselage and the gears should have enough dissipation. The problem can not be solved by oscillation tests on ground, the reason is that the rolling system is not fixed in space using simple boundary conditions.

For low frequency reactions we can use the stability condition of understeer concept as for cars, see fig. 1:

$$aC_v < bC_H, \text{ here}$$

a is distance CG - front gear, b is distance CG - rear gears

$C_{V,H}$ is cornering stiffness of gears

This condition is also guilty for steering by different bracking with the main gears. But it is not guilty when the main gears are sliding. The remaining problem is the elastic reaction of the fuselage and by increasing of the bending stiffness by factor 2 we get a stable, steerable system By animation of the computed reaction after a side-wind gust, one sees that elastic fuselage produces a time lag witch is responsible for this instability.

Fig. 1 shows deviations from straight run by short sidewind gust:

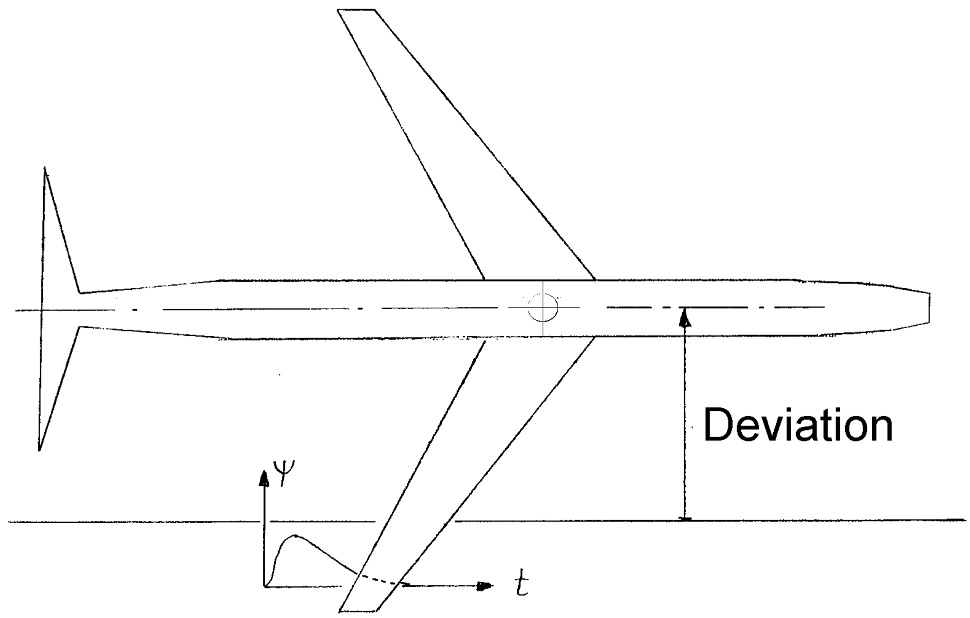


fig. 1a) Heading angle Ψ reaches zero with stiff fuselage, small deviation of airplane occur

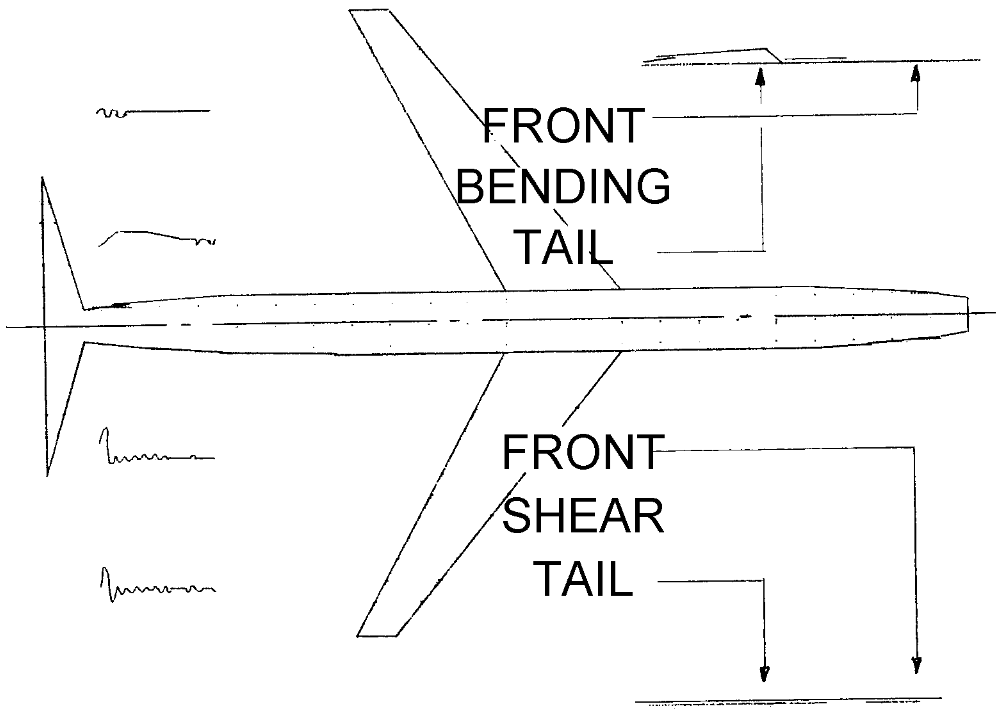


fig. 1b) small bending and shear of fuselage (drift of the airplane is eliminated by reference to undeformed shape)

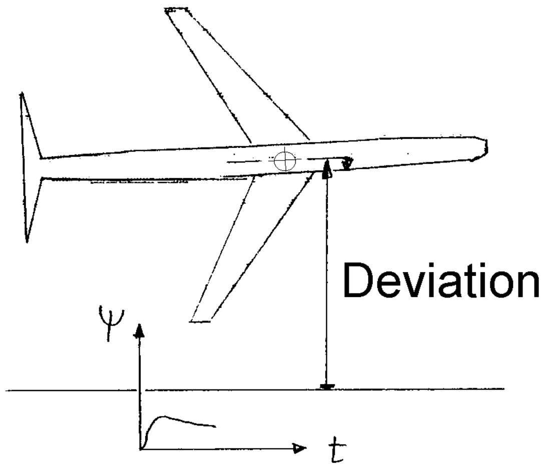


fig. 1c) strong deviation of airplane, heading angle does not reach zero

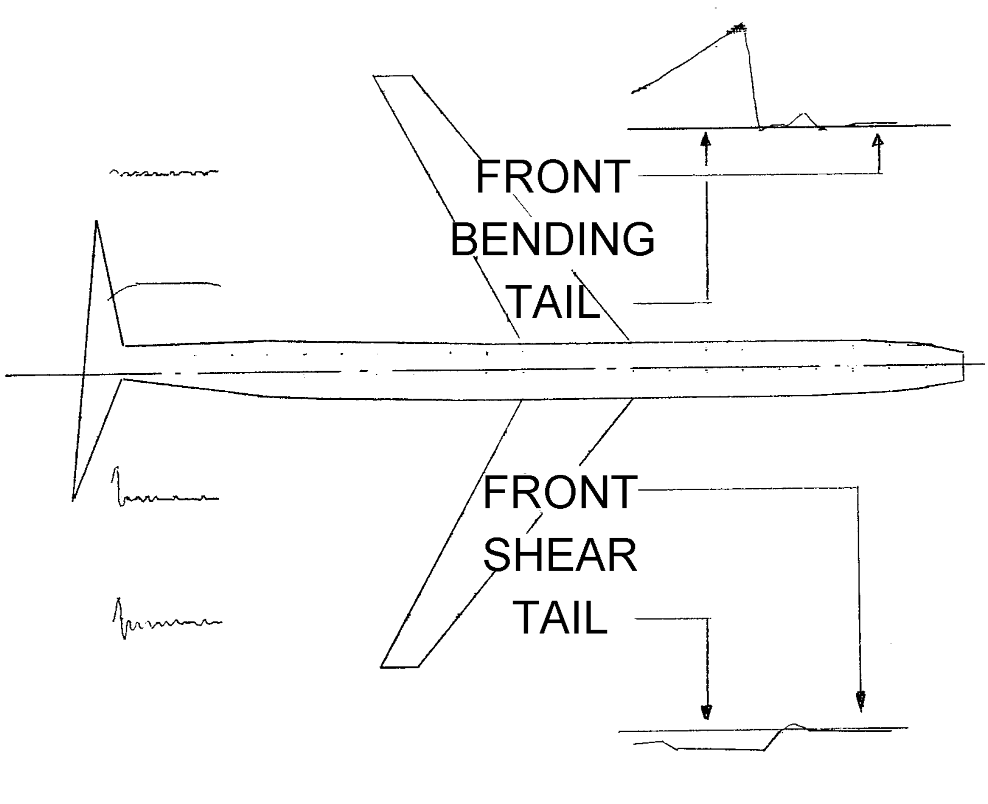
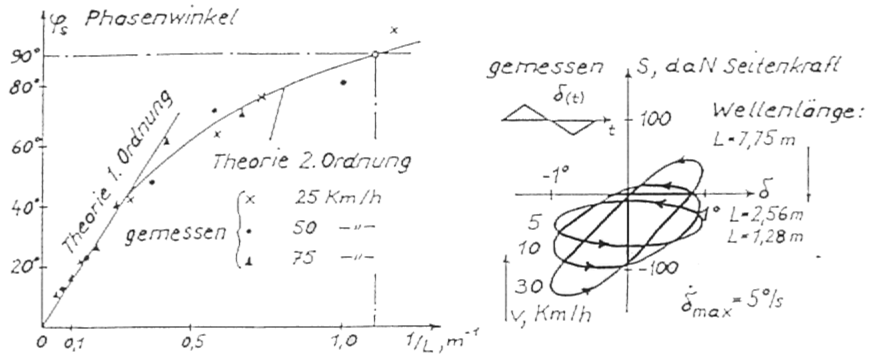


fig. 1d) strong bending and shear of tail (drift is eliminated)

THREE TYPES OF NONLINEAR TIRE MODELS

In the introduction the tire model was a linear one without slipping and only for computing the stationary state of tire forces. The deformations in contact area at begin are zero and increase linear to end of this area. In 1985 [3], two degrees of freedom (lateral deflection of contact, change of rolling direction) were introduced together with sticking and slipping zones of contact area. Sharp steering and driving/braking maneuvers could be computed and explained. Contact pressure distribution must be given. Short steering wavelength show, as in measurements, a considerable lack of steering forces, fig. 2. Nonlinear unstable oscillations for front gear were investigated used Hurwitz criteria [4].



Fig, 2: Computed and measured Phase angle of car Tire for different wave length of Flutter:

a) left hand is drawn the computation of 1.st Order theory, good use down to L ~ 5 m. For shorter wavelength 2.nd Order theory is necessary to get kinematic Flutter point at L = 0,8m.
b) right hand is drawn the Cornering force for triangle shape excitation of Cornering angle. One sees that for a wavelength of L = 1,28m there is no steering effect.

The next tire model was designed to find the longitudinal distribution of contact pressure, together with tangential and lateral slipstress. The tire was simplified to a circular ring on elastic foundation. Nonstationary distributed contact forces and axle reactions were computed and measured for short impacts, fig. 3:

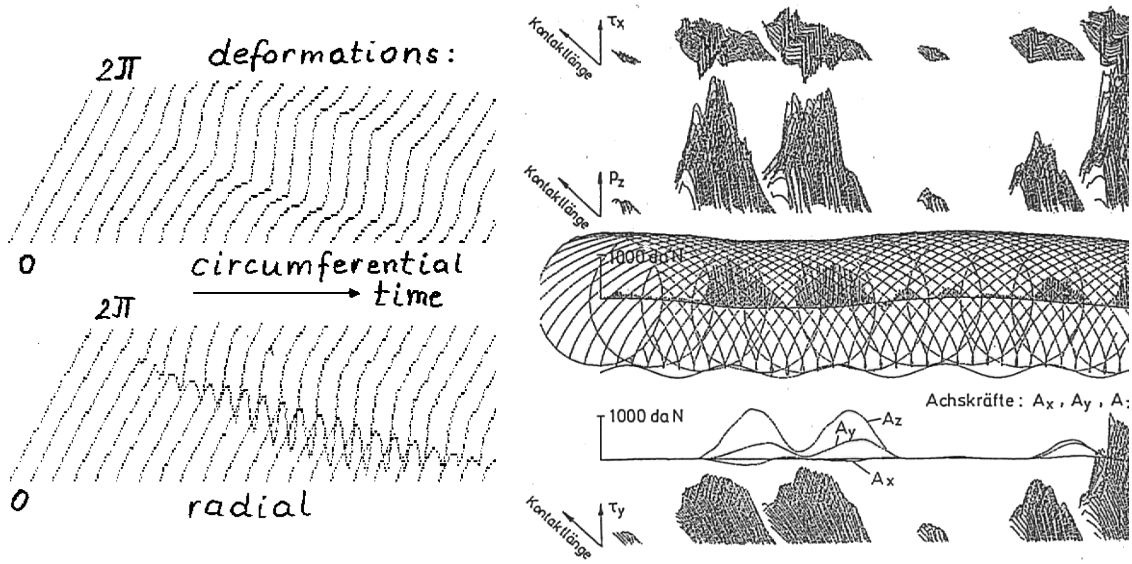


fig.3: Rolling on waves of runway: axle forces $A_{x,y,z}$ in longitudinal, lateral and vertical direction, Shown is along contact length the lineloads τ_x, $\tau_{,y}$, p_z in the same direction.

The most complex model of tire is a membrane-shell model starting with the correct tire section profile and showing the non-stationary pressure distribution longitudinal and lateral in the contact zone. It is a 3D nonlinear model to compare computed and measured shell deformations shows fig. 4.

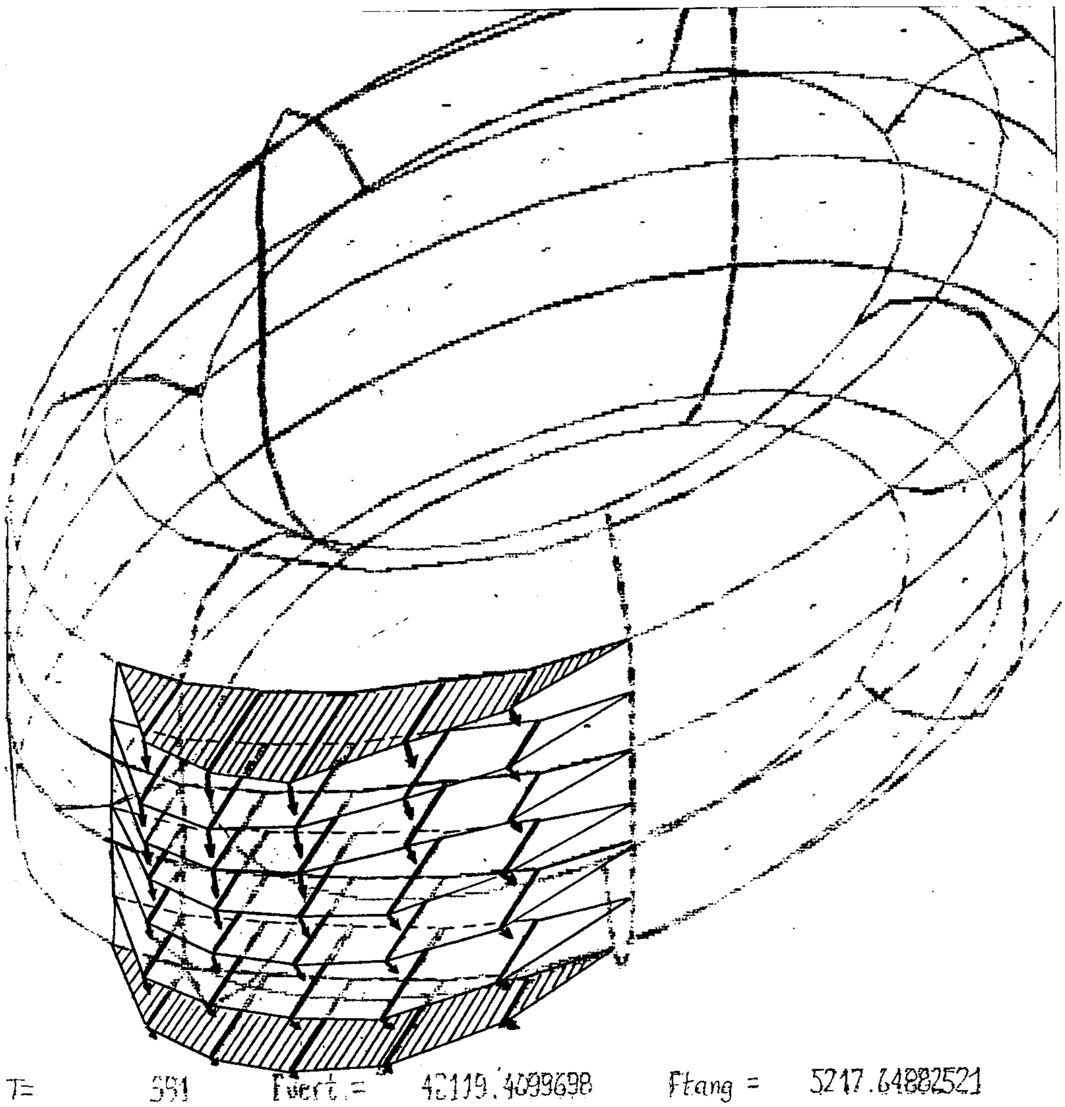


fig.4: Tire model for 3D Computations up to high velocities. Reaching a stationary state at 360 km/h needs a stiff and good damped belt. Inner pressure is p=20 bar. The vertical load is 42 to, the cornering angle is small. All important external and internal forces and deformations can be drawn from this solution because this tire model is guilty for external and for internal mechanics of tires.

THREE TYPES OF SENSORS FOR DYNAMIC TIRE BEHAVIOR

To measure tire sidewall deflection magnetic powder is mixed into strips of the sidewall coating rubber. Rotation of the radial strips produce signals along the tire circum ference [5] and show rotation velocity, circumferential shear and lateral deflection. This signals can be compared to tire model B and C as computed signals, shown in fig. 4.

The tire tread Hall sensor developed at TU Darmstadt has two applications. First it measures the average shear angle of the belt (not metallic!) and this signal is due to radial shear of belt cross-section. Secondly for metallic belt it measures a certain part of the shear of tire tread. In this case it is a signal for tangential contact stresses [6]. Additionally on glass plate tests at TU-Berlin wrinkles were found caused by high lateral forces.

An acoustic 3D reflector, measuring the distance to a microphone at the inside of the belt equator is at TFH-Berlin in development. An analog mechanical device was developed at TU-Berlin for low frequency measurement up to 50 Hz for agricultural tire dynamics and the correlation to 2D computation is very good [7].

CONCLUSION

To get an exact insight into nonlinear rolling stability three independent principles should be used:

l. Analytical methods to get fundamental insight (understeer, flutter of wheels)
2. Nonlinear numerical response of the system in time and deviations from the design state (filtering drifts).Wavelet methods for measured and computed accelerations may cancel this drifts caused by unbounded dynamics.
3. Not only thinking in linearity and eigenforms, also thinking for sub-resonances due to nonsmooth runway and snap through effects, or wrinkling of thin membrane-shells should be considered.

Literature:
[1] AIMS 98, Page 443-453
[2] Constituive Models for Rubber, Page 145-150, Balkema, Vienna 1999
[3] Böhm, F.: From non-holonomic constraint equation to exact transport algorithm for rolling contact. 4th Mini Conference of Vehicle System Dynamics, TU Budapest 1994
[4] AGARD 800, Page 6-1/6-22
[5] Continental/Tewes AG
[6] D.I. Volker Bachmann, Untersuchung zum Einsatz von Reifensensoren im Pkw; Fortschrittberichte VDI Reihe 12, Nr. 381, 1998
[7] Böhm F., Duda A., Wille R., Zachow D.: Investigation of the non-stationary rolling contact of a tire on natural soils//13th International Conference of the ISTVS, München, September 1999

Diagnostic system with neural networks for aircraft

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1. INTRODUCTION

The continuous enhancement of functionality in modern aeroplanes has led to an ever more complex system "aeroplane". It has become necessary to integrate several different subsystems in an aeroplane, which not only have to be served by the crew, but also be constantly monitored. In cases of failure, which are displayed by lamps located in different places within the cockpit, the crew has to initiate countermeasures to guarantee that the erroneous function of the system "aeroplane" react optimally and – in more serious cases – avoid serious consequences to personnel and aeroplane.

In many cockpits – especially in military aircraft – this is achieved with the aid of printed handbooks – the so-called Flight Crew Checklist and extensive training for the crews. The handbooks document the symptoms of failures, mostly a combination of flashing labelled lights, which are called captions. A pattern of captions can be assigned to failures. For each failure there is an "emergency procedure" , an action list for the crew, which details how to solve the problem. When two or more failures occur simultaneously, the crew has to identify the failures and decide which failure is the most serious one. After that they can start to work through the corresponding emergency procedure.

Every step of the emergency procedure has to be confirmed. Many emergency procedures consist of only a few very specific directives for the failure and then refer to a more general procedure after that. The crew has to find the referenced procedure in the handbook and may have to remember to come back to the start point. This means a loss of precious time, more workload for the crew and higher risk because of possible human failure.

The diagnostic system, which is presented here, monitors the aircraft "on the fly" with the aid of Neural Networks. In the case of failures the system identifies the failures, assigns them a priority, informs the crew and displays a emergency procedure on a cockpit display. Because of its modular architecture the diagnostic system can be used for several types of aeroplane presuming that the data needed for diagnosis is assessable to the diagnostic system.

Chapter 2 describes the modular architecture of the diagnostic system. The diagnosis, the source and the preparation of data are detailed in chapter 3. Chapter 4 specifies the electronic form of the Flight Crew Checklist and the creation and editing of the checklists with an editor. The actual status of realisation is described in chapter 5.

2. THE DIAGNOSTIC SYSTEM

The diagnostic system consists of the following modules (see figure 1):

- Interface A/C Diagnosis: Interface between the aircraft systems and subsystems and the diagnosis module.
- Module Diagnosis: data pre-processing and classification with Neural Networks
- Module Online Checklist: electronic Flight Crew Checklist
- Display & Keyboard driver: Interface to input and output devices

The modules Diagnosis and Online Checklist are detailed in chapter 3 and 4. The modules Interface A/C Diagnosis and Display & Keyboard driver won't be described here, because they only serve to interface the diagnostic system with the aircraft systems.

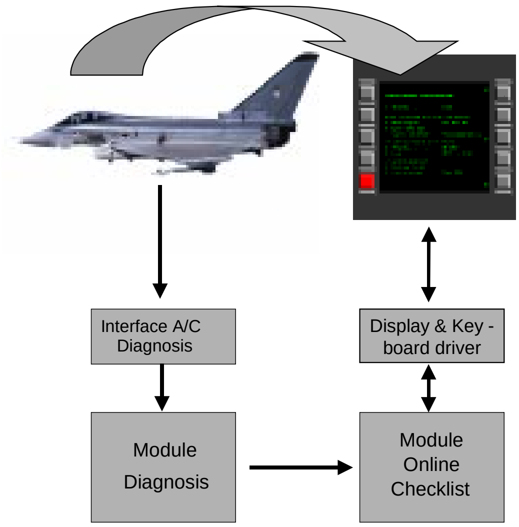

Figure 1: Modular architecture of the diagnostic system

3. DIAGNOSIS

Figure 2 gives a schematic view of the module diagnosis.

The data of the aircraft systems and subsystems are fed into the Neural Network for classification. For some data a pre-processing is necessary. In the case of failures the classifier deliver a list of recognised failures in the order of their importance.

The following chapters detail the sources of the data, their pre-processing and the Neural Networks as classifier.

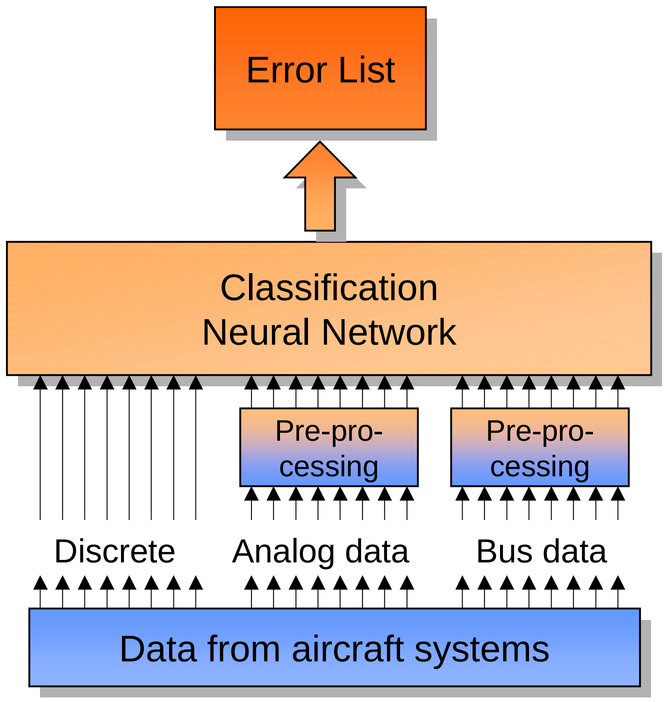

Figure 2: Schematic overview of the module diagnosis

3.1 Input data and their pre-processing

To make an expressive diagnosis you need as much fault related data of the defective system as possible.

The data of the systems can be present in various forms. To handle the data with Neural Networks you have to transform them into an usable format:

- Discretes are handed on directly to the Neural Network. The conditions "on" and "off" or "failure" and "no failure" are assigned the values 1 and 0.
- "Analog" data digitally represent temperatures, pressures etc. They have to be normalised to the intervall [0;1], which is usable for the Neural Network. In some cases there are also more complicated pre-processing steps; for example a logarithmic transfer function or a gradient.
- Bus data have to be extracted from the messages in which they are sent on the bus. After that they can be treated like discretes or analog data (see above).

The input data of the diagnostic system can be distributed into two different categories:

- Data, which represent directly a failure in a system (for example a result of a built in test)
- Data, which need a complex assessment before a failure can be detected (for example a high temperature in a engine during high thrust can be normal while the same temperature in idle can indicate a problem with the engine)

Neural Networks are able to operate with both categories of data.

3.2 Classifier: Neural Networks

The task of the classifier is to assign the actual status of the aeroplane to one of the predefined failure types via analysis of the data from the systems. Every type of failure has a corresponding emergency procedure for recovery.

The classifier is a feed-forward network with a variation of the back-propagation algorithm as learning method. The number of input neurons corresponds to the number of input parameters and the number of output neurons corresponds to the number of failure types.

Depending upon the complexity of the problem a neural network with or without a Hidden Layer was used (see figures 3 and 4). For simple problems it can be achieved without a hidden layer which enhances computer performance (computing time, memory requirements) and training. The usage of hidden layers in complex diagnosis problems can lead to significantly better results, so the additional effort is vindicated.

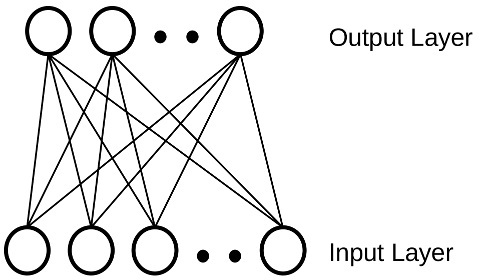

Figure 3: Neural Network without a hidden layer

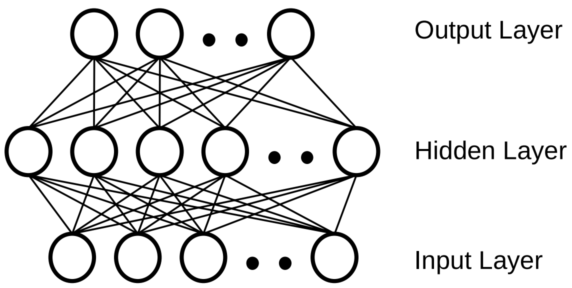

Figure 3: Neural Network with one hidden layer

The neuron model uses an interval of values from 0 to 1, a sigmoid activation function and a synchronous updating. There are some hundreds of neurons in the network depending on the pertinent problem.

The results of the tests with neural networks are:

- It is not necessary to build up a model of the system which has to be diagnosed as it is enough to have a set of test examples which assign symptoms and failure types. The corresponding rules and relations are found automatically by the network during the training.
- Because of the network's ability of generalisation it is also possible to identify incomplete input patterns, for example in cases of defect sensors.

- After the training of single-failures (only one failure at a time) the network is able to recognise two or more failure at a time with high reliability. After the training of double-failures the certainty of recognition rises.

One test example was the TORNADO aircraft alarm panel. The task was to assign 80 captions to 70 failure types. Because some of the input patterns for the classifier are completely enclosed by other patterns the ability of detection descends for rising number of simultaneous failures. Despite that the following recognition rates were attained:

failures	**recognition rates**
Single	100 %
double	97,3%

4. FLIGHT CREW CHECKLIST

Even today the documentation in the cockpit of many aeroplanes consists of printed manuals. During an emergency the procedures – at least those not known by heart – are looked up in the printed manual. The diagnostic system on the contrary allows a direct view of the Flight Crew Checklist on a display in the cockpit. That denotes a important reduction of time.

4.1 Viewing of the Checklists in the cockpit

The following requirements were made on the electronic form of the Flight Crew Checklist (further on called Online Checklist):

- The functionality of the Online Checklist shall be easily portable to various aeroplanes – only with different content of the procedures.
- The visualisation of the Checklists on a display in the cockpit shall be similar to the paper checklist.
- The display in the cockpit shall be able to present coloured test and graphics.

With the help of aircrews the Flight Crew Checklists were analysed. The crews on their side also had some additional requirements for the electronic handbook:

- After the classifier has detected a failure the corresponding emergency procedure shall be opened automatically.
- The crew shall be able to select an emergency procedure by hand.
- Automatic tracking of cross references. That should replace the browsing in the paper documentation.
- A clear portrayal of textual and graphical information

In a specific analysis the structure of the Flight Crew Checklists was stipulated. The information can be divided into categories, procedures and simple steps. It was possible to depict the information in a tree structure, which is a directed, possibly cyclic graph (see figure 5).

A Checklist (for example the collection of all emergency procedures) consists of categories (e.g. “electric failures” or “fire”). The categories themselves are divided into procedures (e.g. “generator failure”). A procedure contains a sequence of steps, which consists of the following information:

- The text, which has to be displayed
- Attributes (e.g. colour) for displaying the text
- Kind of reaction on the step, e.g. :
    - The step has to be confirmed by the crew
    - It is a question which has to be answered with “yes” or “no”
    - The step is only displayed, no reaction of the crew is required
- Cross references to the next step/ steps

Some steps consist of additional information, e.g. references to graphics or tables, which are displayed on their own pages (see below).

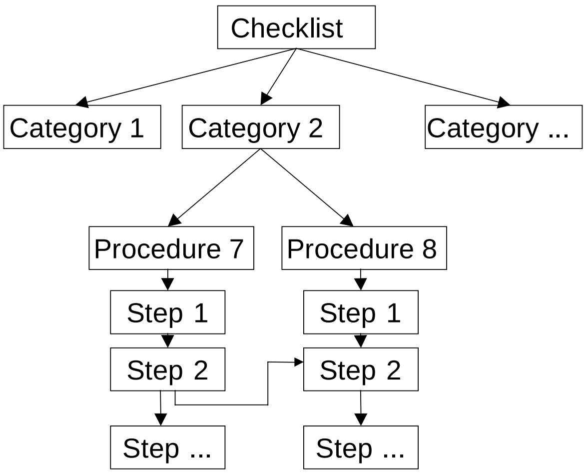

Figure 5: checklist structured in a tree

When an emergency procedure is “launched”, the crew has to work through the procedure step by step. The procedure is composed for the specific error condition of the aeroplane, i.e. the diagnostic system displays only information which is relevant for the actual situation. So the workload for the crew is minimised.

The first step in the realisation of the system was to build a demonstrator for the display outputs with the Tool VAPS[*]. This first model was iteratively improved with cockpit crews participation. The result after some iterations was a demonstrator which made it possible to display and work through a real emergency procedure. This was used to define the requirements for the man machine interface.

The visualisation of the checklists is done in five modes:

- Main mode is a textual display of the emergency procedure. After the initialisation of the system the user is able to select a checklist, category and procedure via a menu. The selected procedure is displayed on the screen (see figure 6). In case of an emergency the corresponding procedure is displayed automatically.
- Tables: Because tables take in nearly the whole screen, they are not displayed in the text –

[*] Visual Applications Builder von Virtual Prototypes Inc.

like in the paper checklists – but on a separate page. They are collected until the failure is solved or the system is restarted.

- Graphics: see tables above. Complex graphics are replaced by a calculator. The input parameter for the calculator are taken from known data (e.g. analog data, see above) or have to be typed in by the crew.
- Status View: A page which gives a graphical representation of the system that is affected by the failure.
- Consequence List: This page contains the restrictions for the flight envelope of the aeroplane which are cause by the failures.

The changing between the five modes is done with two keys. Four additional keys are used for navigating within the procedure, as the crew have to:

- Confirm a step
- Cancel a procedure
- Correct previous steps
- Answer “yes” or “no”
- In calculator mode: change the parameter

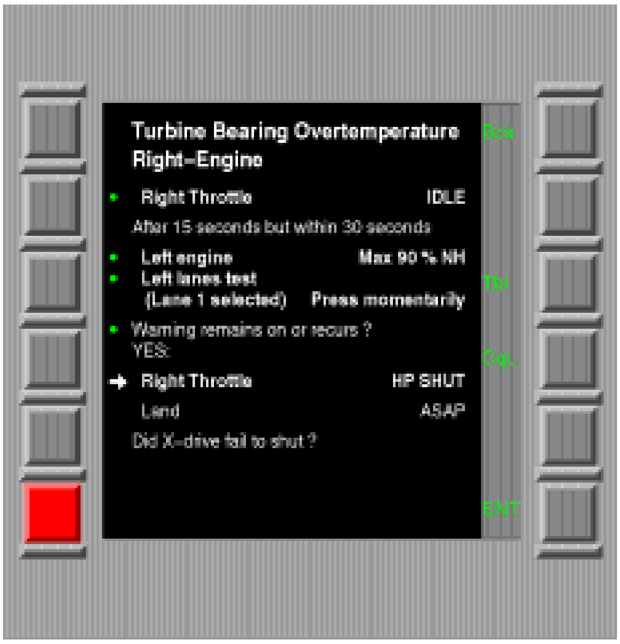

Figure 6: Visualisation of the Flight Crew Checklists on a Mulit-Function Display

4.2 Checklist Editor

The information contained within the checklists are not strictly coupled with the application. It is placed in a certain order in its own database file. So the information of the checklists can be changed independently from the application.

The simple changing of the database file enables the maintenance crew of the aeroplane to update the Online Checklists very easily. It is possible to change the information only once and then to distribute the new database file to every aeroplane. After loading the file into the aeroplane the newest version of the checklists is available for the aircrew.

The editing of the checklists takes place on a computer (e.g. a PC or a workstation). With an editor (see figure 7) the user is able to:

- Browse the checklists in a tree structure
- View the procedures in the same format as displayed in the aeroplane
- Create and remove procedures
- Construct a procedure and its steps
- Edit all attributes of the individual steps
- Check the consistency of a checklist
- Create the database file, which can be loaded into the aircraft

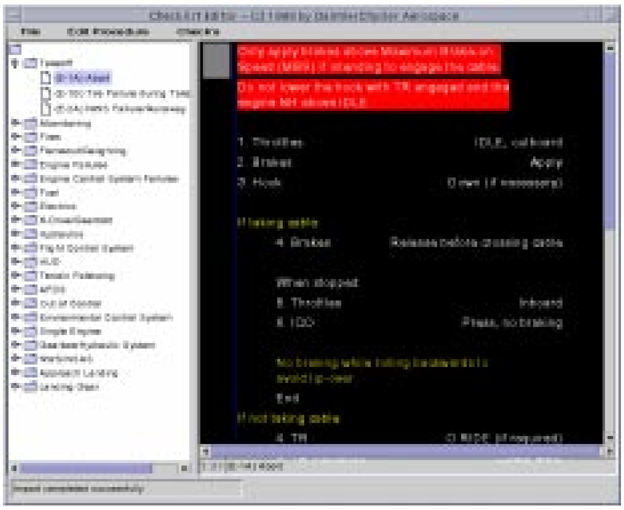

Figure 7: Checklist-Editor: on the left side the tree with all procedures, on the right the visualisation of a procedure

5. REALISATION

The diagnostic system was developed on workstations. The diagnosis module and the Online Checklists were integrated in C / C++, the checklist editor was integrated in Java.

After the completion of integration of the diagnostic system on workstations the verification of the functionality was started:

1) Verification of module diagnosis
The system was integrated on a PowerPC system with digital and analog inputs. The PowerPC system was coupled to a non-flying aircraft systems demonstrator, a so-called Rig. The output of the Online Checklists was displayed in a simulation cockpit. After user induced failures in the Rig the corresponding emergency procedures were displayed in the simulated cockpit.

2) Verification of the system in flight:
The first flight test was in a TORNADO aeroplane with a colour display and a experimental pod (called ADT). A PowerPC system is installed in the experimental pod, similar to one used in the Rig, which is able to drive the colour display in the cockpit. The first flight test was in July 1999.

6. CONCLUSION AND PERSPECTIVE

An electronic form of the Flight Crew Checklists with a automatic detection of failures can reduce the workload of a cockpit crew in an emergency situation and increase safety standards.

The module diagnosis monitors the aeroplane with the help of a neural network, which detects failures because of learned patterns. With the ability of generalisation the neural network is also able to classify unknown caption patterns, i.e. the neural network can also appraise patterns which have not been part of the training patterns. After a detection of a failure the module diagnosis initiates a error message to the crew and an automatic display of the corresponding emergency procedure. If there is more than one failure at a time an additional prioritising is performed, i.e. the detected failures are sorted in order of their seriousness. In principle the diagnosis module allows a very fast assessment of a complex failure situation.

The Online Checklists are an electronic representation of the Flight Crew Checklist on a coloured display in the cockpit. The structure used for the checklists allows the system to exactly tailor the procedure to the actual situation. It is no longer necessary for the crew to browse in the paper checklist.

At the moment we are attempting to improve the hit-rate of failure detection in cases of multiple simultaneous failures. Our approach is to use probabilistic networks for this task. Our first tests seem to confirm this approach.

Helicopter Applications

USCG HH-60J STRUCTURAL USAGE MONITORING EVALUATION

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PURPOSE

The United States Coast Guard (USCG) sponsored a project to evaluate the benefits of monitoring the structural usage of their HH-60J helicopters. The purpose of the project was to monitor the operational usage of four individual HH-60J aircraft for a period of four to six months, determine the fatigue damage accumulation in selected flight critical components, and evaluate the cost and safety benefits of helicopter structural usage monitoring. A concurrent purpose was to demonstrate the proof-of-concept for helicopter structural usage monitoring using an off-the-shelf, low-cost data acquisition system. The emphasis was placed upon the affordability of the entire data acquisition and processing process and ensuring the conservatism, not analytical accuracy and precision, of the results.

BACKGROUND

The USCG operates a fleet of over 200 aircraft and employs the Aviation Computerized Management System (ACMS) relational database for configuration management, maintenance action scheduling, documentation and component history tracking. ACMS gives the USCG the capability to track operational and maintenance history of each component throughout its entire life cycle. This information is used to support the service's Reliability Centered Maintenance (RCM) Analysis Program, and to identify appropriate corrective actions for components that do not meet supportability expectations. The USCG recognizes the potential benefits of increasing the fidelity of the ACMS information to enhance the effectiveness of its RCM Analysis by identifying each flight regime actually experienced by an aircraft during every operational mission. From this flight regime description, calculations can be made to determine the fatigue life expended for each fatigue life critical component based upon the amount of time spent in each flight regime.

CONCEPT

Sikorsky established the HH-60J flight critical component replacement times (CRTs) to protect flight safety by ensuring less than one-in-a-million chance of catastrophic failure, which is six-nines of structural reliability. As shown in Figure 1, there are three elements included in this process. The first element is the assumption that each HH-60J always flies the composite worst case (CWC) flight spectrum. The CWC spectrum is based upon severely damaging regimes, and conservative percentage for total flight times spent in the more damaging regimes. The second element is the determination of the loads and stresses experienced by each flight critical component for each of the CWC flight regimes. Cyclic loads and stresses are measured during flight test, and are enveloped by assuming the total cyclic range to be a constant amplitude between the maximum positive and negative measured values. The third, and last, element is relating the loads and stress to fatigue life.

Stress vs number-of-cycles-to-failure (S-n) curves are derived by testing a minimum of six components to failure at varying constant amplitude loads. These results are plotted on a semi-log scale and are curve-fit to determine the mean relationship between stress level and number of cycles to failure. In order to account for material property variations, time and usage related degradation and other unknowns, the S-n results are adjusted by determining the mean minus three sigma cycles-to-failure values. Using Minor's Rule of linear damage accumulation, and knowing the adjusted stress values for each component and CWC flight regime, component damage accumulation rates, per unit time, are derived for each CWC flight regime. Component replacement time is determined by assuming the CWC percent time in each flight regime.

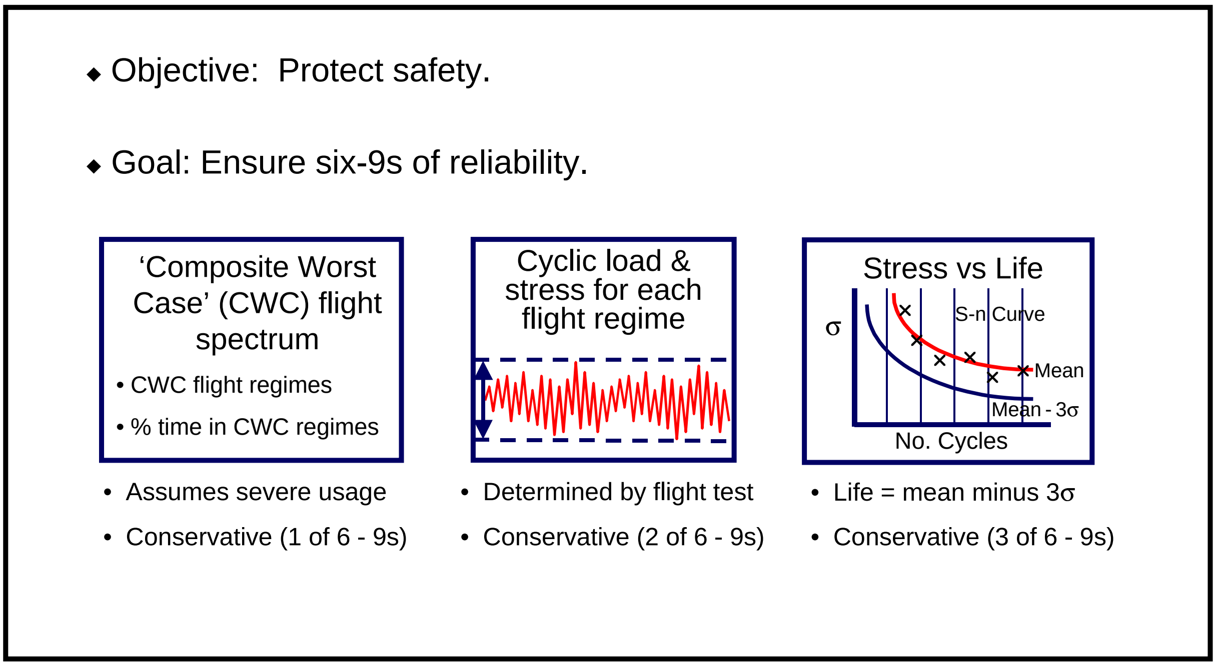

Figure 1: Component Replacement Time (CRT) Determination

The benefits of individual component fatigue life monitoring are shown schematically in Figure 2. As previously discussed, CRTs are based upon the assumption that each component always experiences the CWC usage. There is a safety issue if, however, a component experiences continual usage that is more severe than the CWC usage, and is not replaced until it reaches the defined CRT. Although this is not typical, it does occasionally occur, especially in the training environment. On the other hand, components typically experience usage that is much less than the CWC usage. An economic opportunity would be missed if these parts were replaced when they reached their CRT. Previous helicopter structural usage monitoring and component fatigue life tracking programs have shown that the actual safe usage of monitored components are from two to four-times the number of CRT flight hours. The structural usage monitoring and fatigue life tracking approach used for the HH-60J evaluation consisted of identifying the actual time spent in each of the CWC flight regimes and adjusting the component fatigue damage accumulation accordingly.

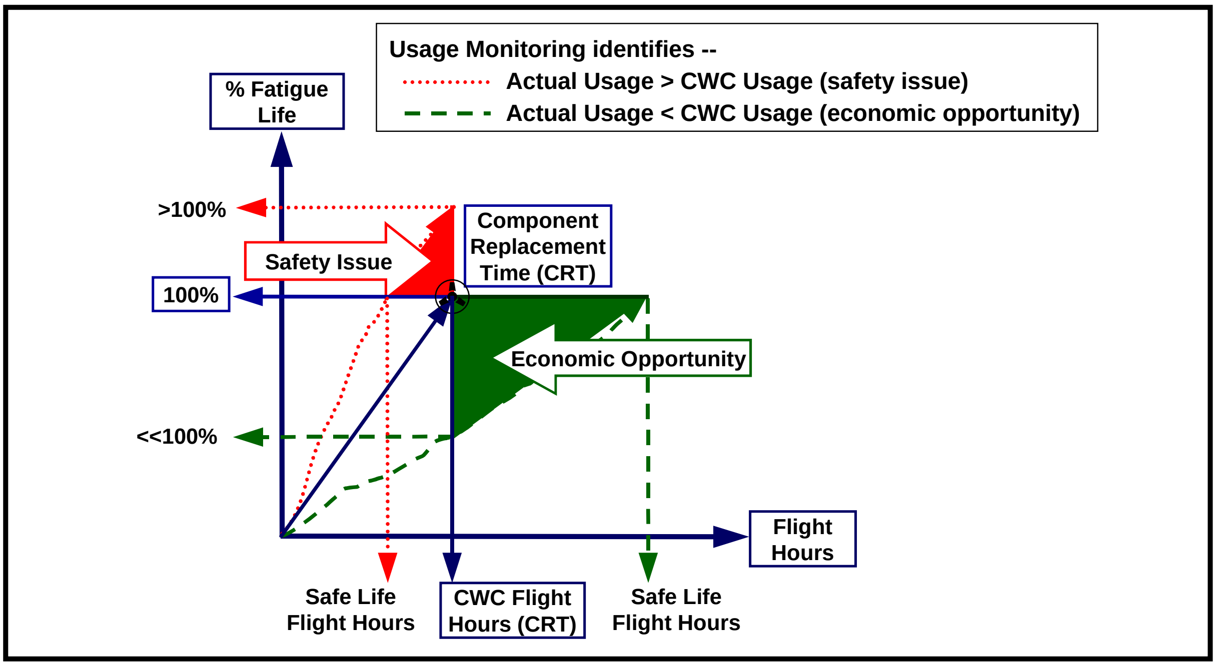

Figure 2: Benefits of Fatigue Life Monitoring

APPROACH

The fatigue life tracking approach is based upon flight regime recognition and is shown schematically in Figure 3.

- Block (1): The HH-60J original equipment manufacturer (OEM), Sikorsky, developed HH-60J CWC flight regimes that represent conservative HH-60J usage and,
- Block (2): measured and adjusted component flight test loads, and
- Block (3): determined fatigue damage rates for each component and each flight regime, as illustrated in Figure 1 and previously discussed. This information was made available as Government Furnished Data (GFD).
- Block (4): the flight parameters that are required to define flight regimes were identified, and
- Block (5) flight data acquisition systems used for capturing the flight regime parameters were installed in four HH-60Js at Coast Guard Air Station Cape Cod.
- Block (6): flight parameter time histories were recorded on each of the four aircraft during operational usage for six months (12/98 – 5/99), the data were periodically retrieved and processed to determine the amount of time spent in each CWC flight regime.
- Block (7): by tracking each flight critical component by serial number,
- Block (8): the amount of time each component spent in each CWC flight regime was determined using flight regime recognition algorithms and software.
- Block (9): component accumulated fatigue damage was calculated by multiplying the CWC damage rates of Block (3) by the amount of time spent in each CWC flight regime of Block (8).
- Block (10): the accumulated component damage was extrapolated to project the fatigue life of each component and provide the basis for the safety and economic benefits evaluation.

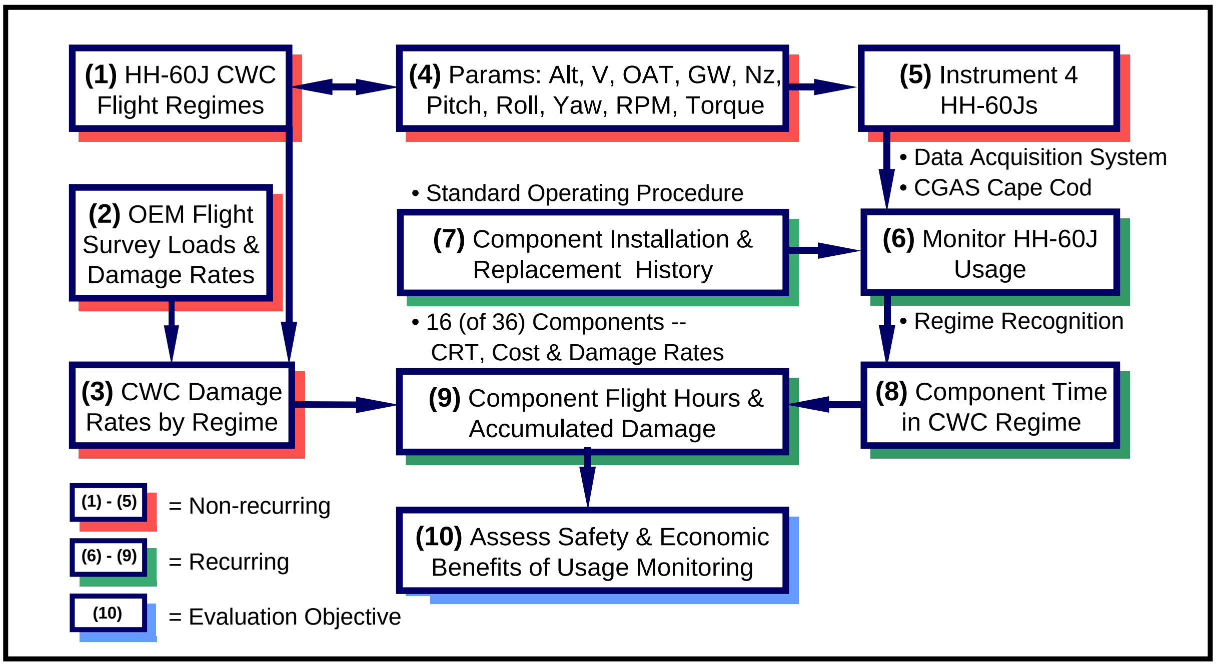

Figure 3: HH-60J Project Approach

IMPLEMENTATION

The low-cost, proof of concept structural usage monitoring approach implemented for the HH-60J evaluation is based upon the same CWC data as used for determining the component CRTs, with all the following conservatism and limitations:

- Based upon 'Composite Worst Case' (CWS) flight regimes –
    - Assumes worst case parameter values, such as –
        - Altitude
        - Airspeed (except for level flight)
        - GW & CG
        - Engine torque
        - Control Position (used for low speed & control reversal regimes)
    - Lacks 'granularity' –
        - Angle of Bank (defined at 30^{o}, 45^{o} & 60^{o} only)
        - Pullout (defined at 1.5 & 2.0g only)

Life Limited Components

The HH-60J life flight critical components that were monitored during this evaluation were selected based upon availability of the following component information –

- Component replacement times (CRTs)
- Component replacement cost
- Component fatigue damage accumulation rates

Information required for the evaluation was available for 16 of the 36 HH-60J flight critical components. The CRT and damage accumulation rates were obtained from Sikorsky Report SER-521343. The 16 selected components, retirement times and replacement costs, per ship set (SS), are shown in Figure 4.

	Component	HH-60J CRT(Hrs)	Replace Cost(SS)		Component	HH-60J CRT(Hrs)	Replace Cost(SS)
1	Main Gearbox Housing	5000	$48510	9	Main Rotor Blade	12000	$357840
2	Aft Walking Beam	14000	$1071	10	Lateral Bellcrank	12000	$5060
3	MR Controls - Forward	11000	$1769	11	Aft Bellcrank	17000	$1354
4	Aft Support Bridge	5400	$4020	12	Main Support Bridge	3500	$11220
5	MR Rotating Swashplate	3600	$14860	13	MR Shaft Extender	10000	$17884
6	MR Pitch Control Horn	4100	$43880	14	Main Rotor Shaft	11000	$17202
7	MR Blade Fold Hinge	5600	$7880	15	Main Rotor Hub	10000	$62480
8	MR Blade Tip Cap	6700	$26360	16	MR Controls Right Tierod	10000	$932

Figure 4: Component Replacement Times & Costs

Flight Regimes

HH-60J CWC flight regimes were developed by Sikorsky and are shown in Figure 5.

No.	HH-60J Regime	No.	HH-60J Regime	No.	HH-60J Regime
1	Taxi, Taxi Turns, Braking	18	Left Sideslip	35	Long. Rev. Fwd. Flt.
2	Hover	19	Right Sideslip	36	Lat. Rev. Hover
3	Left Hover Turn	20	Jump Take-Off	37	Lat. Rev. Fwd. Flt.
4	Right Hover Turn	21	Rolling Take-Off (30KCAS)	38	Yaw Rev. Hover
5	Left Side Flight	22	Vertical Take-Off	39	Yaw Rev. Fwd. Flt.
6	Right Side Flight	23	T.0. Power Climb	40	2.0g Pullout
7	Rearward Flight	24	Max. Cont. Power Climb	41	1.5g Pullout
8	Level Flight @0.2Vh	25	Approach	42	Power Dive
9	Level Flight @0.4Vh	26	Landing From Hover	43	Auto. Entry
10	Level Flight @0.5Vh	27	Run-On Landing	44	Steady Auto.
11	Level Flight @0.6Vh	28	30 Deg. AOB Left Turn	45	Auto. Recovery
12	Level Flight @0.7Vh	29	30 Deg. AOB Right Turn	46	Auto. Left Turn
13	Level Flight @0.8Vh	30	45 Deg. AOB Left Turn	47	Auto. Right Turn
14	Level Flight @0.9Vh	31	45 Deg. AOB Right Turn	48	Control Rev. Auto.
15	Level Flight @1.0Vh	32	60 Deg. AOB Left Turn	49	Auto. Approach
16	Level Flight @1.15Vh	33	60 Deg. AOB Right Turn	50	Auto. Approach & Landing
17	Partial Power Descent	34	Long. Rev. Hover	51	Droop Stop Pounding
				52	G.A.G. (W/Rotor Stop)

Figure 5: Composite Worst Case Flight Regimes

Damage Accumulation Rates

CWC damage accumulation rates for each component and each of the 52 CWC flight regimes are shown in Figure 6. Notice that many CWC flight regimes produce damage in few, if any, of the 16 components.

HH-60J		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
No.	Regime	MGB Housing: Left Tierod Attach Pad Mode	Aft Walking Beam: Non-Chafing Mode	MR Fwd. Bellcrank: Non-Chafing Mode	Aft Support Bridge: Chafing At MGB Attach Mode	MR Rotating Swashplate: MR Pushrod Attach Area Mode	MR Pitch Control Horn: Hub Clearance Cut-Out Radius Mode	MR Blade Fold Hinge: Flold Lock Pin Lug Bore Mode	MR Blade Tip Cap: Top Corner Radius Mode	Main Rotor Blade: Spar Outboard Bolt Hole Chafing Mode	Lateral Bellcrank: Potential No Chafing Mode	Aft Bellcrank: Potential Non-Chafing Mode	Main Support Bridge: Center Section Adj To Lat Bell Lug Mode	MR Shaft Extender: Shaft Attach Shoulder Chafing Mode	Main Rotor Shaft: Potential Non-Chafing Mode	Main Rotor Hub: Arm Lag Side Observation hole Mode	MR Controls Right Tierod: Non-Chafing Mode
		Damage Fraction per Flight Hour															
1	Taxi, Taxi Turns, Braking	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
2	Hover	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
3	Left Hover Turn	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
4	Right Hover Turn	0	0	0	0	0	0	0	0.00015	0	0	0	0	0	0	0	0
5	Left Side Flight	0	0	0	0	0	0	0	0.00014	0	0	0	0	0	0	0	0
6	Right Side Flight	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7	Rearward Flight	0	0	0	0	0	0	0.01629	0.00014	0.00014	0	0	0	0	0	0	0
8	Level Flight @0.2Vh	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9	Level Flight @0.4Vh	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10	Level Flight @0.5Vh	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11	Level Flight @0.6Vh	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
12	Level Flight @0.7Vh	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
13	Level Flight @0.8Vh	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
14	Level Flight @0.9Vh	0	0	0	0	0	0	0	0.00019	0	0	0	0	0	0	0	0
15	Level Flight @1.0Vh	0	0	0	0	0.00012	0	0	0.00558	0	0	0	0	0	0	0	0.00041
16	Level Flight @1.15Vh	0	0	0	0	0.00270	0.00250	0	0.01810	0	0	0	0	0	0	0	0
17	Partial Power Descent	0	0	0	0	0	0	0	0.00010	0	0	0	0	0	0	0	0.00214
18	Left Sideslip	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
19	Right Sideslip	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
20	Jump Take-Off	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
21	Rolling Take-Off (30KCAS)	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
22	Vertical Take-Off	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
23	T:0, Power Climb	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
24	Max, Cont, Power Climb	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
25	Approach	0	0	0	0	0	0	0	0.00019	0.00050	0	0	0	0	0	0	0
26	Landing From Hover	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
27	Run-On Landing	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.00065	0
28	30 Deg. AOB Left Turn	0.00015	0	0	0	0.00097	0.00064	0	0.00070	0	0	0	0.00006	0	0	0	0.00506
29	30 Deg. AOB Right Turn	0.00552	0	0	0	0.00021	0.00009	0	0.00045	0	0	0	0	0	0	0	0.00727
30	45 Deg. AOB Left Turn	0.03350	0.00033	0.00450	0.00500	0.00383	0.00367	0.00167	0.00483	0.00067	0	0.00017	0.00733	0	0	0	0.04983
31	45 Deg. AOB Right Turn	0.09550	0	0.02733	0.00850	0.00733	0.00950	0	0.00333	0.00017	0	0	0.02283	0	0	0.00050	0.06717
32	60 Deg. AOB Left Turn	0.07250	0.00500	0.02750	0.01250	0.02000	0.02500	0.01500	0.01500	0.00250	0	0.00250	0.04000	0.05250	0.07000	0.01000	0.11000
33	60 Deg. AOB Right Turn	0.08250	0.01500	0.11250	0.02250	0.02500	0.03750	0.03250	0.03000	0.00250	0.00250	0.01000	0.03000	0.03000	0.03750	0.01000	0.15750
34	Long. Rev. Hover	0	0	0	0	0	0	0	0	0	0	0	0	0.00714	0.00714	0.00429	0
35	Long. Rev. Fwd. Flt.	0	0	0	0	0.00091	0	0	0.01636	0	0	0	0	0.00091	0	0.00091	0
36	Lat. Rev. Hover	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37	Lat. Rev. Fwd. Flt.	0	0	0	0	0.00091	0	0	0.01636	0	0	0	0	0	0	0	0.00545
38	Yaw Rev. Hover	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
39	Yaw Rev. Fwd. Flt.	0	0	0	0	0	0	0	0.00364	0	0	0	0	0	0	0	0
40	2.0g Pullout	0.04667	0	0.08333	0.00667	0.04667	0.08667	0.06000	0.11000	0.01000	0.01333	0	0.01333	0.02000	0.03333	0.01000	0.07667
41	1.5g Pullout	0.01581	0.00226	0.00968	0.00323	0.00548	0.00806	0.00548	0.00484	0.00032	0.00548	0.00129	0.01452	0.00742	0.01065	0.00452	0.02742
42	Power Dive	0	0	0	0	0.00481	0.00538	0	0.03048	0	0	0	0	0	0	0	0.00019
43	Auto. Entry	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
44	Steady Auto.	0	0	0	0	0	0	0	0.00080	0	0	0	0	0	0	0	0
45	Auto. Recovery	0	0	0	0	0.00143	0	0	0.00286	0	0	0	0	0	0	0	0
46	Auto. Left Turn	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
47	Auto. Right Turn	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
48	Control Rev. Auto.	0	0	0	0	0	0	0	0	0	0	0	0	0.00500	0.00250	0.00125	0
49	Auto. Approach	0	0	0	0	0	0	0	0.00030	0	0	0	0	0	0	0	0
50	Auto. Approach & Landing	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
51	Droop Stop Pounding	0	0	0	0	0	0	0	0	0	0	0	0	0.00001	0.00001	0.00003	0
52	G.A.G.	0	0	0	0	0	0	0	0	0.00002	0	0	0	0	0	0	0

Figure 6: Component Damage Accumulation Rates

Data Acquisition System and Flight Parameters

The flight parameter time histories were captured using the data acquisition system shown schematically in Figure 7, and photographically in Figure 8. The following parameters, which are required for flight regime recognition, were sampled at 5Hz, filtered, in some cases calculated, and output to memory at 1Hz –

- Airspeed
- Altitude
- Rate of Climb
- Outside Air Temp.
- Engine Torque (2)
- Main Rotor Speed
- Vertical Acceleration
- Yaw Velocity
- Pitch Angle
- Roll Angle
- Weight On/Off Wheels
- Time

Data Acquisition System Installation

The system installation requirements for the HH-60J were defined, installation kits were designed and fabricated, and the systems were installed under the pilots' seats in four HH-60J aircraft at Coast Guard Air Station (CGAS) Cape Cod.

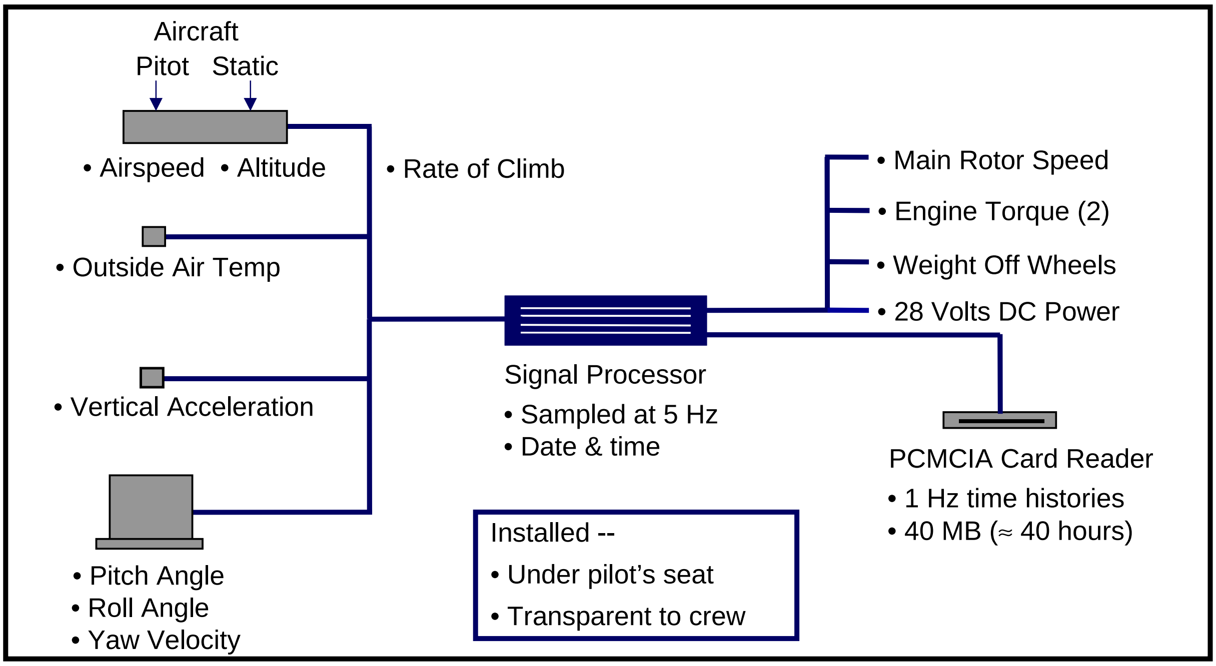

Figure 7: Data Acquisition System & Monitored Parameters

Data Retrieval

The data acquisition system 40 Mb PCMCIA memory cards, which have the capacity of capturing approximately 40 flight hours of 1Hz aircraft usage data, were periodically removed from each of the four HH-60J aircraft and downloaded to retrieve the stored flight parameter time histories.

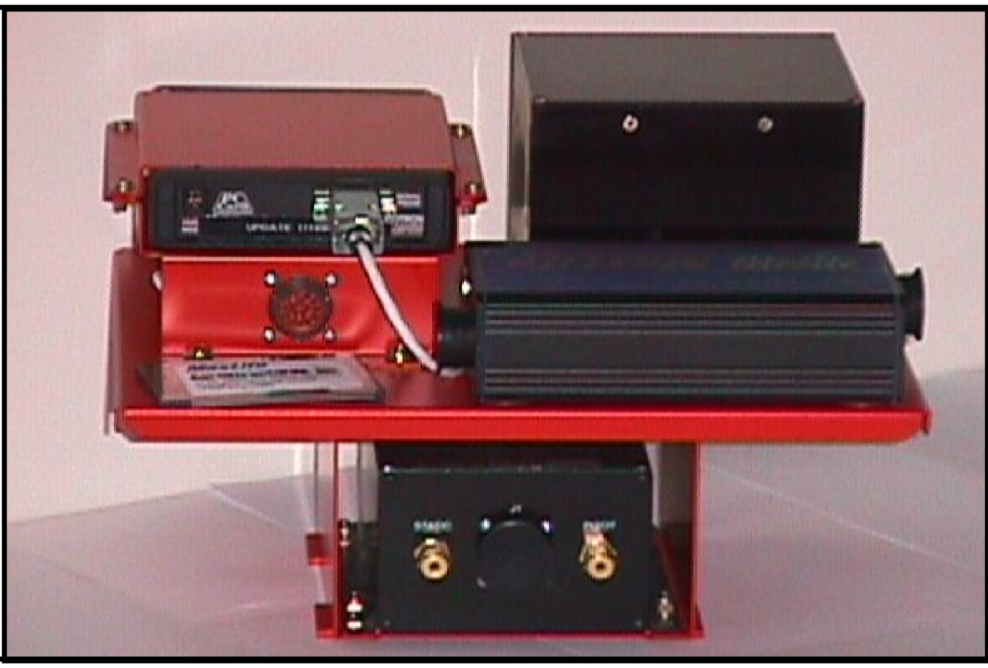

Figure 8: Data Acquisition System

Flight Regime Recognition

Existing flight regime recognition software and algorithms were used to recognize a total of 298 flight regimes. These regimes were originally developed and validated for the U.S. Navy and U.S. Marine Corps CH-46. They were also used for flight regime recognition for the U.S. Army UH-60 in the Digital Source Collector (DSC) Program, and for the SH-60B in the U.S. Navy Joint Advanced Health and Usage Monitoring System (JAHUMS) Program. These legacy flight regimes were also used on two U.S. Navy contracts for structural usage monitoring of the HH-60H, SH-60F and SH-60B. The 298 flight regime times were mapped over to the 52 HH-60J CWC flight regimes.

Flight Regime Recognition Refinement

Mapping of flight regimes from the legacy software and algorithms to the HH-60J required considerable investigation, analysis and judgement. For example, the existing flight regime recognition software and algorithms define any maneuver above 1.25g as being a pullout. The HH-60J CWC flight regimes define pullouts at 1.5 and 2.0g. Therefore, the output of the regime recognition program was post-processed to be consistent with the CWC flight regimes. In addition, the legacy flight regime recognition software collects angle of bank data at 10^o to 25^o, 25^o to 35^o, 35^o to 50^o and above 50^o. Since the HH-60J CWC angle of bank flight regimes are defined at 30^o, 45^o and 60^o degrees, these data were post processed to coincide with the HH-60J CWC flight regime definition.

EVALUATION RESULTS

HH-60J Total Flight Hours

A total of 668 flight hours of data was captured during the period of 4 Dec 98 to 15 May 99. The number of flight hours processed and evaluated for each of the four aircraft is as follows –

HH-60J	Hours Processed
#6001	135.5 hrs
#6007	220.4 hrs
#6014	182.8 hrs
#6021	129.6 hrs
Total	668.3 hrs

HH-60J Time in Regime

The average percent time spent by the four aircraft in each HH-60J flight regime is shown in Figure 9. Assumptions made in establishing this usage evaluation include:

- Droop stop pounding occurs once per flight.
- Half of the ground-air-ground (GAG) cycles end with a full rotor stop.
- Design values of usage are assumed for low-speed and control reversal conditions. Actual monitoring of these conditions would require additional instrumentation and was out of scope for this proof-of-concept evaluation.

	HH-60J CWC Regime	Tot/Avg		HH-60J CWC Regime	Tot/Avg		HH-60J CWC Regime	Tot/Avg
	Flight Hours	668.3		Flight Hours	668.3		Flight Hours	668.3
1	Taxi, Taxi Turns, Braking	0.6049	18	Left Sideslip	0.4000	35	Long. Rev. Fwd. Flt.	0.1100
2	Hover	0.0224	19	Right Sideslip	0.4000	36	Lat. Rev. Hover	0.1400
3	Left Hover Turn	0.0113	20	Jump Take-Off	0.0300	37	Lat. Rev. Fwd. Flt.	0.1100
4	Right Hover Turn	0.0093	21	Rolling Take-Off (30KCAS)	0.3400	38	Yaw Rev. Hover	0.1400
5	Left Side Flight	0.7000	22	Vertical Take-Off	0.0900	39	Yaw Rev. Fwd. Flt.	0.1100
6	Right Side Flight	0.7000	23	T.O., Power Climb	1.4534	40	2.0g Pullout	0.0014
7	Rearward Flight	0.7000	24	Max Cont. Power Climb	0.9395	41	1.5g Pullout	0.1493
8	Level Flight @0.2Vh	10.8188	25	Approach	1.6000	42	Power Dive	0.0000
9	Level Flight @0.4Vh	9.1262	26	Landing From Hover	0.1200	43	Auto. Entry	0.0700
10	Level Flight @0.5Vh	3.4567	27	Run-On Landing	0.6200	44	Steady Auto.	1.0000
11	Level Flight @0.6Vh	15.4962	28	30 Deg. AOB Left Turn	1.4821	45	Auto. Recovery	0.0700
12	Level Flight @0.7Vh	5.0379	29	30 Deg. AOB Right Turn	1.3013	46	Auto. Left Turn	0.4200
13	Level Flight @0.8Vh	26.2673	30	45 Deg. AOB Left Turn	0.2492	47	Auto. Right Turn	0.4200
14	Level Flight @0.9Vh	7.9380	31	45 Deg. AOB Right Turn	0.1476	48	Control Rev. Auto.	0.0800
15	Level Flight @1.0Vh	0.0872	32	60 Deg. AOB Left Turn	0.0000	49	Auto. Approach	0.6700
16	Level Flight @1.15Vh	0.0006	33	60 Deg. AOB Right Turn	0.0000	50	Auto. Approach & Landing	0.0030
17	Partial Power Descent	0.7726	34	Long. Rev. Hover	0.1400	51	Droop Stop Pounding	44.64
						52	G.A.G.	44.64

Figure 9: HH-60J % Time in Flight Regimes

HH-60J Component Fatigue Life

The process for determining fatigue damage accumulation and fatigue life, in flight hours, for each of the selected components, is shown schematically in Figure 3.

Each of the components selected for evaluation is listed in Figure 4. The component fatigue damage fraction per flight hour is obtained by multiplying the Figure 6 damage rates by the Figure 9 percent time spent in each flight regime. The fatigue damage accumulation, in percent of fatigue life available, is obtained by multiplying the fatigue damage fraction per flight hour by the actual number of hours flown. For example, the percent fatigue life expended, in 668.3 hours of monitored usage, by Component 1 (Main Gearbox Housing) for the average of the four aircraft is:

Regime	Damage Rate (Fig. 6)		%Time in Regime (Fig. 9)		%Life/Flt Hr
28	0.000152	x	1.4821	=	0.000225
29	0.005515	x	1.3013	=	0.007177
30	0.033500	x	0.2492	=	0.008348
31	0.095500	x	0.1476	=	0.014094
32	0.072500	x	0.0000	=	0.000000
33	0.082500	x	0.0000	=	0.000000
40	0.046667	x	0.0000	=	0.000000
41	0.015806	x	0.1493	=	0.002360
42	0.000020	x	22.32 (occur)	=	0.000436

Total % life expended per flight hour = 0.032706%

Projected total fatigue life: (100% ÷ 0.032706) = 3057.6 flt hrs

The total flight hour fatigue lives for each of the 16 USCG monitored components, shown in Figure 10, are projected from actual HH-60J usage at CGAS Cape Cod. Resultant component lives, that are less than the HH-60J design CRT, are shaded in blue. Component lives that exceed 20000 hours are truncated to 20000 hours.

If the CGAS Cape Cod HH-60J aircraft do not exceed bank angles greater than ±45 degrees, the fatigue lives would increase as shown in Figure 10.

			Fatigue Life - Flight Hours			Cost per Flight Hour		
	Component	Replace Cost	HH-60J CRT	USCG Usage	USCG Usage <45°	HH-60J CRT	USCG Usage	USCG Usage <45°
1	Main Gearbox Housing	$48,510	5000	3058	8997	$9.70	$15.87	$5.39
2	Aft Walking Beam	$1,071	14000	20000	20000	$0.08	$0.05	$0.05
3	MR Cntls - Fwd Bellcrank	$1,769	11000	14044	20000	$0.16	$0.13	$0.09
4	Aft Support Bridge	$4,020	5400	20000	20000	$0.74	$0.20	$0.20
5	MR Rotating Swashplate	$14,860	3600	19790	20000	$4.13	$0.75	$0.74
6	MR Pitch Control Horn	$43,880	4100	20000	20000	$10.70	$2.19	$2.19
7	MR Blade Fold Hinge	$7,880	5600	6606	6793	$1.41	$1.19	$1.16
8	MR Blade Tip Cap	$26,360	6700	7625	8577	$3.93	$3.46	$3.07
9	Main Rotor Blade	$357,840	12000	20000	20000	$29.82	$17.89	$17.89
10	Lateral Bellcrank	$5,060	12000	20000	20000	$0.42	$0.25	$0.25
11	Aft Bellcrank	$1,354	17000	20000	20000	$0.08	$0.07	$0.07
12	Main Support Bridge	$11,220	3500	13129	20000	$3.21	$0.85	$0.56
13	MR Shaft Extender	$17,884	10000	20000	20000	$1.79	$0.89	$0.89
14	Main Rotor Shaft	$17,202	11000	20000	20000	$1.56	$0.86	$0.86
15	Main Rotor Hub	$62,480	10000	20000	20000	$6.25	$3.12	$3.12
16	MR Cntls Rt Tierod	$932	10000	2176	3854	$0.09	$0.43	$0.24
					Total	74.08	48.21	36.80

Figure 10: HH-60J Component Fatigue Life

COMPARISON OF HH-60J & HH-60H USAGE & COMPONENT FATIGUE LIVES

The U.S. Navy sponsored a program to monitor the actual operational usage of three U.S. Navy HH-60H aircraft for a period of four months. The purpose of the program was to evaluate and demonstrate the benefits of structural usage monitoring.

HH-60H Total Flight Hours

A total of 295 flight hours of data was captured during the period of 24 Feb 99 to 20 Jul 99. The number of flight hours processed and evaluated for each of the three aircraft is as follows –

BUNO	Squadron	Location	Hours Processed
#163788	HCS-5	PT MUGU	109.85 hrs
#163800	HCS-5	PT MUGU	68.05 hrs
#165117	HS-2	NORIS	116.75 hrs
		Total:	294.65 hrs

HH-60H Time in Regime

The average percent time spent in the HH-60H flight regimes is shown in Figure 11. The same assumptions were made for the HH-60H as for the HH-60J.

	HH-60H CWC Regime	Tot/Avg		HH-60H CWC Regime	Tot/Avg		HH-60H CWC Regime	Tot/Avg
	Flight Hours	294.6500		Flight Hours	294.6500		Flight Hours	294.6500
1	Taxi, Taxi Turns, Braking	1.5694	18	Left Sideslip	0.4000	35	Long. Rev. Fwd. Flt.	0.1100
2	Hover	0.0301	19	Right Sideslip	0.4000	36	Lat. Rev. Hover	0.1400
3	Left Hover Turn	0.0238	20	Jump Take-Off	0.0300	37	Lat. Rev. Fwd. Flt.	0.1100
4	Right Hover Turn	0.0138	21	Rolling Take-Off (30KCAS)	0.3400	38	Yaw Rev. Hover	0.1400
5	Left Side Flight	0.7000	22	Vertical Take-Off	0.0900	39	Yaw Rev. Fwd. Flt.	0.1100
6	Right Side Flight	0.7000	23	T:0, Power Climb	1.5339	40	2.0g Pullout	0.0000
7	Rearward Flight	0.7000	24	Max, Cont, Power Climb	2.1716	41	1.5g Pullout	0.0470
8	Level Flight @0.2Vh	9.3026	25	Approach	1.6000	42	Power Dive	0.0000
9	Level Flight @0.4Vh	20.0038	26	Landing From Hover	0.1200	43	Auto. Entry	0.0700
10	Level Flight @0.5Vh	0.1829	27	Run-On Landing	0.6200	44	Steady Auto.	1.0000
11	Level Flight @0.6Vh	15.7443	28	30 Deg. AOB Left Turn	1.5882	45	Auto. Recovery	0.0700
12	Level Flight @0.7Vh	8.7123	29	30 Deg. AOB Right Turn	1.5666	46	Auto. Left Turn	0.4200
13	Level Flight @0.8Vh	6.4450	30	45 Deg. AOB Left Turn	0.1469	47	Auto. Right Turn	0.4200
14	Level Flight @0.9Vh	5.4070	31	45 Deg. AOB Right Turn	0.1212	48	Control Rev. Auto.	0.0800
15	Level Flight @1.0Vh	0.0424	32	60 Deg. AOB Left Turn	0.0040	49	Auto. Approach	0.6700
16	Level Flight @1.15Vh	0.0000	33	60 Deg. AOB Right Turn	0.0020	50	Auto. Approach & Landing	0.0030
17	Partial Power Descent	2.0340	34	Long. Rev. Hover	0.1400	51	Droop Stop Pounding	50.55
						52	G.A.G.	50.55

Figure 11: HH-60H % Time in Flight Regimes

HH-60H Component Fatigue Life

The process for determining HH-60H damage accumulation and fatigue life is the same as for the HH-60J. The total flight hour fatigue lives for each of the 16 USN monitored components, shown in Figure 12, are projected from actual HH-60H usage at Point Mugu (PT MUGU) and North Island (NORIS). Component lives that exceed 20000 hours are truncated to 20000 hours.

HH-60J and HH-60H Monitored Usage

The USCG HH-60J CRTs are based upon the USN SH-60B. The SH-60B, an anti-submarine and anti-ship aircraft, is equipped with extensive mission equipment, and flight critical components have relatively long CRTs. The USN HH-60H is a search and rescue aircraft and has relative short CRTs. A comparison of the monitored usage of the HH-60H with the HH-60J shows the HH-60H is generally flown similar to, or more mildly than the HH-60J. This is illustrated in Figure 13.

HH-60J and HH-60H CRTs and Operational Usage Fatigue Lives

A comparison of the HH-60J and HH-60H CRTs and operational usage fatigue lives is shown in Figure 14. Note that although the HH-60J component projected fatigue lives are less than the HH-60H, the HH-60J CRTs are, in most cases, significantly greater than the HH-60H CRTs.

			Fatigue Life - Flight Hours			Cost per Flight Hour		
	Component	Replace Cost	HH-60H CRT	USN Usage	USN Usage <45^{o}	HH-60H CRT	USN Usage	USN Usage <45^{o}
1	Main Gearbox Housing	$48,510	820	3695	9242	$59.16	$13.13	$5.25
2	Aft Walking Beam	$1,071	12000	20000	20000	$0.09	$0.05	$0.05
3	MR Cntls - Fwd Bellcrank	$1,769	2200	19171	20000	$0.80	$0.09	$0.09
4	Aft Support Bridge	$4,020	7400	20000	20000	$0.54	$0.20	$0.20
5	MR Rotating Swashplate	$14,860	3600	20000	20000	$4.13	$0.74	$0.74
6	MR Pitch Control Horn	$43,880	3100	20000	20000	$14.15	$2.19	$2.19
7	MR Blade Fold Hinge	$7,880	2800	6773	6947	$2.81	$1.16	$1.13
8	MR Blade Tip Cap	$26,360	710	8521	9377	$37.13	$3.09	$2.81
9	Main Rotor Blade	$357,840	12000	20000	20000	$29.82	$17.89	$17.89
10	Lateral Bellcrank	$5,060	4400	20000	20000	$1.15	$0.25	$0.25
11	Aft Bellcrank	$1,354	15000	20000	20000	$0.09	$0.07	$0.07
12	Main Support Bridge	$11,220	2800	19983	20000	$4.01	$0.56	$0.56
13	MR Shaft Extender	$17,884	9100	20000	20000	$1.97	$0.89	$0.89
14	Main Rotor Shaft	$17,202	6600	20000	20000	$2.61	$0.86	$0.86
15	Main Rotor Hub	$62,480	9300	20000	20000	$6.72	$3.12	$3.12
16	MR Cntls Rt Tierod	$932	650	2376	3632	$1.43	$0.39	$0.26
					Total	166.61	44.71	36.38

Figure 12: HH-60H Component Fatigue Life

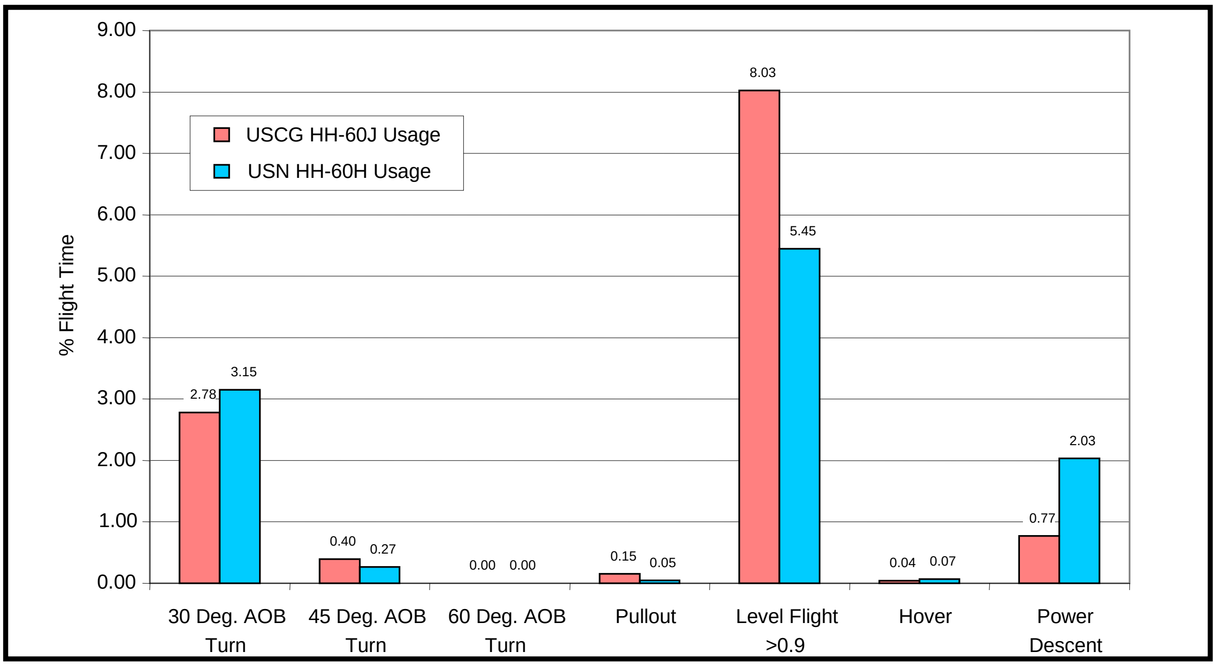

Figure 13: HH-60J & HH-60H Operational Usage

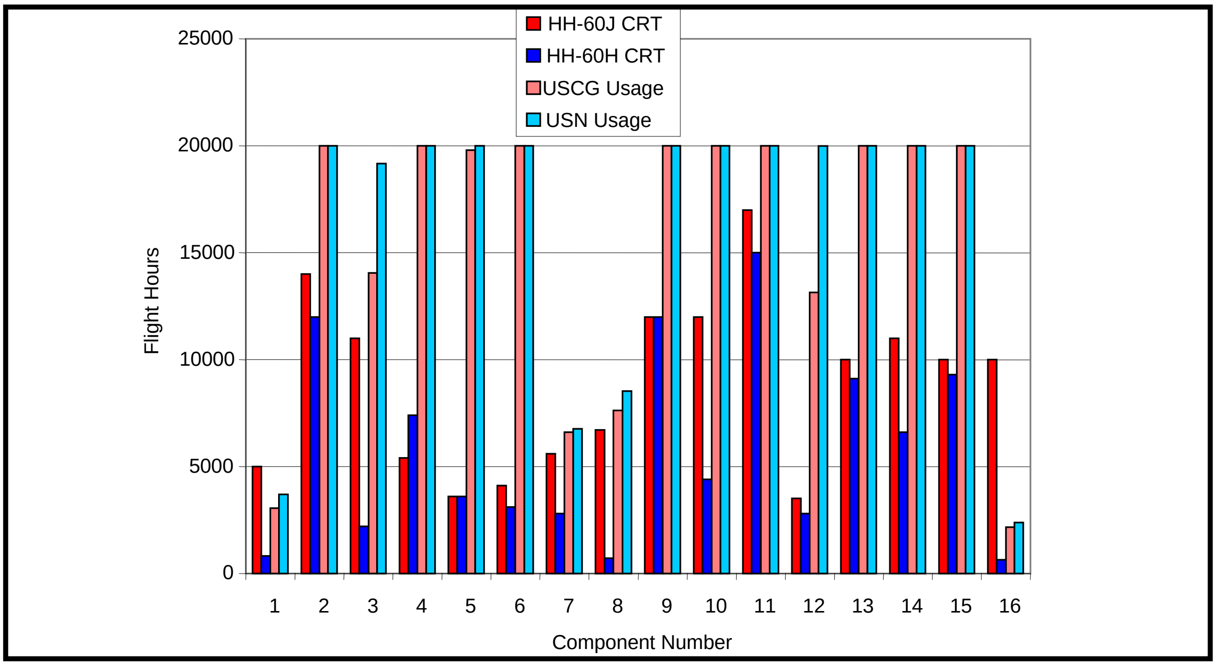

Figure 14: HH-60J & HH-60H Component Lives

HH-60J & HH-60H Component Cost per Flight Hour

Component replacement costs are calculated on a per-flight-hour basis. For example, as shown in Figure 10, the replacement cost for HH-60J Component 1 (Main Gearbox Housing) is $48,510 and the CRT of the gearbox housing is 5000 hours. The life for the gearbox housing, based on average Cape Cod usage, is 3058 hours, and 8997 hours if angle of bank (AOB) does not exceed 45°. A summary of the replacement cost per flight hour for each condition follows –

Component 1 – Main Gearbox Housing: Cost = $48,510 (Figure 10)

<u>Component Replacement Time</u> (Figure 10)			<u>Replacement Cost per Flight Hour</u>
HH-60J Design CRT	=	5000 hrs	$48,510 ÷ 5000 = $9.70 per flt hr
Avg. Cape Cod Usage	=	3058 hrs	$48,510 ÷ 3058 = $15.87 per flt hr
Avg. Usage <45° AOB	=	8997 hrs	$48,510 ÷ 8997 = $5.39 per flt hr

The cost for replacing the HH-60J components, based upon actual monitored usage as compared to USCG design CRTs, is also summarized in Figure 10. The total CRT and monitored cost for all 16 components is displayed at the bottom of Figure 10.

The same analysis was performed for the 16 HH-60H components and the results are summarized in Figure 12.

CONCLUSIONS

The data used for the HH-60J program were somewhat limited in specificity and granularity. However, there is an indication that the HH-60J is operated in a manner similar to the HH-60H, but is being managed to longer CRTs. The USCG, USN and Sikorsky have been evaluating these findings.

The proof-of-concept for low-cost structural usage monitoring and component fatigue life tracking was successfully demonstrated. In addition, the following results were obtained --

- Evaluation of the information, obtained by monitoring operational usage of four HH-60J aircraft at CGAS Cape Cod, identified opportunities to enhance both safety and economics in the operation of the HH-60J.

- <u>Safety</u>: HH-60J operational usage at Cape Cod results in components that experience fatigue damage accumulation at rates in excess of design usage spectrum rates. To protect safety, these components should be monitored and retired prior to reaching the current design CRT. This would have an adverse impact on the budgets required for component replacement.

- <u>Economics</u>: Operational usage also results in components that experience fatigue damage accumulation at rates less than design usage spectrum rates. By monitoring operational usage, most components are projected to have safe fatigue lives in excess of the current design CRT.

- <u>Safety and Economic Opportunity</u>: By monitoring actual operational usage, and limiting the time spent in high bank angle maneuvers, the USCG can decrease the rate of fatigue damage accumulation and reduce operating cost.

HELICOPTER OPERATIONAL FLIGHT DATA MONITORING

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ABSTRACT

Over recent years there has been a growing appreciation amongst fixed wing airlines that Operational Flight Data Monitoring (OFDM) can significantly enhance operational safety. Large public transport helicopters such as those operating over the North Sea have been equipped with Flight Data Recorders since the early 1990s. The opportunity therefore exists to apply the concept of OFDM to helicopters. This paper describes the implementation of a trial to evaluate the potential of a helicopter OFDM programme. Although the trial is still under way, the feasibility of such a programme has already been established and benefits are being demonstrated.

INTRODUCTION

There is now widespread acceptance in the airline industry that Operational Flight Data Monitoring (OFDM) programmes can improve safety, whilst providing a number of other operational and economic benefits. These programmes monitor flight operations by routinely analysing aircraft flight data to detect deviations from normal, expected, or flight manual practice. They provide continuous operational quality control with timely feedback on sub-standard practices, and produce valuable information for the evaluation and improvement of procedures. However, until now, there has been no attempt to apply the concept of OFDM to large public transport helicopters.

The UK Civil Aviation Authority (CAA) have, for many years, been working to improve the safety of helicopters operating over the North Sea. Notable initiatives have included trials of Health and Usage Monitoring Systems (HUMS), introducing a mandatory requirement to fit Flight Data Recorders (FDRs), actively encouraging the fitting of HUMS and, most recently, mandating these systems. The CAA's latest initiative has been to instigate trials of an OFDM programme for helicopters operating over the North Sea, known as the Helicopter Operations Monitoring Programme (HOMP). This paper describes the HOMP trial, which represents the first application of OFDM to helicopters.

OPERATIONAL FLIGHT DATA MONITORING (OFDM)

The objective of OFDM programmes is to enable proactive safety intervention based on analysis of exceedances and trends in flight data obtained on a routine basis from line operations. OFDM programmes are owned and operated by the airlines. Such programmes are powerful pro-active tools for enhancing flight operational safety standards. They improve safety by continuously monitoring operations, detecting adverse trends in operational behaviour, and detecting weaknesses in crews, the aircraft, at certain airports, or in Air Traffic Control (ATC). Reference [1] defines an OFDM system as "A systematic method of accessing, analysing and acting upon information obtained from digital flight data records of routine flight operations."

OFDM systems allow an airline to identify, quantify, assess and address operational risks. The objectives of an airline OFDM system are to:

- Identify and quantify operational risks - identify areas of risk and determine when non-standard, unusual or basically unsafe circumstances occur in operations.

- Assess the risks to determine which are not acceptable. The system will primarily be used to deduce whether there is a trend towards unacceptable risk prior to it reaching a level which would indicate that the safety management process has failed.
- Implement remedial activity where risks are not acceptable. Once an unacceptable risk, either actually present or predicted by trending, has been identified then the appropriate risk mitigation techniques must be applied.
- Measure the effectiveness of remedial action and continue to monitor risks. Once a remedial action has been put in place it is critical that it's effectiveness, in terms of reducing the original identified risk and also not increasing risk elsewhere, is confirmed.

The justification for OFDM programmes is based on the "Heinrich Pyramid" which postulates that for every 1 major accident there are 3-5 less significant accidents, 7-10 incidents, and several hundred unreported occurrences. Today's unreported occurrences are the building blocks of tomorrow's accidents and incidents. Obtaining information on them enables a pro-active approach to be taken to address operational risks before they result in incidents and accidents. Current thinking is that an effective solution to further improving safety in an already well regulated environment is not to regulate more, punish more or increase training, but is to obtain better information on operational risks, and provide positive feedback to improve the system. Only if the risks are known can steps be taken to reduce them, no information means that no action can be taken. Further information on OFDM is given in references [1] and [2].

THE BACKGROUND TO HELICOPTER OFDM

There are a number of factors which have led to the current work on an OFDM programme for North Sea helicopters. Most importantly, all helicopters are now FDR equipped so the required flight data is available.

The introduction of HUMS has reduced the rate of occurrence of helicopter accidents due to technical causes by providing better information on the integrity of the helicopter powertrain. However, about half the accidents involved operational factors, and there is thus scope for improving operational safety by providing better information on operational risks. This is particularly true for operations to/from offshore platforms, with environmental factors such as the close proximity of obstacles (e.g. cranes) and hot gas exhausts from turbines, and problems of structure-induced turbulence.

A final factor is the recognition that OFDM is now a well established practice amongst fixed wing operators, with demonstrated safety and other benefits. These include better targeting of training resources, feedback on the effectiveness of training, improved information for the formulation of operating policy, an ability to assess the effect of policy changes, and economic benefits (e.g. from reduced insurance premiums and improved aircraft performance). Airlines such as British Airways (BA) have already proven the benefits of OFDM. Their OFDM programme, known as SESMA, is an essential part of BA's safety management programme and is welcomed by aircrew, BALPA and management alike. BA are convinced that the SESMA programme has made a difference to their safety statistics. Further details of BA's experience are presented in references [3] and [4].

INITIAL HELICOPTER OFDM STUDIES

The current HOMP trial resulted from a two phase HOMP study carried out for the CAA by Stewart Hughes Limited (SHL), working in co-operation with Bristow Helicopters Limited (BHL).

The first phase was a HOMP feasibility study. This involved analysing a historical database of FDR data recorded over a period of one year from one Super Puma helicopter. A prototype set of helicopter operational 'events' were generated and the data was analysed using ATM's Flight Data Service (FDS) system, loaned to SHL by the CAA. An event is when one or more flight parameters exceed some predefined boundaries to indicate a deviation from normal, expected or flight manual practice. The study was able to demonstrate the potential benefits which could be achieved from a HOMP.

The second phase of the work was a HOMP implementation study. This involved consultations with all the UK based North Sea helicopter operators to obtain their thoughts and concerns regarding a HOMP. In addition, the lessons learned from BA's OFDM experience were analysed. A review of the available tools and equipment for a HOMP was carried out and recommendations were produced on a programme implementation. An example of the management of a full-scale HOMP is shown in Figure 1. Three key personnel are involved:

- A HOMP Operator. This person is responsible for the routine replay and analysis of the flight data and the initial verification of any events.
- A HOMP Manager. The HOMP Manager is a senior current pilot, ideally a training captain, and manages the overall programme, provides the 'expert pilot' input required for investigating events (calling on type-specific experience as required), and liaises with aircrew if necessary. The HOMP Manager reports to the Flight Safety Officer. Alternatively, if an operator does not have a full time Flight Safety Officer, the HOMP Manager could take over some or all of his responsibilities.
- The Flight Safety Officer. This person is responsible for deciding whether any actions need to be taken on the basis of the information provided by the HOMP Manager. Such actions could be changes to company operating and training procedures and manuals, special HOMP investigations, or changes to the programme itself.

The two phase study provided a clear justification for taking further steps towards the implementation of a HOMP, but identified a number of operator concerns and issues which ought to be addressed prior to any full scale implementation. These included perceived difficulties in effectively monitoring the operation of such a flexible air vehicle as a helicopter (in contrast to fixed wing aircraft which operate "on rails"), the effort required to interpret monitoring results, the handling of data from multiple sites, the overall workload of managing a HOMP, and the achievement of both management and aircrew "buy-in" to the programme. It was concluded that these concerns and issues could best be addressed by performing a HOMP trial on a limited number of aircraft. The objectives of this trial would be to:

- Establish how best to monitor helicopter flight operations, and to evaluate the safety benefits of this monitoring.
- Evaluate the tools and equipment selected for a HOMP and eliminate any technical risks associated with these.
- Establish and evaluate a programme management strategy, and determine the workload a HOMP would impose on a typical helicopter operator.

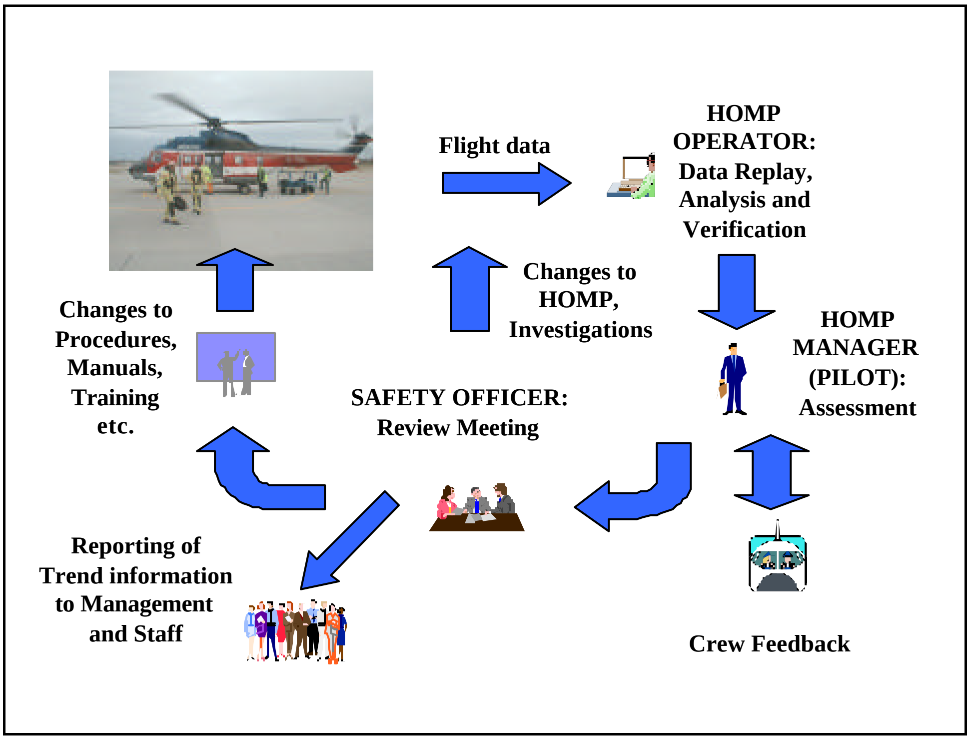

Figure 1: Example management of a full-scale HOMP

- Establish agreements between aircrew and management to ensure that the identity of aircrew is protected, and the focus is on positive feedback.
- Further expose industry to the concept of a HOMP, enabling a more informed consideration of its full scale implementation.

THE HOMP TRIAL

Following completion of the HOMP study, SHL put together a trial programme and team, and are now prime contractor to the CAA for the HOMP trial. BHL agreed to provide the helicopters for the trial, and to provide a senior training captain on a part time basis to run the HOMP at BHL. BAe Systems (formally Marconi Avionics) were selected to provide PCMCIA card based Quick Access Recorders to record the FDR data and BA were selected to provide the data replay and analysis software.

Trial equipment

The FDR data generated by the BHL Tiger IHUMS Data Acquisition and Processing Unit (DAPU) is written to a 20 MB PCMCIA card by the BAe Systems Card Quick Access Recorder (CQAR) shown in Figure 2. The CQAR is connected to the auxiliary output from the DAPU, which provides an ARINC 573 output. Data is written to the card continuously whenever the DAPU is powered up. The CQAR is fitted in the cockpit centre console.

Figure 2: CQAR

A data download PC has been provided for downloading PCMCIA card data from helicopters operating at a location which is remote from the analysis system. Zip disks are used to transfer the data to the analysis PC. The analysis PC can download data from PCMCIA cards or zip discs, and archive data to CDs and tape.

The HOMP software is based on four Flight Data modules in the British Airways Safety Information System (BASIS). North Sea helicopter operators are already using the BASIS Air Safety Reporting module. The four flight data modules are described below.

Flight Data Trace (FDT) performs the data replay, the primary 'event' and flight data measurement analysis, and displays flight data traces. The HOMP trial is the first application of this module. FDT provides facilities for the user to configure the FDR parameter decode, the calculation of derived parameters from the basic parameter set, the event definitions and event limits or constants, the flight data measurements, and flight data trace displays. FDT has two modes of operation. In data analysis mode it automatically replays all the downloaded flight data, detects events and extracts flight data measurements. In data investigation mode it provides numerical listings and graphical traces to enable the user to investigate and classify events. A typical trace display is shown in Figure 3.

Flight Data Simulation (FDS) allows the flight data to be viewed in the form of a cockpit instrument simulation. A new version of this module, simulating a Tiger cockpit, has been developed for the trial. FDS is a valuable tool for use in the event investigation process, and is also a powerful debriefing aid when it is necessary to involve the aircrew to understand the circumstances and causes of a particular event. The self-contained nature and very small size of the viewing program allows it and the relevant portion of flight data to be sent on a floppy disk to aircrew for their review on any PC. This is particularly useful for debriefing crews at remote operating bases. An example FDS display is shown in Figure 4.

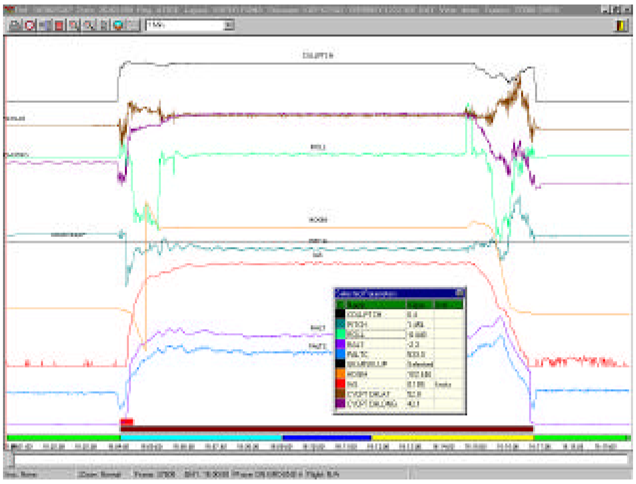

Figure 3: FDT data trace

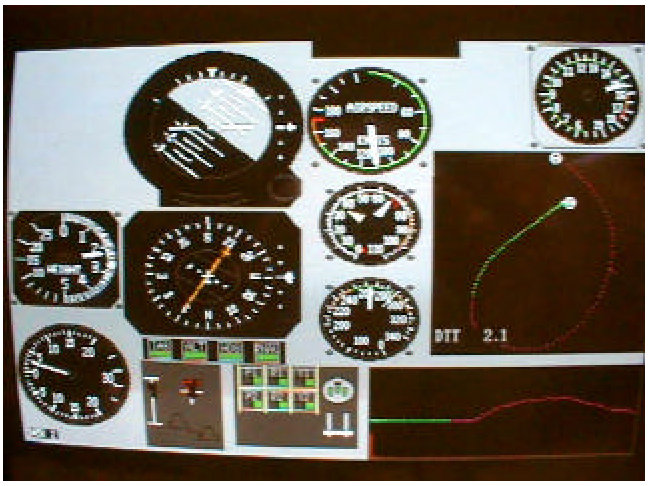

Figure 4: FDS display

Flight Data Exceedances (FDE) performs the secondary analysis of the events and provides graphical displays of event trends. Validated event records are exported to FDE and stored in a database. If the required algorithms are available, FDE can automatically calculate a severity rating for each event. FDE is primarily a management tool. The user can add keywords and notes to events, create links to other modules such as ASR if the event is associated with an Air Safety Report, and track any follow up actions. FDE also generates a number of event trend charts for management reporting. The chart selection page is shown in Figure 5.

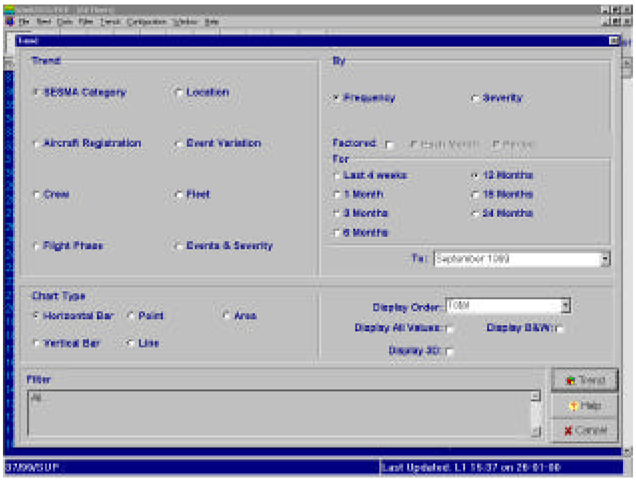

Figure 5: FDE trend chart selection

Flight Data Maximum Values (FDM) performs the secondary analysis of the flight data measurements and provides graphical displays of measurements. FDM facilitates a better understanding of operational variables by showing their distributions and enabling comparisons of distributions across different locations, fleets etc. The FDM module is shown in Figure 6.

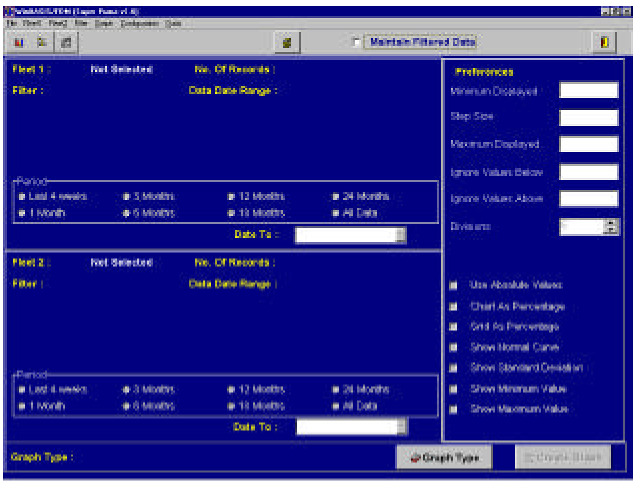

Figure 6: FDM module

Although the FDE and FDM modules are already in use as part of BA's existing OFDM programme, the HOMP trial is the first time these modules have been interfaced to FDT. Further details of the HOMP software are given in reference [5].

Trial organisation and status

The HOMP trial is being carried out on five BHL Tigers. Two helicopters are located at Scatsta on the Shetland Islands and three at Aberdeen. This arrangement enables an assessment of the logistics and management issues associated with the application of a HOMP to helicopters located at a remote base. The CQAR PCMCIA cards are changed once a day by maintenance personnel when the aircraft are returned to the hangar. All the acquired flight data is being retained to enable re-analysis at a future date.

All data analysis takes place on the HOMP analysis system at Aberdeen, and the trial monitoring programme is managed from here on a part-time basis by a senior training captain (Captain Norman) with the assistance of the Flight Safety Department. Captain Norman performs the roles of both the HOMP operator and manager shown in Figure 1. The data from the Aberdeen aircraft is replayed on a daily basis, the data from the Scatsta aircraft is replayed in a batch process when a zip disk is received every few days. The HOMP manager reviews and assesses any events generated and follows these up with aircrew if necessary. He also issues periodic bulletins informing all BHL personnel of what is being learnt from the trial.

A back-up HOMP system is installed at SHL's facility in Southampton to enable SHL to provide development, testing and ad-hoc analysis support to BHL and BA.

Periodic industry briefings are being held for representatives of the North Sea helicopter operators and oil companies to provide information on what is being learnt from the trial and to obtain feedback. Trial progress and technical direction is reviewed on a regular basis by a steering group, comprising representatives from the CAA, Shell Aircraft, SHL, BHL and BA.

Work on the trial commenced in January 1999. By September the CQARs were installed and the downloading of FDR data from the five trials aircraft had commenced. There are on-going developments of the HOMP software but the core replay and analysis modules are fully operational. The trial is currently scheduled to complete at the end of August 2000 however, owing to the positive experience being gained, it is likely that the trial will be extended for a second year.

Configuration of the flight data analysis

The HOMP software performs two types of analysis:

- Event analysis, which provides information on the extremes of the operation.
- Measurement analysis, which provides information on the whole operation.

So far work has concentrated on the event analysis.

A review of helicopter operational accidents and occurrences was carried out using information from the CAA Safety Data Department's UK Occurrence Reporting System. This provided an input to the event specification process to help to ensure that events are targeted at relevant safety issues. A number of accidents and occurrences were linked to the offshore helideck environmental problems such as severe structure induced turbulence, rolling and pitching helidecks, the close proximity of obstacles, and hot gas exhausts from turbines and the flare stacks. Others were linked to problems associated with aircraft handling and pilot disorientation such as a misjudged landing, disorientation on take-off, controlled flight into terrain and water, entry into a 'vortex ring' state, and roll over during taxiing. Further accidents and occurrences were associated with aircraft management problems such as a lack of fuel and landing with the wheels up.

Using both the accident analysis and general experience, a set of event analyses were developed jointly by the CAA, Shell Aircraft, BHL and SHL. Examples of the types of event which have been created are:

- Pitch and roll attitudes and rates for various conditions
- Monitoring of the helicopter rotation on a rig take-off
- High rates of descent, low airspeeds etc.
- Normal, lateral and longitudinal accelerations
- Exceedances of collective pitch, engine torque and rotor speed limits
- Excessive control movements
- Inappropriate engagement of autopilot modes
- Gear selection
- VNO/VNE exceedances
- Low fuel contents
- Heater on during take-off and landing
- Excessive heading change immediately after take-off
- High approach speeds
- Downwind flight
- Flight through hot gas
- Excessive deck movement
- Taxi limits

The event set is subject to on-going development, for example a pilot workload event related to structure-induced turbulence is currently being specified. In addition, the CAA are funding separate work attempting to synthesise low airspeed data from the FDR parameters to enable measurement of low airspeed. If successful, this will enable further events to be created.

Work is also continuing to develop the flight data measurements. These are measurements made on every flight and can be maximum/minimum measurements under different conditions, or measurements at specific points or conditions.

HOMP TRIAL EXPERIENCE - THE OPERATOR'S VIEW

Experience to date

HOMP system

A total of approximately 2200 hours of data has been gathered between September 1999 and the beginning of February 2000. This includes the time the aircraft is powered by flight crew before or after a flight. A very good data recovery rate of approximately 99% has been achieved.

To date, there have been no failures of the CQARs or PCMCIA cards. A number of minor unserviceabilities have been noted in the flight data acquisition equipment. Although this is established equipment and not part of the trial, it has had a slight impact on the usefulness of the data. Picking up these defective parameters is one bonus of the trial as rectification work can then take place, significantly increasing the probability that all FDR parameters will be available if needed for their primary function of accident/incident investigation. In the absence of HOMP, FDR parameters are only required to be checked once per year.

HOMP management

During the early stages of the trial workload was high through having to process the incoming data whilst managing changes to the system (bug fixes, event refinements etc.). As time progresses workload will no doubt ease as less time will need to be spent on the system itself and more time can be spent on analysing data. Workload is to some extent a matter of choice – but the more time spent, the more one is likely to get out of the system.

Obtaining a good response from aircrew and management to the HOMP is an important element of the trial. Aircrew are always aware that they are sitting targets for blame when anything goes wrong and so are naturally defensive. It is therefore important to demonstrate to aircrew that HOMP has some benefits to offer them. If their first encounter with HOMP is to receive a ticking off for some misdemeanour, they will naturally resent it. Therefore an initially low key approach is essential to build trust. So far there have been no problems and BALPA are certainly in favour of a HOMP, provided it is applied sensibly and not used as a management tool to discipline pilots.

At present there is no formal pilot - management agreement other than a statement of intent for the trial. A requirement for one is expected to evolve with increasing management interest in HOMP. For an operator starting their own HOMP on their own initiative, it would certainly be important to have an agreement between management and aircrew as to how the data will be used.

Early lessons learnt

Before embarking on a HOMP the full requirements of the process need to be identified and catered for. For example, in order to draw conclusions from looking at flight data, the analyser must be in possession of all relevant data, including aircraft weight, passenger numbers, weather, crew names etc. This requirement was not fully addressed at the project planning stage, but an interface to BHL’s operations computer system has now been created to quickly extract the relevant information which has proven to be extremely valuable.

With regard to the events and flight data measurements, there is only a limited cross-over between fixed and rotary wing requirements. Some of the current events and flight data measurements may still be too fixed-wing orientated, but input from current commercial rotary-wing flying personnel will help to correct this. It is also important to have a clear idea of what the events and flight data measurements are trying to achieve, and one of the objectives of the trial has been to clarify this. Before establishing an event it is necessary to consider what action will be taken when it is triggered. If the answer is “none”, then perhaps the event should not be included.

Some of the current events are still relatively crude and considerable interpretation is required to see whether they are worth acting upon. The main advantage of helicopters is that they more flexible than their fixed-wing counterparts, but this increases the difficulty of creating algorithms that define good or bad operating practise. It will probably be necessary to gradually increase the complexity of the events to reduce false triggering to a level acceptable for a large scale HOMP. On the other hand, the more parameters that are included in an event definition, the more opportunities there are for it to fail to trigger. The project may have initially focussed too much on trying to catch any possible incidents at the expense of the false triggering rate. However, a low false triggering rate must be a prime objective for a large-scale implementation.

The HOMP has raised a number of issues where there is no clear guidance to crew, and it is necessary to decide whether there is a problem or not, whether guidance should be introduced or whether that would be too intrusive. Of course it is a “good thing” to raise these issues for discussion, but it does increase the workload at the start of a programme. When making changes to an already very safe system (i.e. helicopter transport), one has to be careful not to take an action which ultimately reduces overall safety in some subtle way.

The trial has confirmed that it is a good idea to start off with a small number of aircraft so that events and procedures can be refined before large scale data gathering commences. It is essential to reduce false event occurrence rates to a low level to avoid being overwhelmed. Once this is achieved, there is no major obstacle to moving to a large scale implementation.

HOMP "successes"

The HOMP has already generated some interesting results, these are summarised below.

(1) Take-off with full right pedal

The most interesting result to date relates to an incident involving a take-off with full right pedal, which the HOMP detected. It was the co-pilot’s sector as pilot flying. During lift-off from a platform, the aircraft yawed violently right with associated rolling and pitching as the co-pilot regained control. The helideck crew were concerned and asked over the radio whether everything was OK. The relatively inexperienced co-pilot had not known what had gone wrong and this had caused some loss of confidence.

The HOMP system showed that after the autopilot had been engaged at the end of the rotors running turn-round there was a 2 minute delay, during which there were 2 radio transmissions, implying some distractions. During this time there was a progressive change in apparent aircraft heading to the left caused by gyro-compass precession. When this reached 2° left the autopilot, trying to maintain heading, smoothly started to apply right pedal. Full right pedal was achieved after another 30 seconds. The only cockpit clues were a torque indication of 25% (14% would be normal) and 2° left roll attitude. The co-pilot then started to raise the collective to lift off. At 11.5° collective pitch he started to reduce the tail rotor pedal but almost simultaneously the aircraft yawed 30° right (probably with tyres still in light contact with the deck). A shaky hover was established.

Figure 7 shows the position of the tail-rotor pedals and collective, and the status of the autopilot during the period on deck. It seems almost certain that no-one had their feet on the pedals during the period in question. Although not shown on this graph, the cyclic had also moved slowly and smoothly to the right in an attempt to compensate for the left roll induced by the tail-rotor. The total movement was only about 20% of travel, but it seems likely that no-one was holding the cyclic.

The HOMP system was used to debrief the crew as to the real cause of the problem, and to provide positive feedback to prevent a repeat occurrence. The event was considered to be significant as, in the mid-80s, a Super-Puma was destroyed when the crew lifted off with full right pedal and rolled over on a training flight. The Instructor was demonstrating how the aircraft could be lifted off with feet off the pedals but, unknown to him, the autopilot had applied full right pedal prior to lift-off. In addition, a Super-Puma has rolled over during taxiing because a large amount of right pedal was applied with insufficient right cyclic.

(2) Take-off with cabin heater on

The HOMP has shown that, on average, one aircraft a week is taking off from off-shore with the cabin heater on. No doubt the heater is applied on deck for passenger and crew comfort but, apart from being prohibited in the Flight Manual, when the heater is on during takeoff, the maximum available power is reduced and the power turbine inlet temperature for a given power is increased. In the event of an engine failure, having the heater on could make the difference between ditching and flying away. A general bulletin has been issued drawing this to the attention, and the occurrence rate will be monitored to see if this results in an improvement. If not, a decision may be made to add a heater off check to the pre-takeoff checklist.

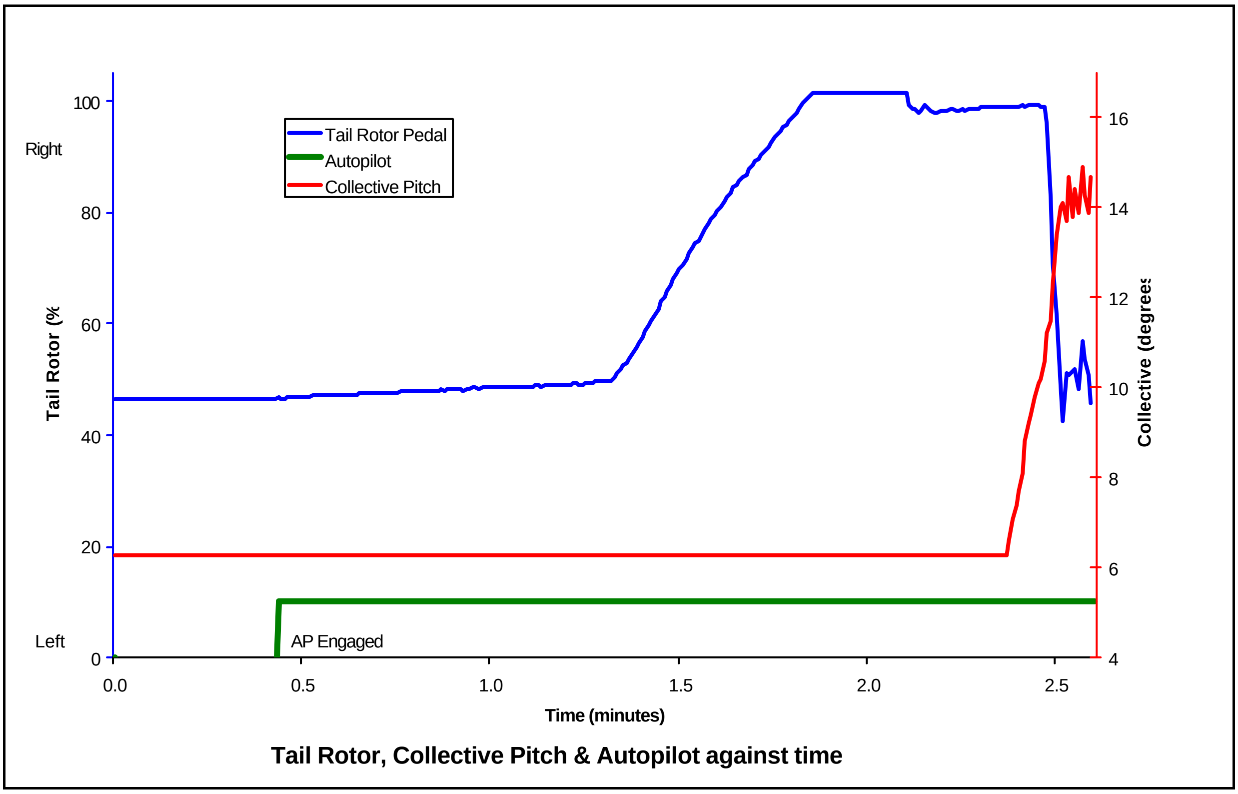

Figure 7: Tail rotor, collective pitch and autopilot vs time

(3) 'Flamboyant' flying

The HOMP has detected a few examples of flamboyant pieces of flying when no passengers are on board the aircraft. Whilst within reason this is good in that it helps pilots increase their flying skills and confidence in the aircraft's capabilities, a fine balance has to be struck to prevent manoeuvres from becoming potentially hazardous. No adverse comments have been made to crew, the HOMP data is currently being used only to assess the extent of the practice. Management have expressed the view that it would be a pity if pilots felt they could not have a bit of fun occasionally when the aircraft was empty, recognising that pilots learn from this experience.

(4) Re-fuelling

On the Tiger both fuel filler caps are on the right hand side of the aircraft. If the aircraft is angled to the left it is possible to take on more fuel. The HOMP detected an example of a pilot using excessive lateral cyclic to list the aircraft to the left, which is not an approved technique. This would obviously reduce the headroom under the rotor on one side of the aircraft and could present a safety hazard on some aircraft types. Excessive cyclic input on the ground can also cause the rotor blades to hit the stops in the rotor head, causing severe damage. Unfortunately wind is a big factor in this and it is impossible to tell from HOMP how close they were to hitting the stops, therefore further investigation is required.

(5) Exceedance of Flight Manual limits

At Scatsta it is necessary to fly procedural IFR approaches at night with a 10 minute separation, if multiple aircraft are returning at the same time an aircraft may be given a delayed approach time. The HOMP has detected examples of aircraft flying with greater than the Flight Manual limit for collective pitch and torque in the cruise to achieve a particular arrival time. This may be an example where there is merit in reviewing local ATC procedures. Collective pitch limits were reduced shortly after the aircraft first came into service in light of the low time before degradation of the gearboxes. Interestingly, there has been a recent increase in the rate of gearbox replacements!

(6) Flight through hot gas

Quite a few 'flight through hot gas' events have been detected with temperature rises of 5 to 12 degC in 3 or 4 seconds. These have mostly occurred on one of the Ninian Platforms where there are known problems with the platform's generating turbine exhaust plumes passing close to or over the helideck. In one case an approach had not been particularly well flown. The aircraft had been descending and slowing with engines at a very low power setting, and the collective was raised just as the aircraft flew into hot gas, demanding rapid acceleration from the engines. The sudden change of ambient air temperature coupled with the acceleration demand could have caused an engine surge, resulting in loss of power. Whilst the Ninian platforms are normally serviced by Scatsta crews, this flight was performed by an Aberdeen crew who were possibly less familiar with the characteristics of the Ninian.

(7) Inadvertent loss of airspeed

During an attempted visual return to an airfield from offshore, the crew decided at a late stage that the weather was too bad and the co-pilot, who was the pilot flying, initiated a climb shortly before crossing the coast in order to make an instrument approach. During the climb, the speed was generally low and at one point reduced to 30 kts. Probably in response to a prompt, the pilot then rapidly lowered the nose, and the climb was continued at 70kts, which is the minimum recommended climbing speed in IMC. Airspeed had been below 70kts for about 1 minute. The captain was probably busy arranging the IFR approach, hence the delay in monitoring and prompting.

Although this incident is still under investigation, it seems likely that the pilot was concerned about the proximity of the terrain and was attempting to climb at best angle of climb speed (45kts) although this is well outside the flight manual IMC flight envelope.

Although a helicopter cannot stall at low airspeed, the drag curve is very steep and control can easily be lost in IMC. The dangers of low airspeed had been highlighted by a previous serious incident on another type. Following a go-around from an offshore installation at night, the crew decided to climb at low airspeed but due to extreme turbulence they failed to notice that airspeed had fallen below zero. They lost control of the aircraft at about 1000 feet, and picked up a 6000 feet/minute descent rate, pulling out of the dive at 75 feet above the sea. The co-pilot of the HOMP flight had joined the company since that incident, and perhaps the training system had failed to make him aware of the lessons learnt.

CONCLUSIONS

Reference [4] states that Operational Flight Data Monitoring (OFDM) is probably the most important safety tool available to aviation. Its benefits on fixed wing operations have been well proven by airlines such as British Airways.

North Sea helicopters have to operate in conditions which are more demanding than those faced by most fixed wing aircraft. There is therefore good reason to try to apply the concept of OFDM to helicopters. However there are also some potential difficulties which could limit its effectiveness on helicopters, such as difficulties in defining operational events for such a manoeuvrable and versatile air vehicle.

A trial Helicopter Operations Monitoring Programme (HOMP) has been implemented on five North Sea helicopters to properly evaluate the benefits of OFDM for helicopters. A substantial amount of flight data has already been gathered and analysed, and the HOMP equipment and software has been proven to be reliable and effective. Although still under way, the trial has already proven the feasibility of a HOMP, and has already been able to demonstrate safety benefits.

Captain Norman writes: "When the project first started, I was quite sceptical. I did not think it would really be possible to get much out of HOMP because helicopter operations are so diverse compared to fixed-wing. However I am now a convert!".

The HOMP trial has really only just started to explore all the possibilities of helicopter OFDM, there is considerable scope for future developments and enhancements to realise the full value of such a programme.

ACKNOWLEDGEMENTS

The authors wish to acknowledge the active part played by the CAA Safety Regulation Group and Shell Aircraft Limited in promoting, funding and guiding the HOMP activity.

REFERENCES

[1] CAA SRG. "Operational Flight Data Monitoring, SRG Guide to Best Practice" (Draft).

[2] Wright D. "Operational Flight Data Monitoring - Why and How". Royal Aeronautical Society Conference, London, March 2000.

[3] Savage J. "Operational Data Monitoring - Experiences Good and Bad". Royal Aeronautical Society Conference, London, March 2000.

[4] Holtom, M. "FOQA: Aviation's Most Important Safety Tool". Joint meeting: 52nd FSF Annual International Air Safety Seminar, 29th IFA International Conference and IATA, November 1999.

[5] Rogan E. "Analysing and Presenting HOMP Data". Royal Aeronautical Society Conference, London, March 2000.

ASSET MANAGEMENT WITH HUMS TECHNOLOGY

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PART I

Presented at the 20th International Symposium on Aircraft Integrated Monitoring Systems(AIMS) 22-25 May 2000 in Garmisch-Parternkirken, Germany.

ABSTRACT **S 2**

This paper describes a comprehensive Helicopter Health and Usage Monitoring System (HUMS) with an integrated unique Rotor Trim and Balance System (ROTABS ^{TM}).

Due to serious safety issues with large helicopters serving the oil industry in the North Sea in the 1970s HUMS was developed and established as a means of dealing with such issues.

Operating in different environments with no similar safety issues on hand some operators were interested in this future technology from the maintenance point of view believing that superior maintenance enhances safety. Having followed the UK programs very closely, BFG took a different approach to the system design philosophy and the issue of Component Life Extension was successfully addressed.

This different approach has led to the award of the US Navy COSSI programme with follow on contracts for the US Marines and US Army reaching a fleet of helicopters of over 2500.

This presentation will explain the different approach and the benefits expected.

List of Abbreviations

CAMS	Computerised Asset Managment System
CAMS	Condition Assessment & Monitoring System
CDU	Cockpit Display Unit
COSSI	Combined Operation and Support Savings Initiative of the US NAVY.
COTS	Commercial Off The Shelf
DAM	Data Acquisition Module
DASP	Data Acquisition & Signal Processor
DFT	Discrete Fourier Transform
DLU	Down Load Unit
DOF	Degrees of Freedom
DRAM	Dynamic RAM
DTU	Data Transfer Unit
DTC	Data Transfer Cartridge
EMI	Electro Mechanical Interference
EVM	Engine Vibration Monitoring
FOI	Frequency of Interest
GBS	Ground Based System
CGBS	Central Ground Based Station
CGBC	Central Ground Based Computer
RGBS	Remote ground Based Station
HUMS	Health and Usage Monitoring System
IAS	Indicated Airspeed
MPU	Main Processor Unit
M/R	Main Rotor
NVM	Non Volatile Memory
OBS	On-Board System
PPU	Primary Processor Unit
RAM	Random Access Memory
RAU	Rotabs Acquisition Unit
RGBS	Remote Ground Based System
ROM	Read Only Memory
ROTABS	Rotor Trim and Balance System
ROTRIM	Rotor Trim S/W
RTB	Rotor Track(Trim) and Balance
SPM	Signal Processing Module

DEFINITION OF HUMS

Health monitoring is a process which provides a means of determining the continued serviceability of components, systems, or structures, without the need for component removal for inspections. This is achieved mainly by monitoring the airframe, transmission, rotor, and engine systems.

Usage monitoring is a process which assesses the life consumption of life-limited components, systems, and structures by monitoring the actual damage exposure (due to combinations of loads, speed, and temperatures, etc.). This includes monitoring of exceedances.

HISTORY OF THE KT-1 HUMS

Operating in a different environment with no similar safety issues on hand, a Swiss Resue operator called REGA was interested in this future technology from the **maintenance point of view** believing that **superior maintenance enhances safety.**

Rega’s primary goals are a 100% accident free mission completion, maintaining a high state of readiness and mission availability, and reducing as much as possible the consumption of aircraft life limited parts. The HUMS plays an active role in accomplishing theses goals.

List of Abbreviations (continued)

SR	Sample Rate
SRAM	Static RAM
STC	Supplemental Type Certificate
T/R	Tail Rotor
TSO	Type Standard Order
VPU	Vibration Processing Unit

In 1992, Rega began a modernization of their Emergency Medical Service (EMS) and Rescue helicopter fleet by purchasing 15 Agusta A109k2's.
One of the systems specified for the new helicopter fleet was a HUMS to improve economical management of the fleet with respect in particular to the following items:

S 3

1. Control of increasing personnel costs
2. Engine cycle life
3. Improving trouble-shooting (Particularly when crews have normally no time to monitor unusual events in flight i.e. short airborne times and/or high work loads)
4. Multilingual operations
5. Deployed aircraft
6. Improving Q.A.

After intensive evaluation of what was on offer at that time Rega contracted with Keystone Helicopters, now Keytech, from West Chester, PA and her partner, Technology Integration Inc. of Boston, (now BFGoodrich) to supply a HUMS system. TII in conjunction with Keytech delivered a prototype system in 1993.

SELECTION CRITERIA **S 4**

Rega based the selection on the:

a. Weight of the proposed system **(<13 Kgs as the weight was critical**)
b. Small Dimensions (1/2 ATR)
c. Integrated Air Data Sensor
d. Modular in design to facilitate expansion and flexibility.
e. No pilot workload increase
f. User friendly
g. Meet with the OEMs' approval.
h. Cost per aircraft system

REQUIRED CAPABILTIES

S 5

a. Exceedance logging and Annunciation
b. Engine trending
c. Aircraft technical configuration Control
d. Electronic aircraft & pilots logs
e. Parts and Assemblies Tracking
f. Multiple Data Imputs
g. Fuel Management
h. Multiple Base locations
i. Fleet fault Analysis
j. Training aid

THE SYSTEM

The KT-1 HUMS consists of two basic systems with a number of subsystems. These are the

1. **On-Board-System** (OBS)
2. **Ground Based System** (GBS).

THE ON-BOARD-SYSTEM (OBS)

S 6

The OBS is installed on the helicopter and consists of

1. Main Processor Unit (MPU)
2. Cockpit Display Unit (CDU)
3. Data Transfer Unit (DTU) and Data Transfer Card (DTC).

S 7

Sensors and interfaces to the OBS are considered part of the OBS. The OBS is designed to provide a complete operational history of each engine, the airframe, and associated trending data. When aircraft power is turned on the system performs an automatic self-test and is ready to record the aircraft start within ten seconds. The system interfaces with the aircraft through standard instrumentation where possible (N_1, N_R, TQ, TOT, etc.) and by added sensors when necessary (fuel flow, airspeed, accelerometers, etc.). Data is gathered 10 Hz and stored in a buffer. If no exceedance is noted or recording requested, data is averaged for a one second time period and sent to the DTU and stored on the DTC. If an exceedance is noted or a record is requested, then raw 10Hz data is sent to the DTU for 10 seconds before an event until 10 seconds after an event. The unit is self contained in that it will record and remember total times and cycles, and exceedances. The unit is designed with an open architecture so that additional systems and parameters can be added as needed.

FUNCTIONS

S 7

The **MPU** performs the following functions automatically, without pilot involvement:

S 8

- Continuous surveillance of the onboard sensors for engines and rotors, air data and an ARINC 429 bus. All Sensors are sampled at a minimum rate of 10 Hz.

- Calculation of engine performance, fuel flow, cycles (engine / airframe / hoist), times and exceedances of prescribed limits on any or all measured parameters.

S 9

- Display of exceedances and cautions on the **CDU**.

- Storage of all raw data and calculated parameters to a removable memory card in the **DTU**.

- Recording of all programmed events and abnormal events.

- Onboard storage of selected aircraft, performance and exceedance data in a nonvolatile memory.

SOME TYPICAL PARAMETERS

Airframe	Transmission Torque Main Rotor RPM Main Gearbox Oil Temp. Main Gearbox Oil Press. Main & Tail Chip Detectors Rotorbrake Hook & Hoist Squat Switch Hard Landing
Engines	Torque NG NP TOT Fuel Flow Oil Press & Temp Bleed Valves Hydraulic Pressures Stater/Generator performance
General	Air Data (from internal Air data sensor) Navigation Data

SIGNAL TYPES

- Digital
- Analog
- Pulse
- ARINC
- Computed

S 10 **S 11**

The **CDU** provides an interface between the HUMS and the pilot. It provides a means of showing exceedances which may have occurred, allows the pilot to select information for monitoring, and allows the pilot to record events or conditions he may want to examine at a later time. The CDU can provide a digital readout of most of the engine and airframe parameters as well as navigation, flight and wind data. In the A109k2 it also provides some data not found from other instruments such as fuel flow, and weight and balance. The pilot may also interact through the CDU to perform Power Assurance Checks and review exceedance data.

S 12

The **DTU** holds the DTC which contains all of the recorded data packets that are logged every second of operation. When the data transfer cartridge is reviewed a complete history of the flight is available to include a real time replay in seconds. The DTC is a standard PCMCIA memory card. The card is capable of recording a number of flights, depending on the number of exceedances or records taken during the flight. The DTC is placed in the DTU at the beginning of the day or beginning of the pilot's work shift. The card is expected to be downloaded from the aircraft at the end of each day, when pilots change or when it is nearly full. A DTC is not required in order for the MPU to function.

THE GROUND BASED SYSTEM

The GBS consists of :

1. Central Ground Base Station (CGBS) located at a main base
2. Remote Ground Base Stations (RGBS) at other bases in the operation

S 13 **S 14**

The GBS consists of a Sun Sparc Workstation, with an internal hard drive, an external hard drive, a data card reader and laser printer. In the Maintenance Center the central computer is networked to other computers and printers at the operational remote sites via ISDN connection. The system is multi-user, and multitasking. The relational data base is capable of automatic generation of standard and special trending reports, maintenance tracking reports, and management reports.

The GBS retrieves the data from the aircraft either directly by means of a PCMCIA-type memory card which is removed from the DTU in the cockpit by a member of the crew and placed in the DLU or, in the case of the Central GBS, by automatically polling the remote bases during the night and transferring the data by modem.

FUNCTIONS

S 15

The GBS performs the following functions:

- Preflight upload of significant data to the DTC.
- Download of flight data from the DTC.
- Automated entry of data into a data base for each aircraft to include:
  - a. Times and Landings
  - b. Engine Cycle Counting
  - c. Exceedance history
- Flight and Mission logs **S 16**
- Maintenance due and history.
- Interface with parts tracking and inventory system
- Engineering review of flights (individual or groups of flights)

ACCESS & DOWNLOADING

S 17

The user friendly GBS can be operated by personnel not trained in digital signal processing, fault modeling, data interpretation, or scientific programming. The GBS has three levels of password protection to insure data is not inadvertently disturbed. The three categories of data are:

- Flight data-No password required
- Technical data for maintenance (e.g. trending data)- Only Maintenance department.
- Exceedances (Data stored without pilots name)- Chief of Maintenance or his deputy.

REPORTS

S 18

The following reports are generated:

- Histories : Aircraft / Parts /Pilot /flight
- Aircraft : Status /Cofiguration / fleet
- Exceedances : Aircraft/Parts/Pilot/ flight
- Maintenance : Due / Performed / Bulletins
- Flight : Replay / Exceedances
- Warranty : Status / Remaining
- Analysis : Trending plots / Histograms/ MTBF
- Log Summaries: Aircraft / Engine

The GBS provides the Event Reports (start, takeoff, landing, cycles, etc.) and the Trending Data (power assurance, temperature, fuel flow, torque and speeds). The one second performance data samples for temperature, fuel flow, torque, speeds and the power assurance checks (when performed) is stored on the DTC. Before being downloaded to the GBS this data is processed and corrected to standard day for trending. Trending procedures include the comparison of the normalized data with baseline performance data supplied by the manufacturer and the calculation of deviations from the normal. The trending data is reported in tabular form and includes the date and time of the event, the aircraft and engine total time and cycles when the event occurred, the type of event, its duration, value and level. The GBS has provisions for the addition of Performance Modeling and Diagnostics.

S 19 / S 20

Analysis is displayed by means of a Strip Cahrt that is easy to operate and read.

S 21

An example of the REGA HISTOGRAM illustrates Usage differences due to mission profile.

THE REMOTE GROUND BASED SYSTEM

S 22

The Remote Ground Based Stations (RGBS) are optional and are comprised of:

- Remote Base Computer (RGBC)
- Down Load Unit (DLU)
- ISDN connection
- Printer

S 23

The RGBS perform the following functions:

1. Pre-flight upload of significant data to the DTC
2. Automated download of flight data from the DTC and entry of data into a PC for transfer by ISDN to the main maintenance Computer, i.e. (CGBS) which polls automatically the data during the night

S 24

ACCESS & DOWNLOADING

The access and downloading is the same as for the main (CGBS)computer.

S 25

FLIGHT TESTING OF THE HUMS

This system was flight tested extensively for approximately a year and, although quite good, was found to have problems achieving TSO standards. Another prototype was flight tested in 1995.

The HUMS is installed on the A109k2 on a non interference basis. This means that the pilot does not have to make flight decisions based on the HUMS nor should the HUMS interfere with or distract the pilot from flying the aircraft. However if the data is available, and the system is shown to be reliable, the pilot can be expected to use HUMS data in making decisions. Flight test insured that the data being presented to the pilot is accurate and timely, is non obtrusive, and is reliable. A HUMS by its nature must be reliable if it is to be believed by the people who work with it, or the manufacturers who base their decisions on it. It has to behave almost like a human when it comes to notifying the pilot. Nothing is more irritating than to find the HUMS had triggered the Caution Light for a .1% exceedance of the rotor RPM. Flight manuals and maintenance manuals are written with the pilot or mechanic in mind, not a computer picking up data at 10 Hz. Some things in flight manuals are contradictory. For instance, the Main Rotor Normal Operating Speeds could be 97 - 100%, and the Main Rotor Limit could be 100%, the pilot can fly at 100% and would not be exceeding a limit. The HUMS would give continuous exceedances. If the pilot runs at 100% but encounters gusty air, and the rotor speed governor is a little slow to compensate, momentary excursions over 100% are possible. A pilot might think nothing of this, but the HUMS would faithfully report it. To make the system serviceable and not an annoyance, the engineers and pilots must make adjustments to the HUMS so that it almost thinks like a human pilot. This involves normally close coordination with the manufacturer to set appropriate limits which they will accept. Sometimes this means that a small time delay or a specific parameter is relaxed slightly to achieve the same results but without nuisance reporting

At this point in time the flight manual and maintenance manual are the prima facia documents approved by the certifying authorities for flying or maintaining the aircraft. The HUMS can only mimic these documents. The OEM's need to provide input into how the HUMS is implemented.

During the flight testing at Rega, there were many such issues. For example, The engine manufacturer used 85% N_1 as a level to determine a partial cycle count of the engine. It was possible during flight to set the collective, and using only the cyclic to load or unload the rotor such that N_1 varied from 83 to 87%. During that flight 3.5 cycles were achieved when 1 or less would have been normal. This was, of course, excessive. With permission of the manufacturer, the HUMS was programmed with a dead band. If N_1 went above 85%, it must drop to a lower value before a partial cycle was counted.

Another big issue with the HUMS was electromagnetic interference. When the HUMS is normally benched check in the laboratory, the environment is clean and the power supply is shielded and steady. The helicopter environment is quite different, with a good amount of EMI being generated, and because of the weight, shielding is at a minimum.

During flight testing of an autopilot modification, the HUMS has already proved itself to be an invaluable tool. The autopilot experienced a collective up runaway. Since the aircraft was in a hover, the pilot's attention was outside of the cockpit. The first indication of trouble was the Overtorque Warning. The pilot grabbed and lowered the collective, while at the same time checking the torque indicator. He noted the torque value under 110%. In the A109k2, the pilot is allowed a momentary excursion up to 110%, but not pass 110%. The pilot thought that he had remained within limits, even though the HUMS did show an exceedance. Since the autopilot had malfunctioned, the flight was terminated and the aircraft returned to base.

During the review of the HUMS data, maintenance found the torque value had went to 123% during the runaway. After all of the data was reviewed, The helicopter manufacturer was consulted and they agreed to an additional 10 hours of flight time while monitoring the aircraft for signs of deterioration. An exceedance of this type would normally have resulted in a costly and time consuming tear down inspection. The vast amount of data recorded during the event was instrumental in the decision.

CERTIFICATION

S 26

BFGoodrich/TII TSO'd and certified the KT-1 system with the FAA in 1996. Throughout this certification, Rega had also been refining her requirements for the HUMS, normally requesting that additional parameters and systems be added to the HUMS.

BENEFITS OF THE HUMS

S 27

Rega has received immediate benefit from the use of the HUMS, particularly in the areas of:

- troubleshooting
- extended engine life through better cycle counting
- operational improvements, pilot's awareness of the aircraft condition and the effect of certain maneuvers on the aircraft
- engineering and maintenance awareness of the status of the aircraft and it's components
- better communications with the airframe and component manufacturers
- a small but encouraging reduction in Hull insurance premiums.
- Training aid

When Rega has approached the helicopter manufacturer or the engine manufacturer with a problem or potentially expensive inspection or replacement following some exceedance such as on overtorque, the manufacturers have been very helpful because so much data could be presented about the event. A thorough and safe analysis could be made which was beneficial to both parties. This has led to a better feeling of safety for both the pilots and maintenance staff. It has also led to more awareness of the limits of the aircraft. By this means a number of unnecessary component (i.e. Main Gear Boxes) changes have ben avoided. Also a good life improvement was achieved for the generators through monitoring of spiking and changing the start up proceedures.

CONCLUDING REMARKS

S 28

The KT-1 HUMS is now fully deployed at Rega and recognised as a welcome and beneficial addition to the Rega fleet.

With the system being updated to their requests there are a few more benefits they expect to see from use of the HUMS such as:

- increase in maintenance efficiency
- more efficient use of personnel
- reduction in replacement of life limited parts
- better planning through trending
- reduction in paperwork
- more trust in the airworthiness of the aircraft.
- increased reliabilty & availability through reduced forced ground time.
- Maintenance Credits

The KT-1 HUMS approach from the asset management point of view as opposed to the approach from the safety point of view has proven to be correct for the long term.

Used in this manner the HUMS can be easily considered a Computerised Asset Management System (CAMS).

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Hand over to part II

ASSET MANAGEMENT WITH HUMS TECHNOLOGY

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PART II

Presented at the 20th International Symposium on Aircraft Integrated Monitoring Systems(AIMS) 22-25 May 2000 in Garmisch-Parternkirken, Germany.

Abstract:
This Part II provides a review and update on a comprehensive program aimed at using integrated mechanical diagnostics and comprehensive usage analysis to effect safe changes in maintenance and operating procedures on several types of military and civil helicopters (Reference 1). The US Navy has embraced Integrated Mechanical Diagnostics (IMD)–HUMS technology as a component of its plan to reduce O&S (Operating and Support) costs, increase readiness within an environment of diminishing resources, and enhance flight safety. A team led by BFGoodrich Aerospace and the program offices of the SH-60 Multi-mission Helicopters (PMA-299) and the CH-53E Executive Transport and Heavy Lift helicopters (PMA-261) was selected in a 1997 competition run by DARPA to implement the BFGoodrich IMD-HUMS system on the H-60 and CH-53E under the Commercial Operations and Support Savings Initiative (COSSI). The scope of the program includes outfitting each aircraft model with a production-ready on-board system, deploying ground stations within the fleet O&S environment, and linking the entire information system to various Government functional organizations, the aircraft OEM, and the IMD-HUMS supplier. The system performs automated monitoring, status evaluation, diagnostics, and reporting on the aircraft. Some of the features of the in-flight system include continuous monitoring of engines, flight regimes, and critical mechanical systems; automated rotor track and balance calculations. The ground system performs usage and structural life consumption calculations, maintenance required forecasting and reporting, and links to the logistics system. Plans are to equip all 400 Navy H-60's and all of the approximately 200 CH-53E's following a 2-squadron fleet trial. Projected potential O&S savings attributable to the IMD system and supporting changes in procedures on these 600 aircraft are substantial. In August of 1999, the US Army joined the IMD

List of Abbreviations

CAMS	Computerised Asset Managment System
CAMS	Condition Assessment & Monitoring System
CDU	Cockpit Display Unit
COSSI	Combined Operation and Support Savings Initiative of the US NAVY.
COTS	Commercial Off The Shelf
DAM	Data Acquisition Module
DARPA	Defense Advanced Research Programs Agency
DASP	Data Acquisition & Signal Processor
DFT	Discrete Fourier Transform
DLU	Down Load Unit
DOF	Degrees of Freedom
DRAM	Dynamic RAM
DTU	Data Transfer Unit
DTC	Data Transfer Cartridge
EMI	Electro Mechanical Interference
EVM	Engine Vibration Monitoring
FCF	Funtional Check Flight
FOI	Frequency of Interest
GBS	Ground Based System
CGBS	Central Ground Based Station
CGBC	Central Ground Based Computer
RGBS	Remote ground Based Station
HUMS	Health and Usage Monitoring System
IAS	Indicated Airspeed
IMD	Integrated Mechanical Diagnostics
JDUPO	Joint Dual-Use Program Office
MPU	Main Processor Unit
M/R	Main Rotor
NALCOMIS	Naval Aviation Logistics Command Management Information System
NVM	Non Volatile Memory
O&S	Operation & Support
OBS	On-Board System
PPU	Primary Processor Unit

Presented at the 20th International Symposium on Aircraft Integrated Monitoring Systems(AIMS) 22-25 May 2000 in Garmisch-Parternkirken, Germany.

program targeting the UH-60 Blackhawk fleet to receive the same system as the naval H-60 fleet. The Army program has objectives that are similar to those stated above, although Blackhawk-unique maneuvers and maintenance procedures will be incorporated in the system. In September 1999, the US Marine Corps and BFGoodrich entered an agreement to incorporate the AH-1Z 4-bladed Cobra and UH-1Y 4-bladed Huey in the IMD program.

Introduction

The United States Navy , United States Marine Corps, and US Army in partnership with BFGoodrich Aerospace, have embarked upon an ambitious program to improve operational readiness and flight safety while greatly reducing maintenance-related costs. The Naval H-60 and USMC CH-53E HUMS program is now on the cusp of an important milestone – installation of the first fully functional production-ready system on two aircraft types, the SH-60B multi-mission helicopter and the CH-53E heavy lift helicopter. The system has evolved under Joint Dual-Use Program Office's (JDUPO) Commercial Operations and Support Savings Initiative (COSSI) and is now referred to as IMD-HUMS. This paper presents the system's current state of evolution and outlines how the system will continue to evolve as we progress toward fleet-wide deployment. This paper will describe the components and processes that make up the fully functional and integrated system. It will also outline the near term implementation plan to prepare for eventual transition from initial installation to fleet-wide deployment.

List of Abbreviations (continued)

RAM	Random Access Memory
RAU	Rotabs Acquisition Unit
RDC	Remote Data Concentrator
RGBS	Remote Ground Based Station
ROM	Read Only Memory
ROTABS	Rotor Trim and Balance System
ROTRIM	Rotor Trim S/W
RTB	Rotor Track(Trim) and Balance
SPM	Signal Processing Module
SR	Sample Rate
SRAM	Static RAM
STC	Supplemental Type Certificate
T/R	Tail Rotor
TSO	Type Standard Order
USMC	United States Marine Corps
VATS	Vibration & Analysis Test System
VPU	Vibration Processing Unit

System Description

The IMD HUMS consists of an airborne system and a ground based system, both implemented with the maximum practical open systems architecture. The architecture enables efficient data and information exchange between the two systems and among other ground base stations. Figure 1 summarizes the functional capabilities of the system. Note that the system is modular and scaleable; ie, the functionality summarized in Figure 1 can be implemented fully or partially, and the system can be scaled from the smallest single engine helicopter to the largest 3-engine helicopters such as the CH-53E. The airborne system acquires and processes data related to four specific areas; current performance and limit exceedances, mechanical diagnostics, rotor track and balance, and service life utilization. The data used by the system is obtained primarily from the installed state sensors that are part of the basic aircraft (including busses), and specially mounted accelerometers and shaft position sensors.

Figure1. Dual Use IMD-HUMS System Functionality

Information of immediate benefit to the flight crew is automatically (and can be selectively) displayed in the cockpit. However, the majority of the raw data and processed information is exported to the ground based station after landing. The ground based station is used to conduct preliminary analyses to aid local maintainers. In the Navy and USMC implementation, the data and processed information is then forwarded to a networked computer for trending, prognostics and subsequent planning. The software is designed with flexible, configurable, published interfaces to allow other functionality to be readily integrated. The NT-based operating system uses an ODBC-compatible Oracle™ database. These features enable the effective transition of several Navy diagnostics and maintenance programs to the current highly integrated and flexible system. The Army program will utilize a stand-alone ground station with capabilities for parts tracking and maintenance management as well as integration with a web-based readiness reporting system

Components

Airborne System. The airborne system consists of the original manufacture helicopter fitted with additional hardware and instrumentation. The hardware includes a Main Processing Unit, an optical tracker, weight-reducing Remote Data Concentrators (one for the SH-60 and two for the CH-53), a Cockpit Display Unit (CDU) and a Data Transfer Unit (DTU). Additionally, at least 30 and up to 70 sensors are mounted on the engine and drive train/rotor components to provide condition indicators. An independent tachometer and 1/rev indexer (at the tail rotor) are added to complete the sensor suite. The system can accept entirely analog or digital (1553 bus) signal input or both simultaneously.

The Main Processor Unit (MPU) is composed of a Primary Processor Unit (PPU) and a Vibration Processor Unit (VPU). See Figure 2. The PPU serves as the system controller by managing information both in and out. It receives information from the Remote Data Concentrator (RDC) and the VPU. The RDC provides information derived from the state sensors and processors that make up the original manufacturer's equipment suite. The VPU provides selected raw data signals as well as processed signals. During flight, the MPU acquires data at 10 hertz, and stores most data at 1 hz, unless there is an exceedance or other noteworthy event. In the case of an exceedance, the MPU will acquire and record raw signals from the VPU for the exceedance duration plus and minus 15 seconds. The CDU provides an interface that allows the operator to view this data in real-time and provides password protected maintenance information. Based on the information requested, the MPU sends information to the CDU or both raw and processed flight data to the Ground Base System via the DTU. The DTU uses a PCMCIA flash memory card as a medium for temporary flight data storage. It is easily removable from the aircraft for data transfer to the ground station. The card serves a dual function as it is also used to upload the data needed to configure the on-board system as well as algorithms and other pertinent information.

Figure 2. General Configuration of the Airborne System Primary

Components

As mentioned above, the IMD HUMS airborne system functions include data acquisition and processing and aircrew advisories for selected events. It is also designed to automate several processes. It can be used to automate rotor track and balance Functional Check Flight (FCF) procedures. Likewise, it automates several engine checks. Maintainers can use the system to conduct flight line troubleshooting during diagnostic checks. FCF crews use the system to determine their maintenance effectiveness and also for basic information purposes. The algorithms and data used by the OBS to perform these and other functions are defined in a configuration utility resident on the ground station.

Ground Based System.

The Ground Based System is made up of a series of networked ground stations which configure flight-specific analysis to support either pilot or maintainer queries. The system provides access to a larger data set for trending, prognostics and planning. The ground stations are the primary user interface with the IMD system. The system is responsible for automatically logging and maintaining all flight and maintenance data, performing aircraft configuration and parts tracking, supporting maintenance and engineering analysis of the flight data, generating engineering and management reports, and archiving data.

Figure 3.
Ground Station Software Integrated With NALCOMIS

The Navy/USMC IMD HUMS Ground Based System is integrated with the Naval Aviation Logistics Command Management Information System (NALCOMIS) ***providing a complete asset management solution***. NALCOMIS is the Navy's squadron-level version of a standard aviation maintenance management information system. It is currently being upgraded to the newer version, known as Optimized NALCOMIS OMA, for use in the IMD project. It includes functions for maintenance management and record keeping, configuration and parts life tracking, flight record keeping and quality assurance. The IMD system is intended to reduce operation and support costs by providing timely and accurate information to aircraft fleet operators, maintainers, and flight personnel regarding the maintenance and serviceability of their aircraft. It automates maintenance activities scheduling and facilitates maintenance actions recording. Users can generate maintenance forecast and maintenance history reports for any collection of aircraft or assemblies, providing for timely and opportunistic scheduling of maintenance activities. The Portable Ground Station is a version that is to be used on deployment and at the flight line. It includes a sub-set of the Ground Station functions.

IMD Functions

The IMD HUMS offers a ***comprehensive*** service suite, providing for Health Monitoring, Usage Monitoring and a Maintainer Interface. Health Monitoring includes Rotor Track and Balance, with both continuous and prompted monitoring, Engine Performance Assessment with prompted checks and condition trending, and, in addition, it includes mechanical diagnostics of all drive train components, bearings and gears. Planned upgrades will include rotor assembly diagnostics. Certain health monitoring functions can be accessed via the Flight Data Recorder Interface. Usage monitoring checks incoming data against preset thresholds and alerts the aircrew if exceedances are observed. This service includes Operational Usage (time tracking and cycle counting) and Structural Usage Monitoring (regime recognition, component usage and usage application). The three primary diagnostic functions (Mechanical Diagnostics, Rotor Track and Balance and Structural Usage) are presented below.

Usage Monitoring.

One primary IMD dual use program objective is to introduce and institutionalize a family of automated structural usage data acquisition and processing algorithms. Given this capability, parts life determination is individualized and now based upon the actual helicopter usage. The usage monitoring subsystem determines the percentage of flight time the helicopter has spent in each flight mode (regime) as well as the specific regime sequence. The regime data is then used to calculate the rate that various structural components are being used up and when they need to be removed from service to maintain the required reliability rate.

A regime is the basic building block of an aircraft usage monitoring system. Some examples of regimes are takeoff, hovering, level flight, various turns and landing. Time histories of flight parameters are analyzed to determine the instantaneous phase of flight. Normal acceleration (Nz), power and yaw rates are parameters that define subsets of regimes that can exist within the confines of a basic regime. The time spent within each regime, during a given flight is measured and tabulated as part of a usage spectrum. It is unlikely that an aircraft will be flown into every regime on a single flight. However, over a period of time, the aircraft can be expected to fly into every basic regime, depending upon the distribution of missions. The continuing summation of multi-flight experience defines the mission-specific and composite usage spectra for the aircraft and its components.

Regime recognition is performed to map recorded aircraft parameter data to a set of ground and flight regimes. The process output includes several summary reports as well as calculated adjustments to the useful life of specific components. The first report called the regime sequence report (i.e., flight profile) represents the time history of the aircraft operation, listing the sequence of regimes encountered. The flight spectrum report summarizes the distribution of time spent in each regime and how often the regime is repeated. Computed component usage is then aggregated to the sum of the usage already carried by the system for that specific component.

In addition to providing an accurate determination of parts usage, the algorithms introduce improved data collection accuracy via automation. Usage data are continuously collected during each flight of each aircraft - a process that produces a comprehensive amount of "raw" usage information. Automated analysis processes convert this data into summaries that are then archived and automatically distributed to enhance the logistics decision-making process. This automated data collection enables individualized parts life determination, addressing the actual usage of each aircraft in the fleet. Additionally, all fleet aircraft within the model type are now treated with the same effective margins of safety by the improved system of algorithms. This approach retains the high confidence levels (6-9's, or "one-in-a-million" probability of catastrophic failure) historically embodied in the original safety regulations while it provides an objective means for eliminating inappropriate and unwanted parts life penalties

Note: The "equivalent safety" imperative described above dictates a need for affordable human oversight using automated and semi-automated procedures. IMD HUMS provides this oversight capability. The oversight required will diminish as confidence in the system improves, but it will always be present. The system objective is a process that allows engineering management the opportunity to randomly inspect the data as a quality assurance function or to inspect on exception.

Rotor Track and Balance.

The physics behind rotor-induced vibration for both main and tail rotors is well understood. All helicopters exhibit varying degrees of low-frequency vibration generated by the main and tail rotors at multiples of the rotor rotation frequency. These low frequency vibrations can be very unpleasant to the helicopter occupants (whose modal frequencies are the same) and are the driving forces behind rotor track and balance initiatives. One type of vibration is a function of the blade passage rates of the main and/or tail rotors. These vibrations can be minimized through thoughtful design. The other type of vibration is caused solely by small differences among the (nominally similar) blades themselves. Manufacturers allow for three types of rotor/blade adjustments to reduce the vibration; hub-weight pockets/brackets, adjustable pitch-control rods and one (or more) adjustable tabs mounted on the blades' trailing edges.

Two basic approaches are used to minimize vibrations; minimizing blade track deviation and minimizing directly measured vibration. The blade track deviation approach seeks to minimize deviations at one point in the blade azimuth. The concept is that if the deviations are small, resulting vibration will also be low. A more direct approach is to measure and minimize the actual vibration. ROTABSTM is the IMD COSSI rotor balancing system that uses vibration data obtained from fuselage mounted sensors for both balancing and tracking. This technique obviates the need for hand-held or fuselage-mounted optical tracking devices. It is particularly well suited for full time operation and tactical military situations. The IMD COSSI rotor track and balance software recommends adjustments to some or all of the three previously-mentioned alternatives (weight, control rod, tabs) to effect an efficient solution. It includes a rotor-balancing algorithm that uses vibration and track data when available. However, the algorithm also functions properly with vibration data only, for example, when a tracker is either not installed or is unable to operate. The balancing system has been validated with a series of acceptance trials at Patuxent River Naval Air Station. Flight testing in the fall of 1998 and early this year demonstrated that the ROTABSTM algorithms are very robust and capable. The technique succeeded in bringing out-of-balance blades into balance on the first trial each time for ten trials on two different aircraft types. The two aircraft tested were the 4-bladed SH-60B support helicopter and the heavyweight, 7-bladed CH-53E Cargo Helicopter. The tests were conducted to confirm that the ROTABSTM algorithms could derive track and balance solutions equal to or better than those of the NAVAIR 01-1A-24 procedures. In each case, the algorithms recommended changes that brought the blades into acceptable vibration levels and often offered changes that would reduce the vibrations to an extremely low level. Typical results after a single adjustment sequence are shown in Figure 4. This type of performance is intended to offer more options to maintenance flight commanders. During tactical situations, the system can be configured to provide the minimum number of adjustment changes needed to bring vibrations to an acceptable level. During routine operations, a more comprehensive set of changes might be invoked to minimize vibration. For example, fine-tuning the rotor's performance might reduce the need for adjustments in a subsequent tactical situation. Planned improvements include vibration- and tracking-based diagnostics for rotor head faults, such as faulty lead-lag dampers, worn pitch control rods or vibration dampers.

Figure 4 . One-Pass Optimization of 6 Degrees-of-Freedom Balance (H-60 vertical and roll data shown)

Mechanical Diagnostics.

The diagnostics function provides both comprehensive integrated component- by -component mechanical diagnostics (IMD), as well as traditional NAVAIR 01-1A-24 procedures for the CH-53E, and A1-H60CA-VIB-000 procedures for the SH-60 models. The IMD diagnostics focus on individual gears, bearings and shafts. The function includes advanced diagnostics software that is both modular and upgradeable. The IMD COSSI system was designed to provide mechanical diagnostics capabilities far in excess of those offered by the Navy's NAVAIR 01-1A-24 Vibration and Analysis Test System (VATS) or equivalents, while still providing equivalent functionality in the aforementioned areas. Figure 5 shows the extent of the components covered by the IMD COSSI system for the SH-60 in comparison to those covered by the VATS. IMD COSSI data acquisition is fully automated and occurs without aircrew intervention, unless specifically requested. The system will autonomously provide Flight Safety Advisories in the event that signals associated with critical components exceed preset thresholds. The system software is designed for and tested to DO178 Level B certification in accordance with the United States Federal Aviation Administration procedures.

IMD COSSI's expert diagnostics utilize advanced signal processing and decision logic and to determine component condition. The condition assessments are based on the signals and indicators obtained from a several complementary algorithms using a variety of confirming sources, since relying on a single sensor to indicate a component's condition has been known to give spurious indications.. To preclude this, the results obtained by analyzing signals obtained from multiple sensors are combined to provide a single condition indicator that is both unambiguous and reliable.

In normal operation, the VPU acquires data from a selection of sensors and tachometers, as commanded by the PPU. The PPU has a master configurable data acquisition schedule commanding the VPU to acquire data when (and only when) data capture windows (flight regimes) are correct. When data quality has been confirmed, the tachometer channels are first processed to provide drive train speed information. Each data channel is calibrated and gained as designated suite of sensors, and all channels (regardless of type) are reviewed for data quality. The data quality assurance routines provide a means to reject data in the event of a required, and then a series of shaft, gear, and bearing diagnostics are applied to components associated with that particular sensor. The outputs of these calculations are diagnostics indicators, which the VPU then sends to the PPU for evaluation and combination. Diagnostics indicators from like components and different sensors are then combined using a variety of proprietary evaluation methodologies to arrive at a health condition for that particular component. Each component health condition is constantly evaluated during flight to assure vehicle safety and provide up-to-date guidance for maintenance planning.

All component diagnostic indicators, condition data, and selected raw data channels are transferred via the data transfer unit to the ground based station for additional analysis, reports, manipulation, and archiving. The ground-base station can support helicopter maintainers and technicians with diagnostic troubleshooting guidance and on-line repair procedures. Similarly, the system supports engineers and analysts, enabling data review and diagnostic algorithm evolution to address new or optimized diagnostics procedures. In this manner, the system provides useful information both immediately and practically, while enabling the analyst to review data and mature the system.

The origins of this effort are documented in "SH-60 Helicopter Integrated Diagnostic System (HIDS) Program Experience and Results of Seeded Fault Testing" (Hess, Hardman and Neubert, 1998). The product of those, and related efforts, have produced the verified and validated processes, procedures and algorithms which comprise the IMD HUMS.

Deployment Schedule and System Characterization

Two aircraft (one aircraft of each type) are now fitted with the IMD HUMS system. The CH-53E has entered formal Development Test after successfully completing engineering tests. Five additional aircraft of each type will be fitted in late-1999. These aircraft will be deployed in operational squadrons and serve as data sources for accelerated system characterization and service suitability evaluations. Navy, Marine and BFGoodrich engineers and maintenance specialists will jointly analyze data obtained from operational service. It will be used to hone the system sensitivities and allow the customer to confidently set cautionary thresholds and exceedance levels. The data will subsequently be used to determine the effectiveness of developmental algorithms by comparing their performance to the results obtained by using current techniques.

Together with Optimized NALCOMIS OMA, IMD HUMS enables the Navy and Marine Corps to start the transition to true condition-based aviation maintenance. This new capability to capture actual usage and condition, coupled with total visibility into the current component configuration for each aircraft, makes possible the process re-engineering that leads to extensive operations and support cost savings.

The Blackhawk adaptation of this system will enter testing in late 2000 on the UH-60A and early 2001 on the HH-60L. The AH-1W and UH-1L systems will enter testing during the early phases of the upgraded aircraft EMD testing at PAX River.

Conclusion

The IMD HUMS has become reality under an aggressive schedule due to the close cooperation between user, aircraft OEM, and system provider. Through this effort a system has been produced which will fulfill the promise of creating ***a complete asset management solution which in turn will*** improve operational readiness and flight safety with reduced maintenance-related costs.

References

1. Hayden, R.E,, and Muldoon, R.A. "US Navy/ US Marine Corps/BFGoodrich IMD COSSI Program: Status", American Helicopter Society 55th Annual Forum, Montreal, Quebec, CANADA, May 1999.

2. Hardman, W., Hess, A., and Neubert, C. “SH-60 Helicopter Integrated Diagnostic System (HIDS) Program Experience and Results of Seeded Fault Testing”, American Helicopter Society 54th Annual Forum, Washington, DC, May 20-22, 1998.

Integrated Ground-Based Logistic / Engineering Support System for Military Aircraft's

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Abstract:

The new generation of military helicopters TIGER and NH 90 will operate in complex scenarios which are determined by multirole mission capability, fast reaction and short turn around times.
These new operational aspects are requesting new maintenance and logistic concepts based on integrated support systems providing good access to the helicopters on board systems and to the infrastructure on ground as well.

This document describes a new logistic and engineering support system which is currently under development at DORNIER and will be applied for the German UH TIGER helicopter.
This integrated logistic and engineering support system reflects the immediate battlefield requirements as well as the new command and control structures and therefore is providing comprehensive and flexible technical and logistical support.

As well as operating from main operating bases and under austere conditions the logistic / engineering support system has to support the operational readiness of military aircraft's. The technical resources and the corresponding service processes are the key elements for evaluating, maintaining and restoring of the operational readiness.

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1. Introduction

The current and future german military aviation scenario can be characterised by the following situation:

- Defence budgets are continuously decreasing
- Frequent and short period operations with high demands on flexibility and mobility (rapid reaction forces)

But the rapidly and frequently changing scenarios require also flexible and state of the art IT-solutions for logistic support. However the present IT-solutions are rather impeding than improving the logistic support process due to the following major shortages:

- They are outdated and obsolete
- They cover only a few parts of the necessary aspects
- They are not comprehensive but expensive
- They are difficult to be maintained

These IT-system deficiencies are strongly impacting the efficiency of military aircraft operations, mainly in terms of economical aspects. It becomes more and more evident that a replacement of the current IT-systems is overdue and highly recommended.

From this background it is in fact a challenge to reach the goal, namely to improve the effectiveness of a weapon system and consequently the efficiency of the involved forces and to achieve in a like manner an economic use of the resources.

One of the prerequisites to improve the effectiveness of military aircraft operations is to have at all times a full and permanently updated picture from the state of the operational readiness of all involved aircraft weapon systems. This knowledge is mainly derived from the information, which is typically received from integrated monitoring systems installed into aircraft weapon systems. Only with this knowledge the mission activity process is to be optimised, leading to an effective use of all technical and logistical resources.

Improvement is achieved by installation of a comprehensive logistic support level, based on a process driven approach. This process is usually starting with the acquisition and (pre-)processing of information within the aircraft weapon system and subsystems, followed by an immediate transfer of the data to the logistic support facilities and ends finally with loading of the relevant operational data into the aircraft weapon system.

Following this approach Dornier GmbH is currently developing a Ground Support System comprising a mission support part and an engineering support part, suitable for the German Helicopter UH TIGER application.

2. New and Comprehensive Approach

Dornier´s realisation concept for a ground support system - comprising the aspects of mission planning as well as the provision of logistical support - is consequently oriented to improve the effectiveness of military aircraft operations under all relevant conditions especially under the aspects that operation from main operating bases and in austere conditions will be necessary. We are convinced and think that it is of common understanding, that the primary optimisation potential is located in the logistic support processes.

However as long as military mission fulfilment is considered solely under the aspect to translate the resolutions of parliament or military commands into deeds, one may have some reservations to understand our approach, but only in the first view.

Moreover the partitioning of military processes into executive processes and into associated support processes (logistic) is important and advantageous, as it opens new prospects for the design of integrated ground support systems.

Military executive processes are initiated on the highest level, for example by a parliament's resolution for a distinct military mission. According to the military rules this resolution is translated into the chain of commands and ends in a combat order for the relevant operational unit. Once the resolution has been released, then the order has to be executed under all circumstances, whilst the economical aspects are usually of secondary importance.

Concerning the associated and parallel running support processes, these are again triggered and driven by the actions of the operational units. For the support processes however, there are only global directives settled in terms of global orders and budgets. Therefore these processes can generally be compared with business processes in civil industry.

These logistical business processes should not to bee seen to end in itself, but as an essential support, covering all logistic levels, e.g. operational units, depots, repair shops and this logistic support is, what has to be optimised for the operational units. (see figure 1:Overall Scenario)

On operating units level there are the direct and immediate interfaces between executive processes and logistic support processes to be found. As these process interfaces are bearing the main optimisation potential, in the following considerations, this aspect forms the guideline for the realisation.

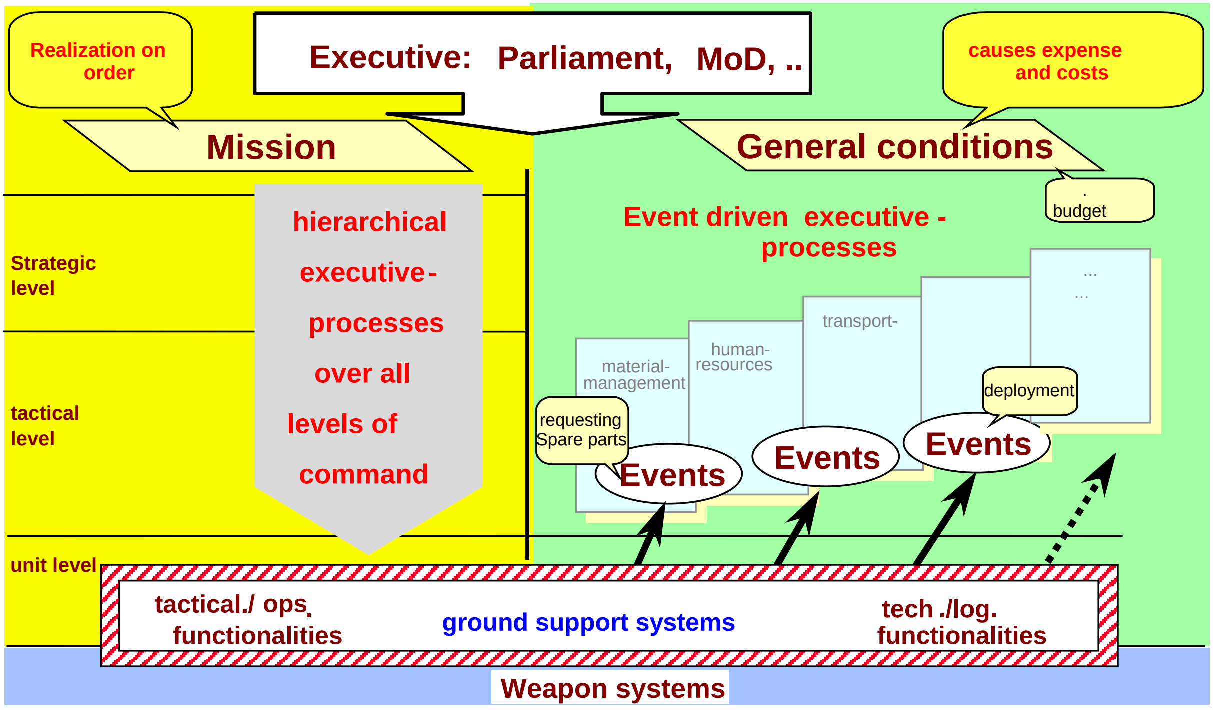

Figure 1: Overall Scenario

In order to raise comprehensively the identified optimisation potential, the aircraft weapon system has to provide some essential technical features:

- A maximum of information, necessary to perform a reliable assessment of the aircraft status, has to be directly accessible at the aircraft. This requires a high grade of onboard data acquisition, data (pre-) processing and data storage capabilities, as they are typically provided by integrated aircraft health- usage monitoring and diagnosis systems. (See: OLMOS, Smart FDRS).The aircraft weapon system has to provide means for electronic data transfer to and from the ground station, e.g. TORNADO MDTS and OLMOS. Breaks in transfer media's have to be avoided wherever possible.
- The integrated onboard monitoring and diagnosis system has to be designed to be flexible to enable adaptation, modification and updates of the integrated software. Experiences gained by the user and usage shall flow permanently into that enhancement process.
- The aircraft weapon system should be able to receive and manage data (e.g. parameter and configuration updates) from an external engineering support system, transmitted via standardised data interfaces and data carriers.

With these prerequisites, as set up above for the aircraft weapon system, it will be possible, that the data available at the aircraft can be efficiently processed and managed by the engineering support system, which is physically representing an IT network system. Such an engineering support system provides full information to all hierarchical command levels within an operational unit, which is necessary to evaluate, maintain and restore the operational readiness of the aircraft weapon system and its technical resources.

Figure 2: Engineering Support System IT-Network

The design of the engineering support system is determined by the following major functional and operational requirements:

- Availability of all task relevant applications for each user at each workstation
- No need for additional or interim manual data processing activities
- All data are available in a consistent data base

Taking into account all operational aspects, the engineering support system has to provide a comprehensive and self-sufficient logistic support in different environments, for example operating from a stable base using a well established communication infrastructure, as well as operating in the field or out of area with communication restrictions and limitations.

Thereby it is insignificant whether the operating unit operates a single type of aircraft, or whether temporary task forces are using different types of aircraft's that have to be supported during a mission.

To achieve this, the engineering support system has to provide the appropriate data interfaces necessary for different aircraft types, in order to enable fast and comprehensive data transfer. Moreover the engineering support system has to have the capability to process different data types and formats as required for different aircraft types as well.

However of highest importance is the all time availability of the information about the actual state of the operational readiness of the aircraft weapon systems involved, which forms the basis for the optimisation of the current operating processes within the military aviation scene. The operating processes are starting with the retrieval of the status information directly at the aircraft and with this information the optimisation process can start to improve the operational readiness, covering all logistic levels.

On operational unit's level, i.e. for the maintenance battalions, the engineering support system provides the following functionalitie's:

- Evaluation of all aircraft status information regarding the operational readiness and initialisation of the necessary steps for appropriate on condition / condition monitored maintenance actions.
- Configuration management covering both, all relevant and significant aircraft items and ground support equipment.
- Provision to use Interactive Electronic Technical Publication features.
- Evaluation and assessment of mission fulfilment ability and stamina.
- Planning and management of spare parts.
- Co-ordination of all technical resources and optimisation of workshop load to guarantee an efficient mission fulfilment.
- Provision of cross servicing ability at any mission location.

3. Engineering Support System based on a Commercial Off-the-Shelf IT-Solution

As described in the in the conceptual considerations above, the requirements for an aircraft engineering support system are compatible to those, known from the civil industry. Therefore the possibility to introduce a COTS-based customised IT-solution, seems to make sense and therefore should be considered.

In contrast to a fully customised and single source solution, the main advantage of such a COTS based IT-solution is the ongoing innovation and improvement process of COTS products, which is driven by the requirements of a high number of users.

Another advantage - but not of less importance - will be the inherent capability to evaluate and compare the economic aspects of military actions, for example the German Air Force engagement in Yugoslavia, based on the experiences of the civil industry.

The recommended IT-solution for the new and improved engineering support system is the application of an widely used enterprise resource planning software, namely SAP R/3, which has been well proven and assessed to be well suited during an feasibility study.

4. General Approach of Process Optimisation

Pre-requisite for a successful realisation of an engineering support system, based on a COTS-product, is the detailed analysis of the operating processes which are typically allocated to aircraft maintenance units. This is in fact a thorough job which has to be done together with the customer / user.
Common and widely used Business Process Reengineering (BPR) tools (e.g. ARIS) are usually starting from a process level which prevents a comprehensive view. Therefore a suitable approach has to be chosen, which allows a general and comprehensive view, whilst applying an appropriate tool filter, without preventing the usage common BPR tools.

The first step to be performed is the description of the whole process landscape, enabling a general overview on that area, which will be in the focus of the reengineering tasks, i.e. the Main Application Area. In the second step a more detailed look is given at the corresponding maintenance management processes, including material management processes, which are applied on maintenance battalion level (Object Groups). In the next step, covering the process matrix levels, the identified macro processes, i.e. the maintenance processes are assigned to objects (Aircrafts, AGE).
Then in the next step on an even more detailed level the most important tasks, which are defining the process flow are identified and optimised together with the user (Task Flow Level).

Following this methodology, a comprehensive overview of the processes is provided and the related functional requirements are documented as well.
This forms also the basis for the customisation of the chosen COTS-product, using common BPR-tools.

The result of this approach is a commercial of the shelf software product, customised to an extend, thus considering the special requirements of military environment.

This conceptual way to proceed is shown in Figure 3: Documentation Level.

5. Summary

The German military aviation scenario is characterised by continuously decreasing defence budgets, rapid and frequent change new mission scenarios, and even worse, by insufficient and obsolete logistical IT-support.

The analysis of the actual situation results that the primary potential for the optimisation of military aircraft operations is identified and located in the logistic support processes. Therefore the effectiveness of these operations can be significantly improved by a consequent process optimisation, applied to all logistic support levels.

However any process optimisation requires basic technical information, which is normally received from integrated monitoring systems, installed into aircraft weapon systems. As a consequence a high grade of onboard data recording, data processing capacity has to be provided. This also is a key factor for a continuously updated knowledge about the actual state of the operational readiness of the involved aircraft weapon systems. This present knowledge forms the basis for an optimal mission planning, taking into account an effective usage of technical and logistical resources.

With respect to the special military requirements, the new improved engineering support system has to provide a comprehensive and self-sufficient logistic support in different environments, for example operating from a stable base using a well established communication infrastructure, as well as operating in the field or out of area with communication restrictions and limitations.

Based on this approach Dornier GmbH is developing a ground support system, comprising mission support and engineering support as well for the German Helicopter UH TIGER. Especially during all development steps of the engineering support part, the military user is fully involved in the business process optimisation leading to an agreed and harmonised solution.

References

[1] U. Schulz — Onboard life Monitoring System: Hardware Equipment, Onboard an Ground, 14th AIMS Symposium, 16th Sept. 1987

[2] U. Schulz, J. Bott — Integrated Engine Control and Monitoring with Experiences Derived from OLMOS, 16th AIMS Symposium, 17th Sept. 1991

[3] A. Schick — Technical Logistic Support with Modern Solid State Flight Data Recorder System, 18th AIMS Symposium, 19th Sept. 1995

[4] A. Schick — A New Smart Flight Data Recorder for Helicopters and Small Aeroplanes, 19th AIMS Symposium, 05th Mai 1999

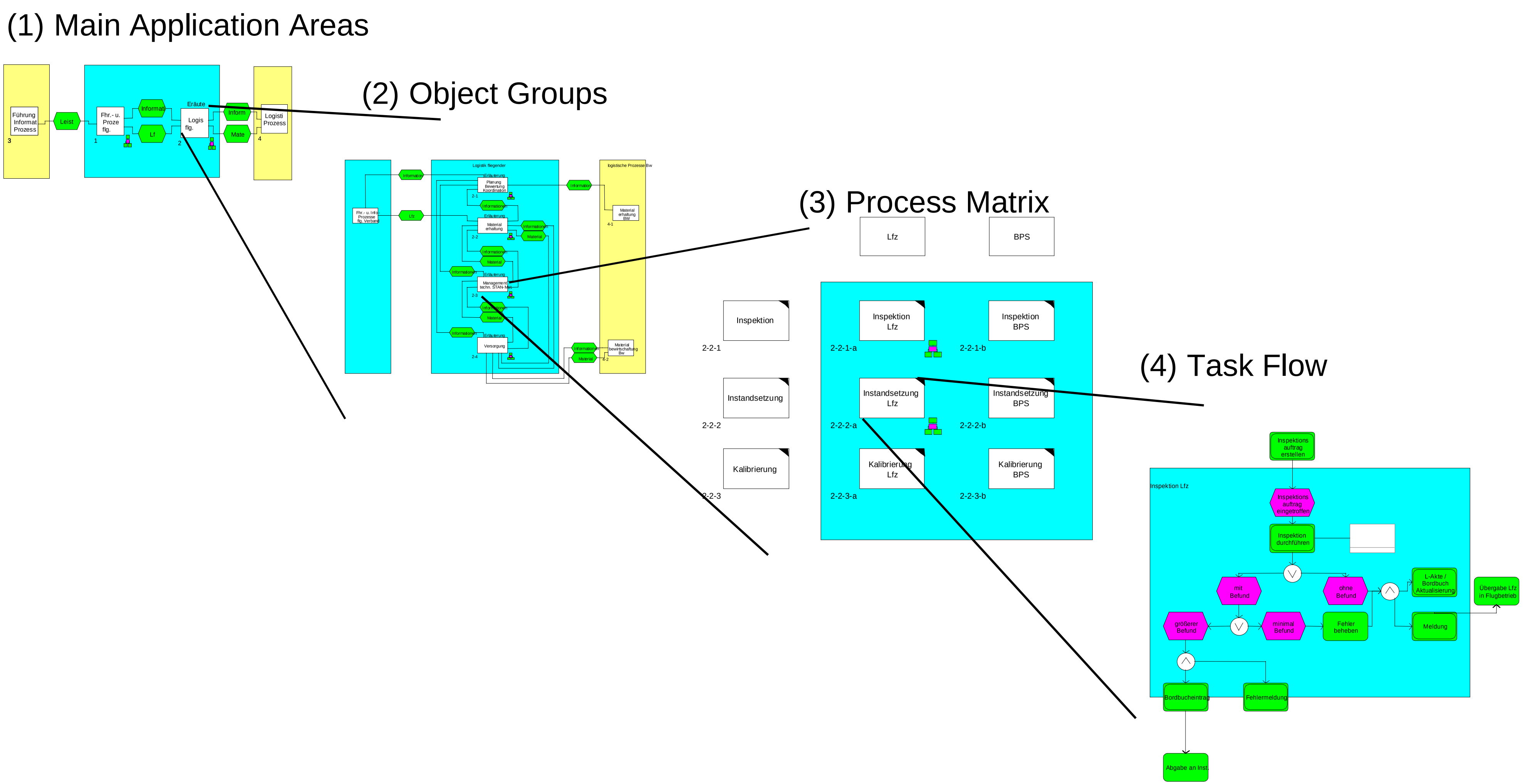

Figure 3: Documentation Level

Engine Condition Monitoring

DATA CONDITIONING METHODS FOR AERO ENGINE SIGNALS AND THEIR INFLUENCE ON THE ACCURACY OF FATIGUE LIFE USAGE MONITORING RESULTS

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Abstract

During the design of monitoring systems for aircraft engines it is common to include algorithmic provisions to reduce the anticipated noise content of the input signals by applying some sort of filters at a suitable stage of processing. A tacit assumption is often made that such filtering will make the system more robust with respect to noise or even data errors.

The elimination of uncorrelated noise in a signal emanating from a physical process may have a large effect on the predictability of this signal, thus enabling high data compression rates in a dedicated or embedded flight data recording system. However, little is known about the influence of filtering the input into engine fatigue life usage calculations on the outcome of various models used in present monitoring systems.

A simplified, yet realistic mathematical model is used to describe the thermal response, stress and fatigue behavior of fracture critical parts in an aero engine compressor. Using this model, the consequences of applying digital recursive filters to recorded engine data are investigated. The analysis concentrates on statistical methods to assess the accuracy. From the results some guidelines are derived that allow a more systematic selection of filter parameters when a predefined accuracy of the fatigue life usage results is required.

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Introduction

Existing engine monitoring systems for military aircraft include methods to calculate the fatigue life usage of rotating engine components known to have the potential to destroy the aircraft or to cause at least high financial loss in case of an in-flight failure. At first sight, on-board fatigue life usage monitoring (LUM) seems to have reached maturity over the years and will be applied to the engines of the Tiger helicopter and of the Typhoon Eurofighter.

These systems use on-board processing based on a few (less than 10) engine or aircraft signals to calculate the life consumption for all fracture critical parts of the engines. Figure 1 shows a typical spool speed signal of the HP spool speed of a military engine for one flight. The calculation uses mathematical models of the thermal, mechanical and material properties of the engine and its components. A simplified example of such a model will be presented later in this paper.

The algorithms are based on existing knowledge of failure mechanisms and take into account the experience (e.g. test results, inspection findings) available at the time when the on-board software is specified. Although some flexibility can be built into the monitoring software (e.g. by using loadable parameter sets), the software is not able to cope with newly detected damage mechanisms or with unanticipated configuration changes of the engine. It is therefore necessary to perform regular updates of the on-board software. Even if the system architecture is carefully designed, a fleetwide software update together with the accompanying adaptations in the logistic system is at least very expensive and carries the risk of introducing inconsistencies into the fatigue life usage data [BP97, PR95].

The most challenging situation for existing LUM systems occurs, if damage (e.g. cracks) is detected during inspection of components with long in-service times without any proper coverage of the corresponding areas in the on-board LUM algorithms. Current practice is to use statistical correlations between the computed damage at monitored areas and model calculations of loads at the newly detected area to get an idea, which parts would have be to removed from service or to be inspected. The inevitable consequence is a considerable uncertainty and loss of usable life for the

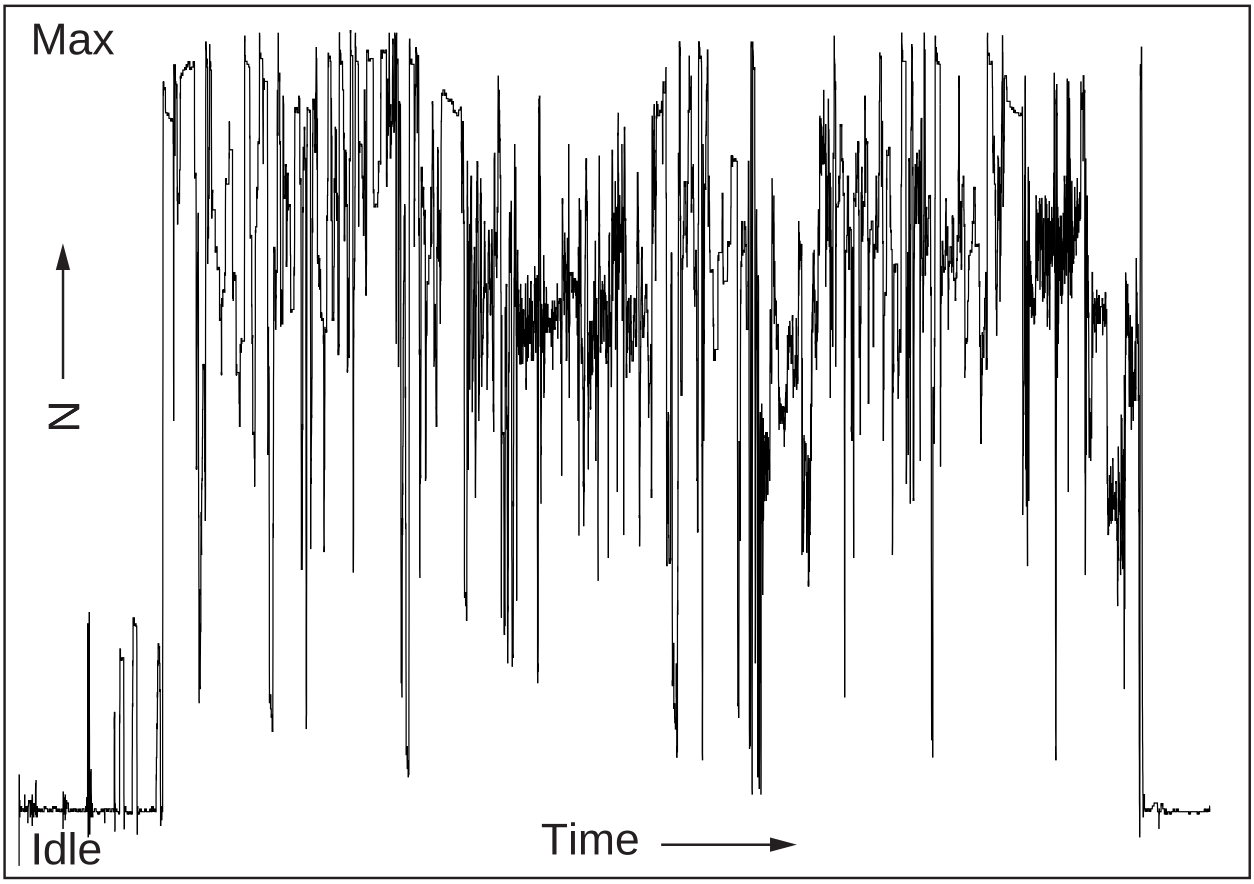


Figure 1: *Spool speed signal of a military jet engine for one flight*

components in question. The only way out of this dilemma would be the availability of a complete running history of the part, including recordings of all flight data necessary to recalculate the load history and life consumption at newly detected critical areas. None of the existing systems currently provides such long time history data.

Aviation recorders, either in the form of crash protected devices for accident investigations or as very large capacity airborne quick access data recorders for maintenance support are now profiting from the availability of storage media (e.g. solid state memory devices) with high reliability and enormous capacity. Many of the existing tape based devices are currently being replaced by solid state FDRs, both in the military [SS95] and in the civil transport [Gro99] environment. However, the direction of development seems to be aiming either at an ever increasing number of parameters or at higher sampling rates, which are needed for incident and accident investigations [HK99]. In contrast LUM applications only need a few parameters with relatively low sampling rates.

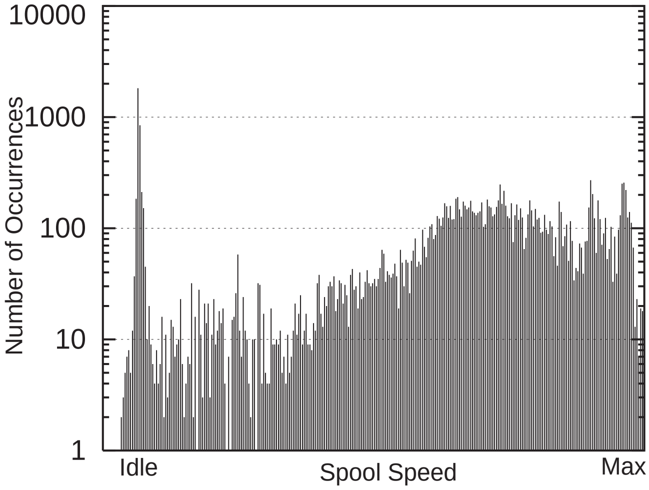


Figure 2: *Histogram of spool speed values for one flight*

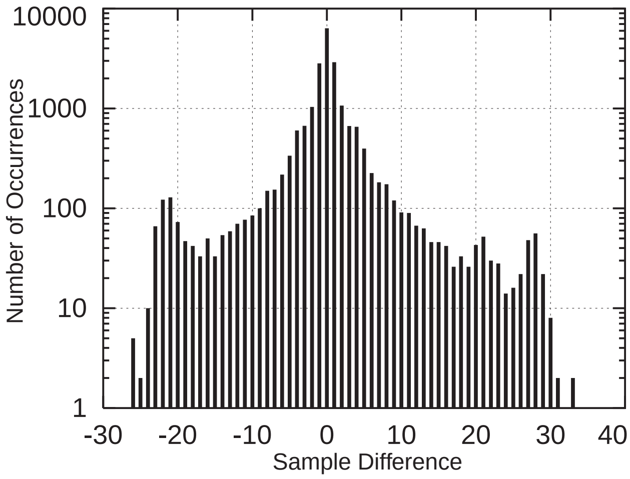


Figure 3: *Histogram of spool speed delta values for one flight*

Long Term Recording of Engine Data

The monitoring function of the RB199 engines, which is part of the German Tornado OLMOS system [BP97], uses a data rate of 2Hz and only 8 input parameters plus some logical signals. All those data are of course available on the general purpose FDR [SS95], which only stores a few hours of data, however. No regular readout of those data is performed, unless other problems require an analysis of the FDR data, and it would be completely impractical to require a data readout of all FDR parameters just to get the engine data.

Although some previous work on flight data compression has been reported [SM89], no reliable figures were available on the information content of the signals entering a LUM calculation. The current practice in most existing recording systems is to use some sort of Run Length Encoding (RLE), which is known to be rather sensitive to small disturbances in the signals.

Some preliminary investigations were performed to get an idea on the benefits of various data compression methods, if applied to the signals needed for a LUM calculation. It turned out that different strategies have to be applied to the different signal types and that it is absolutely necessary to gain detailed insight in the signal properties to design a recording strategy yielding practicable data volumes when the full running history of an engine shall be stored. A strategy using delta coding together with statistical coders was found to be most promising. Those techniques depend heavily on the statistical properties of a signal. Examples for a spool speed signal are shown in Figs. 2 and 3.

The determination of the power density spectrum of the autocorrelation of a signal, which is the most important tool to detect noise, needs some precautions to produce meaningful results

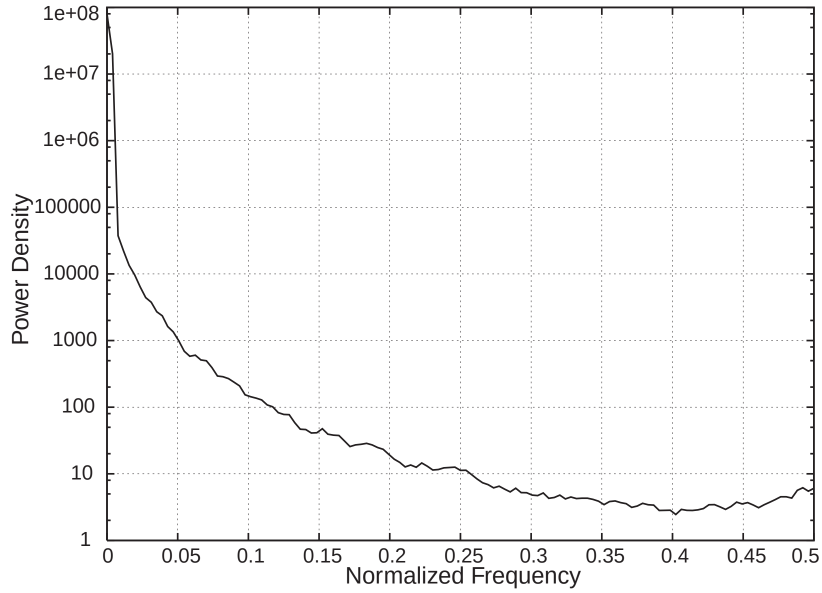


***Figure 4:** Power density spectrum of spool speed autocorrelation*

[Smi97]. The method is usually applied to periodic signals, where distinct frequencies with some physical meaning (resonances) are sought for. If applied to other signal types, adapted analysis methods are required. A computer program "SPPOWR" in [SD93] was used to produce the spectrum data shown in this presentation. The program computes the average periodogram of a real data sequence by averaging the Fourier transforms of overlapping segments in the data. A typical segment length used, that has to be a power of 2, was 128. To mitigate boundary effects, a Hanning data window is applied to the single segments.

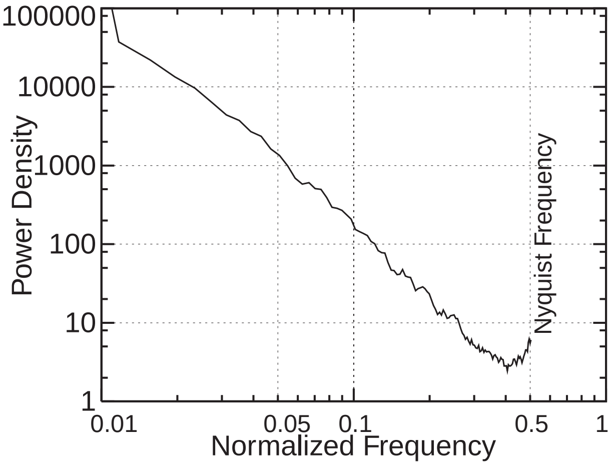


***Figure 5:** $\sim f^{-\beta}$ behavior of spool speed signal*

Figures 4 and 5 reveal a signal behavior that is found in many systems with interacting processes with different time scales. An enormous literature exists on this so called *1/f* noise [Ber94]

An understanding of the signal properties is necessary to find preprocessing techniques with minimum influence on the information content. Digital filters are the most powerful tools to accomplish certain intended signal modifications [OS89, PB87].

Digital Lowpass Filters

Systems that smooth input signals, thus removing noise are usually called filters. They can be treated as a black box, that modifies a discrete input signal x into an output signal y. Certain systems (discrete, linear, time-invariant, continuous) are fully characterized by the so-called impulse response. That is the output of the system, if a delta impulse (=1 for time step n=0, 0 otherwise) is supplied as input. An arbitrary input signal x(n) will then produce the output y(n), where y(n) is the convolution of the impulse response with x(n).

For the present investigation only the special case of Butterworth filters was considered. These filters are easily implemented in the time domain and have a smooth amplitude response. To speak of amplitudes we have to look at the representation of the impulse response in the frequency domain. The convolution theorem states, that the Fourier transform of the output signal is equal to the product of the Fourier transforms of the input signal and that of the impulse response. If represented in the usual way as complex number, the convolution theorem yields the result, that the magnitude of the Fourier transform of the output signal is equal to the product of the Fourier transforms of the input signal and of the system's impulse response.

For Butterworth filters this magnitude of the Fourier transform of the impulse response is a very smooth function of frequency (Fig.6). This amplitude response approaches that of an ideal filter, if the filter order increases. An ideal low pass filter suppresses frequencies outside of its pass

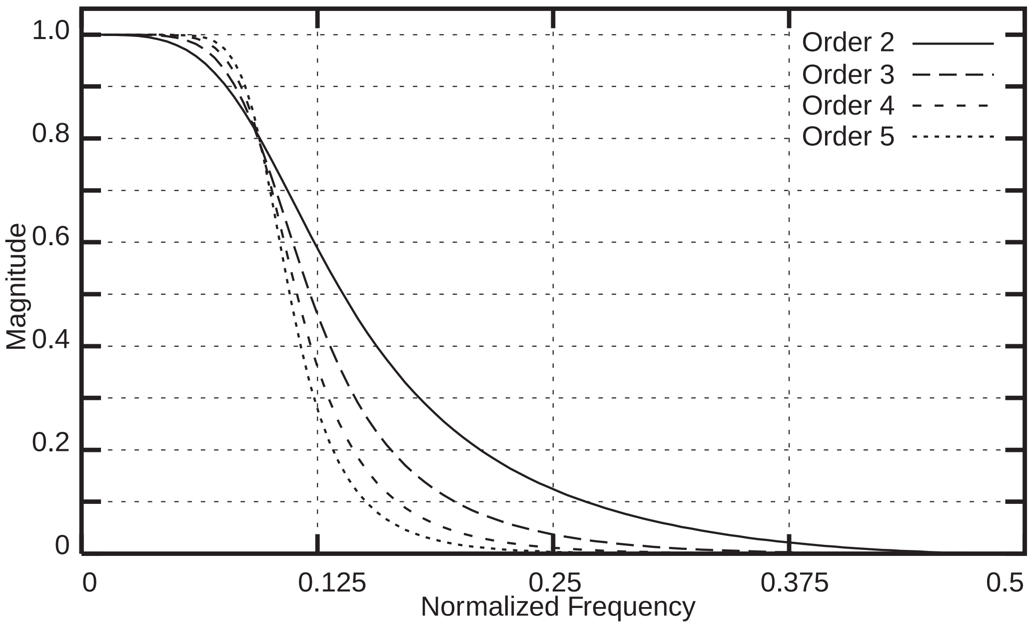


Figure 6: *Amplitude response of 4 digital Butterworth filters with nearly identical cutoff frequency*

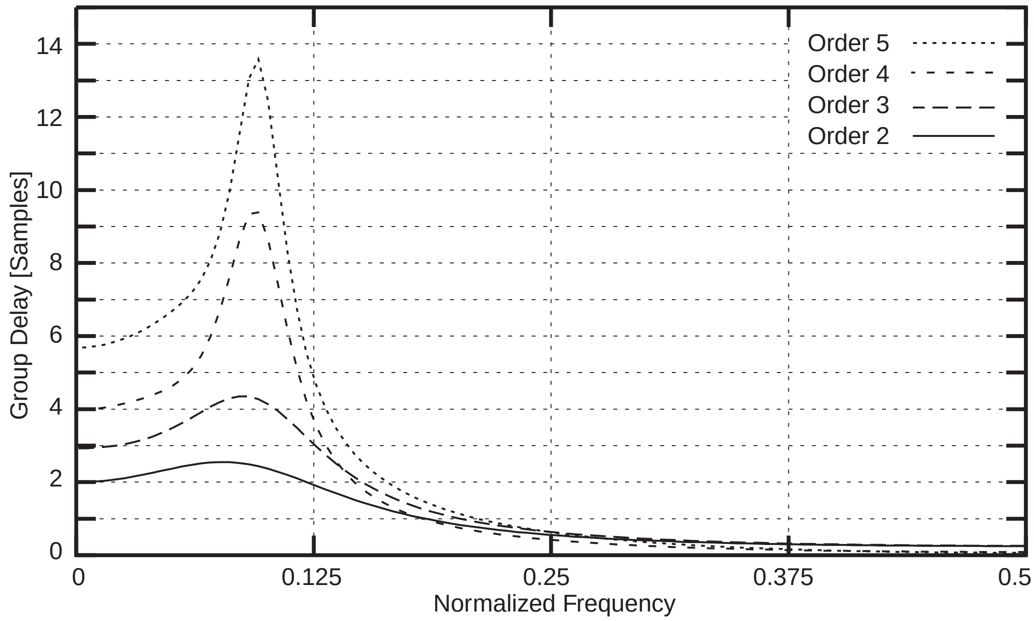


Figure 7: *Group delay (negative phase derivative) for the 4 filters of Figure 6*

band completely and has no influence on frequencies inside its pass band. Unfortunately such filters can not be obtained with practicable effort and one has to stay with approximations of the desired ideal behavior. Butterworth filters put emphasis on the quality of the amplitude response at the expense of the phase response. A linear phase response is tantamount to the filter exerting a simple time delay on the signal. Since both goals are interrelated, one has to accept a nonlinear phase response, that means the frequency contents of the signals are variably delayed (Fig. 7). A ripple in the amplitude response, which is quite common for linear-phase filters, would induce the risk of creating spurious new extreme values in the output. As a consequence new cycles without physical meaning could be created, thus distorting subsequent calculations of fatigue damage.

A short sketch of how to design and implement digital Butterworth filters follows. The available literature concentrates on analog Butterworth filters and on the Fourier transform of their impulse response. To design a digital Butterworth filter from the given Fourier transform of its analog impulse response, the following steps have to be performed:

1) Determine the analog impulse response

2) Compute the Laplace transform of the analog impulse response

3) Adjust the frequency representation with a bilinear transform
4) Compute the Z-transform of the digital impulse response
5) Compute the Fourier transform of the digital impulse response.

The result is the digital impulse response. (See example in Fig. 9). Applying the outlined procedure, a digital Butterworth filter is determined by its order N (the number of samples used to compute the actual output), the sampling interval of the discrete input signal T and the cutoff frequency ω_g. By selection of these three parameters the filter is fully determined.

1) Poles of analog filter: For $k \in \{0, \cdots, N-1\}$ let $z_k := \omega_g e^{i\pi \frac{2k+1+N}{2N}}$.

2) Determine coefficients b_k for $k \in \{0, \cdots, N\}$: $b_k := \binom{N}{k} \dfrac{{\omega_g}^N T^N}{\prod_{l=0}^{N-1}(2 - T z_l)}$

3) Determine coefficients a_k for $k \in \{0, \cdots, N\}$ by expanding the expression: Let $v \in \mathbb{C}$ arbitrary: $\displaystyle\sum_{k=0}^{N} a_k v^k = \prod_{l=0}^{N-1} \left(1 - \left(\frac{2 + T z_l}{2 - T z_l}\right) \cdot v\right)$.

4) Now let $x : \mathbb{N} \to \mathbb{C}$ be a discrete input signal, then the filtered output signal $y : \mathbb{N} \to \mathbb{C}$ is obtained by: Let $n \in \mathbb{N}$:

$$y(n) := \sum_{k=0}^{N} b_k x(n-k) - \sum_{k=1}^{N} a_k y(n-k).$$

Input and output shall be 0 for negative n: $\forall n \in \mathbb{Z} : n < 0 \Rightarrow x(n) = y(n) = 0.$ One remark on the cutoff frequency ω_g: The magnitude of the Fourier transform at the point: $2\arctan(\frac{T\omega_g}{2})$ is $\frac{1}{\sqrt{2}}$ for Butterworth filters of arbitrary order. At the corresponding frequency (transition or cutoff frequency) the amplitude has a turning point. Higher frequencies will be suppressed increasingly. The phase response is rather nonlinear in this transition area, with an extremum of the group delay (derivative of the phase response) near the cutoff frequency. (Figure 7).

For our investigation 25 Butterworth filters were used with a maximum order of 5. This choice was based on the idea to avoid longer lags between data acquisition and data storage, which becomes difficult to handle at engine shut down. Besides from that, the implementation of higher order filters poses increasing difficulties in accuracy and stability. The cutoff frequencies were chosen to match the values 1.0, 2.0, 3.0, 4.0 and 5.0 for the phase derivative (group delay) at zero frequency. This results in a linearly varying output signal delayed by whole multiples of the sampling interval for a linearly varying input signal after the overshoot oscillations (only present for filter orders greater than 1) have decayed. A derivation is given in [Gra00].

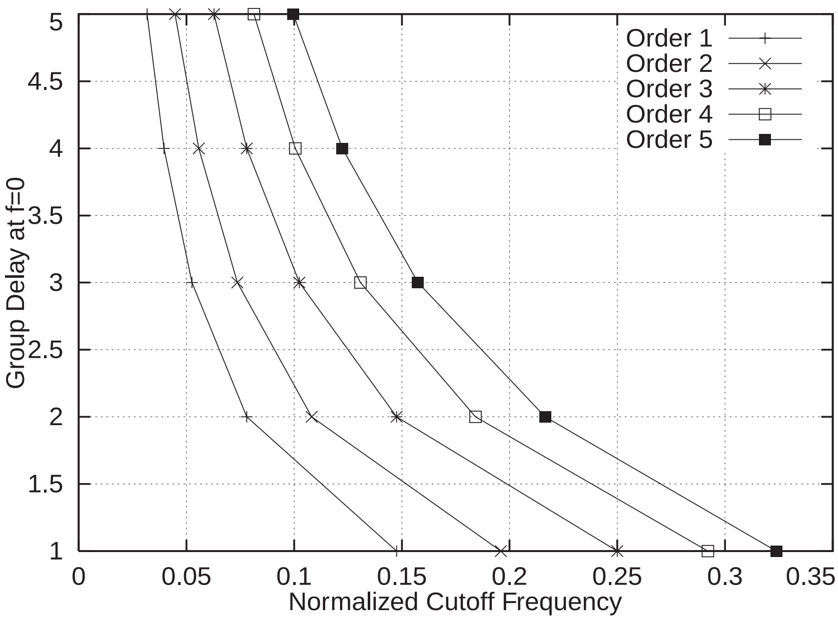


Figure 8: *Correspondence between group delay at f=0 and cutoff frequency for Butterworth filter family*

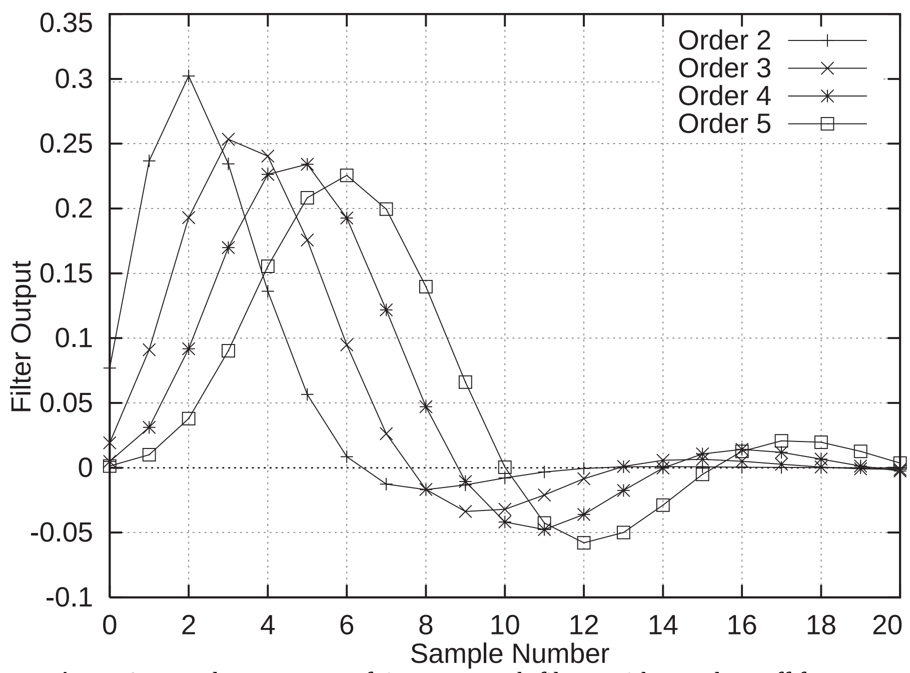


Figure 9: *Impulse response of 4 Butterworth filters with equal cutoff frequency*

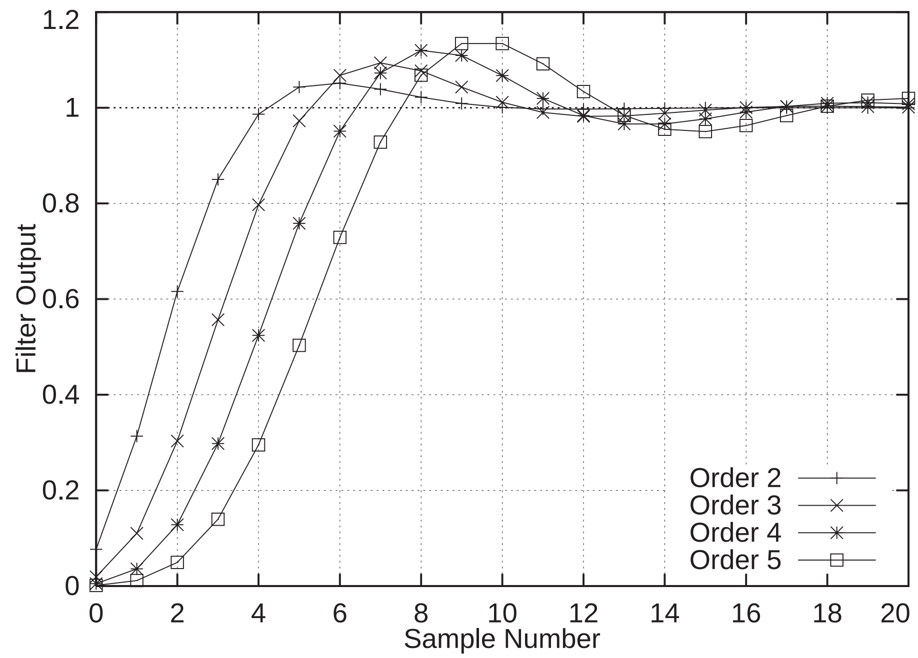


Figure 10: *Step response of 4 Butterworth filters with equal cutoff frequency*

The correspondence between group delay at zero frequency and the cutoff frequency is shown in Fig.8. The usual frequency scaling with the Nyquist frequency set to 0.5 is used in this and all figures with a frequency abscissa. For convenience the 25 investigated filters are addressed by their filter order and their delay time for linear input in the following discussion. Filter (5,2) means order 5 and 2 time steps delay at f=0. The conversion into cutoff frequency can be done using Figure 8. For example, the filter (5,2) has an approximate cutoff frequency of 0.22. In the Figures 9 and 10 impulse and step responses are shown for the filters (2,2), (3,3), (4,4) and (5,5), which all have a cutoff frequency near 0.1. This makes them suitable candidates for the demonstration of the effects of filter order.

Overshoot is only present for filter orders >1, and its magnitude increases with increasing filter order. Order-1 filters approach the input signal in an asymptotic manner without overshoot. The overshoot properties of filters are best visible in their step response. (Fig.10). The overshoot properties will turn out to be crucial for the accuracy of computed fatigue life usage from spool speed (N) signals. On the other hand, the impulse response is immediately linked with noise reduction. A single spike in an otherwise smooth signal is reduced by the amount shown in Figure 9. E.g. the filter (2,2) will transform a single noise pulse into an output with 30% the height of the raw pulse with a subsequently decaying oscillation.

To give an idea on the coefficients of the Butterworth filters, 2 formulas are given:
The output y(n) of filter (2,2) at time step n is determined by:

$$y(n) = (x(n) + 2\, x(n-1) + x(n-2) + 14\, y(n-1) - 5\, y(n-2)) / 13.$$

The corresponding formula for filter (4,4) is: y(n) =

$$0.0049061458\, (x(n) + x(n-4)) + 0.019624583\, (x(n-1) + x(n-3)) + 0.029436875\, x(n-2) + 2.3615368686\, y(n-1) - 2.301335\, y(n-2) + 1.0470692\, y(n-3) - 0.18576948\, y(n-4).$$

Filter Application Examples

In this chapter results of the application of some of the proposed filters to measured engine signals are shown, together with the influence of filtering on the statistical properties of the modified signals. For the intended compressed coding, which capitalizes on statistical predictability, the advantages of applying filters will become visible in an example with heavy noise content (Figure 13).

As visible in Figs. 4 and 5, properly acquired spool speed (N) signals usually have a negligible noise content. That means that nearly every filter will have an influence on the nonrandom information contained in the signal. This may have further consequences, when the signal is used as input in a subsequent fatigue life usage calculation. Figures 11 and 12 demonstrate the effects of filter order on filtered signals, if the Butterworth filters (2,2) and (4,4) are applied to a high pressure spool speed signal of a military jet engine, sampled at 2Hz. It is clearly visible that the filter (2,2) underestimates most peaks, whereas filter (4,4) follows the input signal more closely and produces an overshoot at the absolute maximum at Time≈56s. In both figures the time delay for f=0 is used to synchronize the output with the raw input. The same method would have to be applied when recovering data from a recording system that applies filters. Although theoretically only valid for slowly changing signals – note the strong frequency dependence in Fig.7 – shifting the filtered signal by an integer amount is computationally very simple and in most cases sufficiently accurate to bring into phase signals that have been processed by filters of different delay.

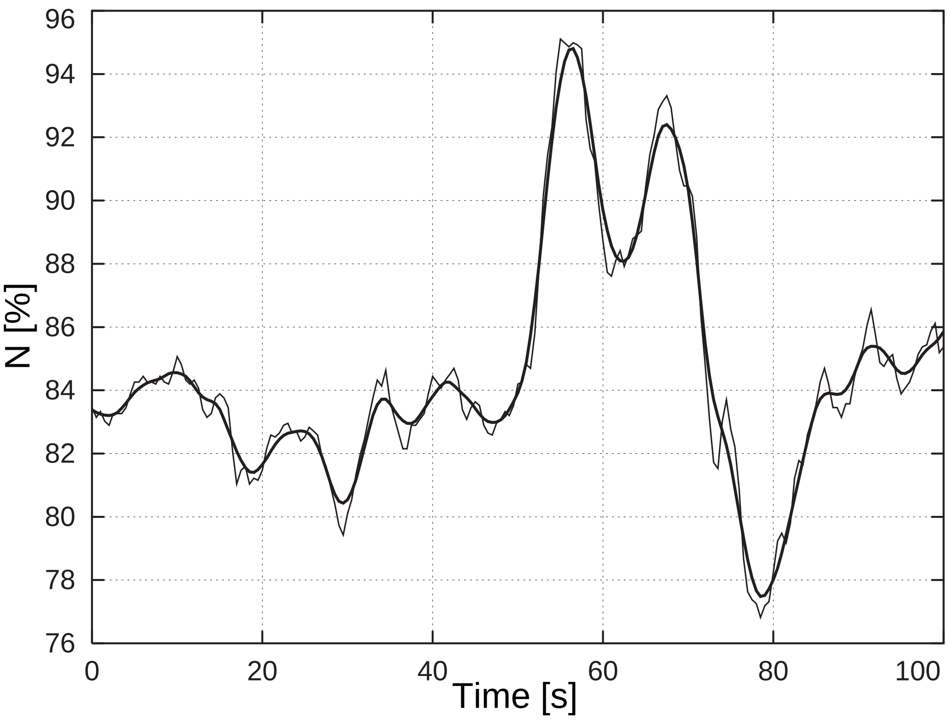


Figure 11: *Application of Butterworth filter (2,2) on spool speed signal (corrected delay)*

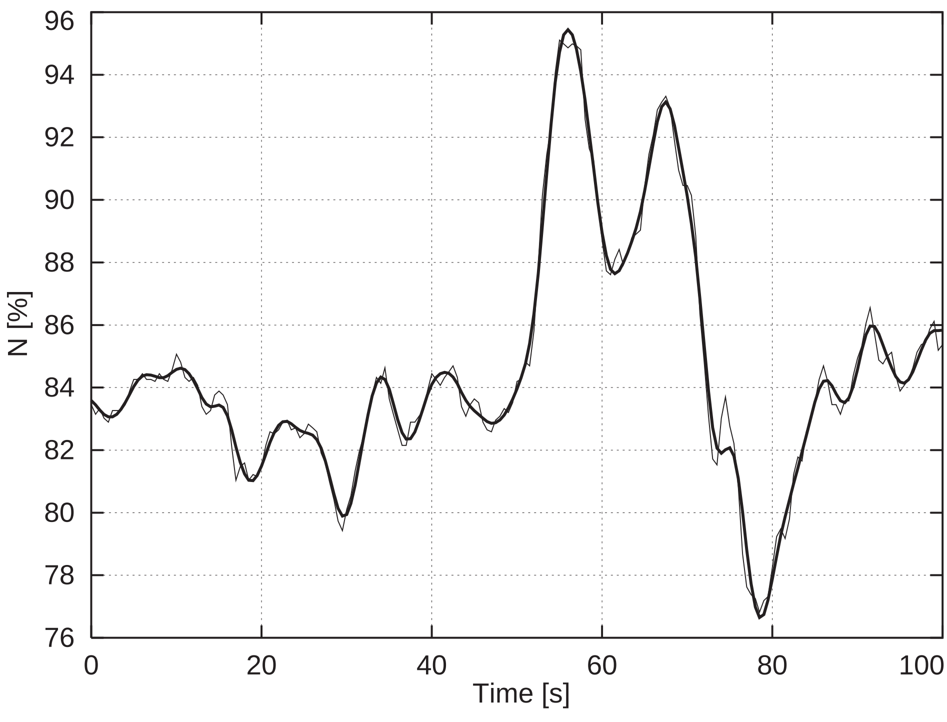


Figure 12: *Application of Butterworth filter (4,4) on spool speed signal (corrected delay)*

The second example deals with a turbine blade temperature (TBT) signal mea-

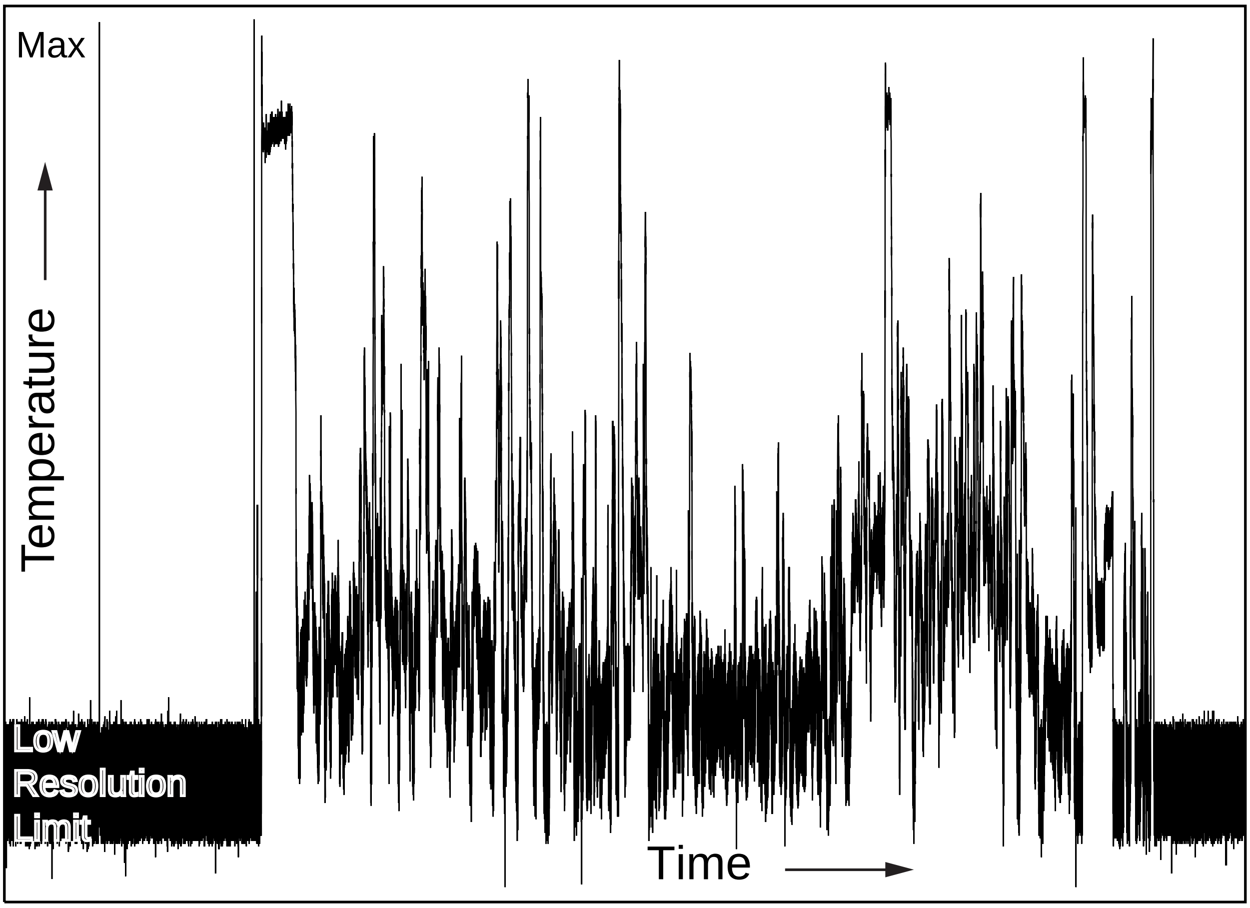


Figure 13: *Turbine blade temperature raw signal with high noise content*

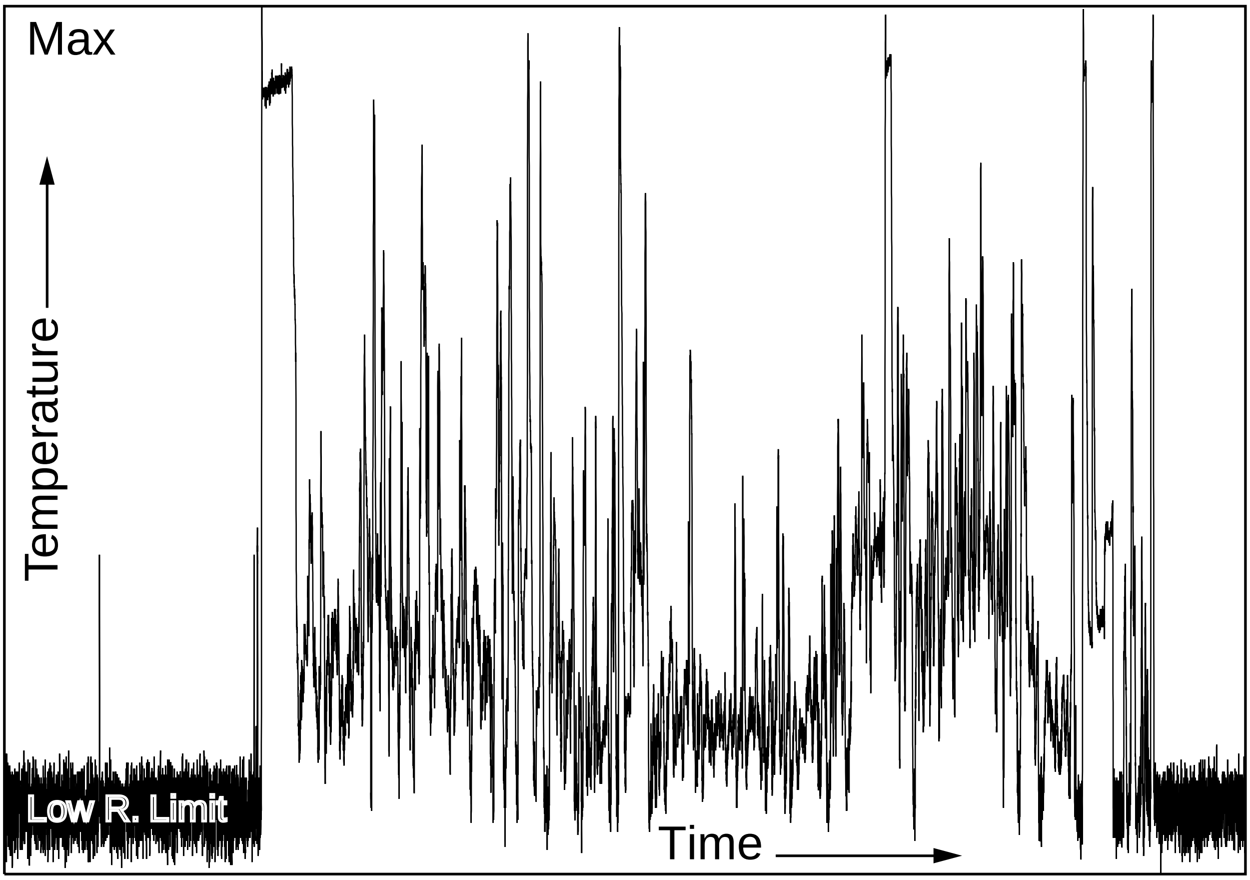


Figure 14: *Turbine blade temperature signal after application of filter (2,2)*

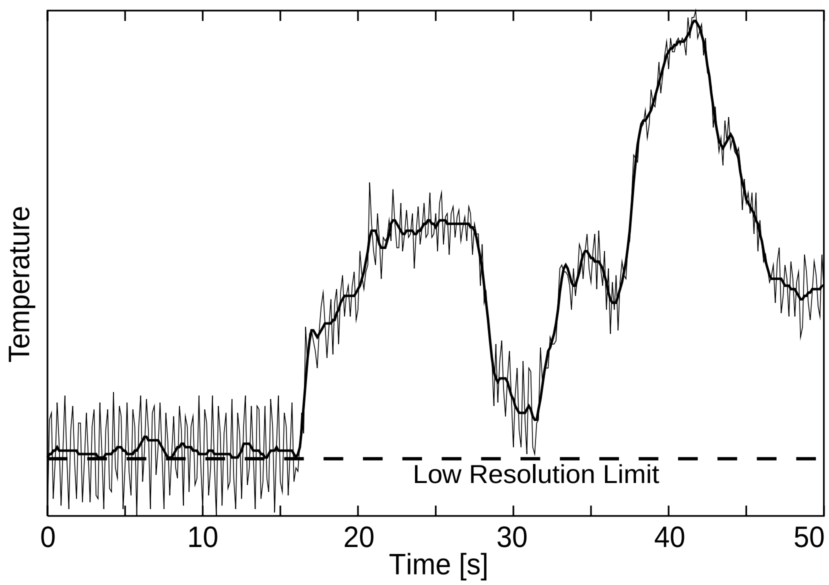


Figure 15: Turbine blade temperature signal with periodic noise, result of application of filter (4,4)

sured by an optical pyrometer. The main reason to measure this signal is for the engine control system, where it is used to prevent the engine from running to hot. Besides from that, the TBT signal is the most important parameter for monitoring creep life consumption of turbine blades. For some blade types creep may be the life limiting factor. Creep is usually only present, if high spool speeds and very high blade temperatures occur simultaneously. The metal temperature calculation in the creep monitoring algorithm usually takes the measured TBT as input to some heat conduction equation. The temperature at the blade cross section considered to be at the highest creep risk therefore follows the TBT with some time delay and considerably smoothed.

The pyrometer output is fed into a linearizing amplifier, where it is converted into a voltage signal with sufficient signal strength to be input into a standard A/D converter for further processing or recording. Due to the ~T^4 law of heat radiation no usable output is produced by the amplifier at idle or moderate power setting of the engine. This is acceptable for the engine control system, because only high values will have the chance to activate the limiter function. Therefore the function checks performed by the maintenance personnel often ignore the behavior of this signal at low power settings. The signal shown in Figure 13 has very high noise content, especially in the mentioned low resolution range. If such a signal is fed into a recording system that tries to reduce storage space using statistical prediction, the compression efficiency will be rather poor (it would, of course, be poor with any other compression technique, e.g. RLE, too).

A closer look at the signal details in Figure 15 reveals that there is a nearly periodic background noise of considerable amplitude superimposed to that part of the signal following the power setting of the engine. Without going into details this is a known phenomenon for certain combinations of amplifiers and control units for a particular engine type. For use in a creep life calculation, the "high" frequency behavior of the TBT signal is clearly of negligible influence. A suitably chosen filter can be used to cut off the superimposed fluctuations, with the side effect of greatly improving the predictability of the signal.

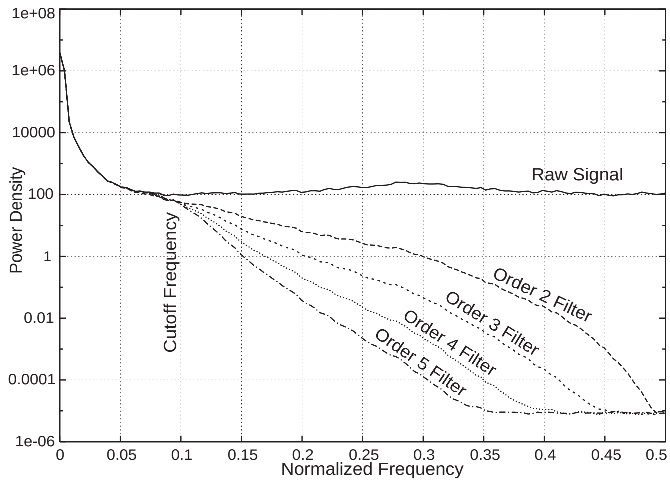


Figure 16: Influence of filter order on stop band attenuation

To select appropriate filter parameters for a recording application it is advisable to look at the periodogram (power density spectrum of the autocorrelation function). Figure 16 shows a periodogram for the data of Fig. 13. As already noted for the N signal spectrum in Figs. 4 and 5, the spectrum starts with a ~$1/f^{\beta}$ part characteristic for the long-term correlations present in the signal. At f=0.08 the spectrum becomes horizontal, thus indicating "white noise". There is an additional maximum around f=0.28, that coincides with the periodic content of the signal with a physical fre-

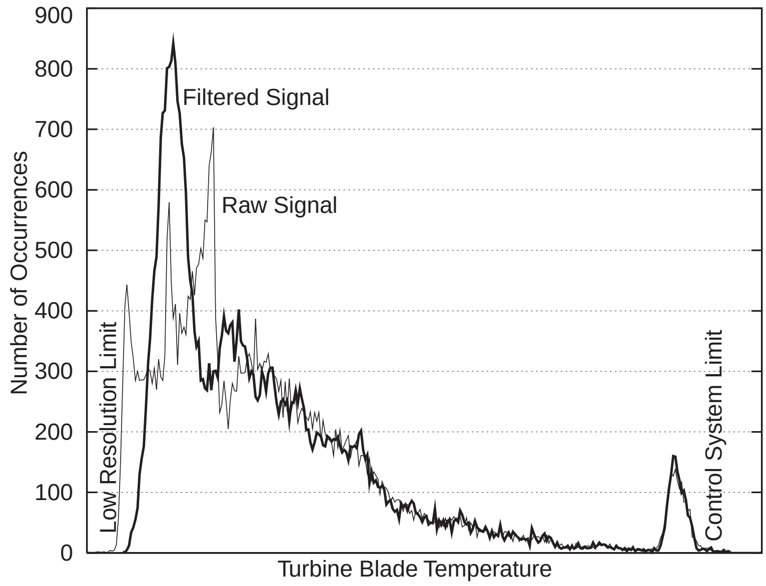


Figure 17: *Histogram of raw and filtered temperature signal*

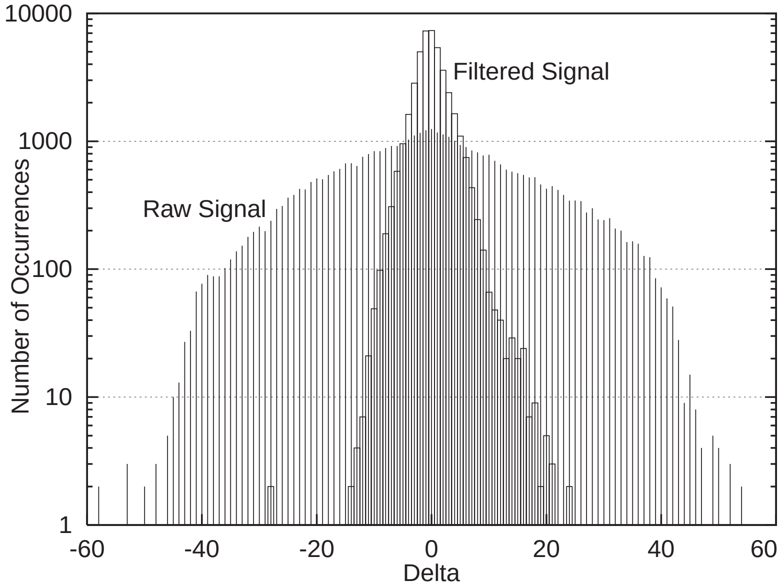


Figure 18: *Influence of filter application on histogram of temperature delta values*

quency ≈2.2Hz (sampling interval: Δt=0.125s), visible in Fig.15. The four filters already used as examples in Figs. 6 and 7 have been applied to the raw data of Fig.13. As expected the power density of the filtered signals starts to drop around f=0.08 and decreases at a rate dependent on the order of the applied filter (all 4 filters have approximately the same cutoff frequency f_{cut}=0.1). The effect of filter application is clearly visible also in the time domain, either in the overall plot of Fig.14, where filter (2,2) was used, or in the detail plot of Fig.15, with filter (4,4). Comparing Figs.13 and 14, the reductions of noise amplitudes in the low range and of sharp data spikes to ≈30% of their raw value is to be noted. This is to be expected from the impulse response of the filter (2,2) shown in Fig.9.

Influence of Filters on Statistical Signal Properties

Using the TBT data of the previous chapter, some consequences of filter application will be shown. The histogram of signal occurrence counts in Figure 17 reveals an important result for this signal type. The filter mainly influences the low temperature range, whereas the high temperatures are nearly unaffected. The 3 peaks in the raw signal histogram are a consequence of the periodic noise and are completely removed in the filtered signal. The filter output of the four different filters was fed into a creep calculation for a turbine blade, leaving other input parameters (e.g. spool speed) unchanged. The relative difference between all results was less than 0.08%, including the result with the raw TBT. The reason is that only temperatures in the peak at the upper end of the range can significantly contribute to creep damage. Other parameters indicating blade creep, as time spent above certain temperature limits will also remain unaffected. The use of such simple counts for inspection planning is discussed in [Bra00], showing only a poor blade failure prediction capability.

The most important signal properties for the application of delta encoders are of course the bandwidth and distribution of differences between successive samples (delta). A small bandwidth with most data centered around zero will improve the prediction success of a coder, thus improving compression rates. Fig. 18 (note the logarithmic scale) shows a dramatic reduction of the delta scatter. Even a very simple coder only exploiting the number of bits necessary to code the delta values could save nearly two bits/sample due to the reduced range. More sophisticated coders, whose description is beyond the scope of this presentation, use statistical information gathered

Figure 19: *Occurrence counts for order-2 contexts of raw blade temperature delta signal*

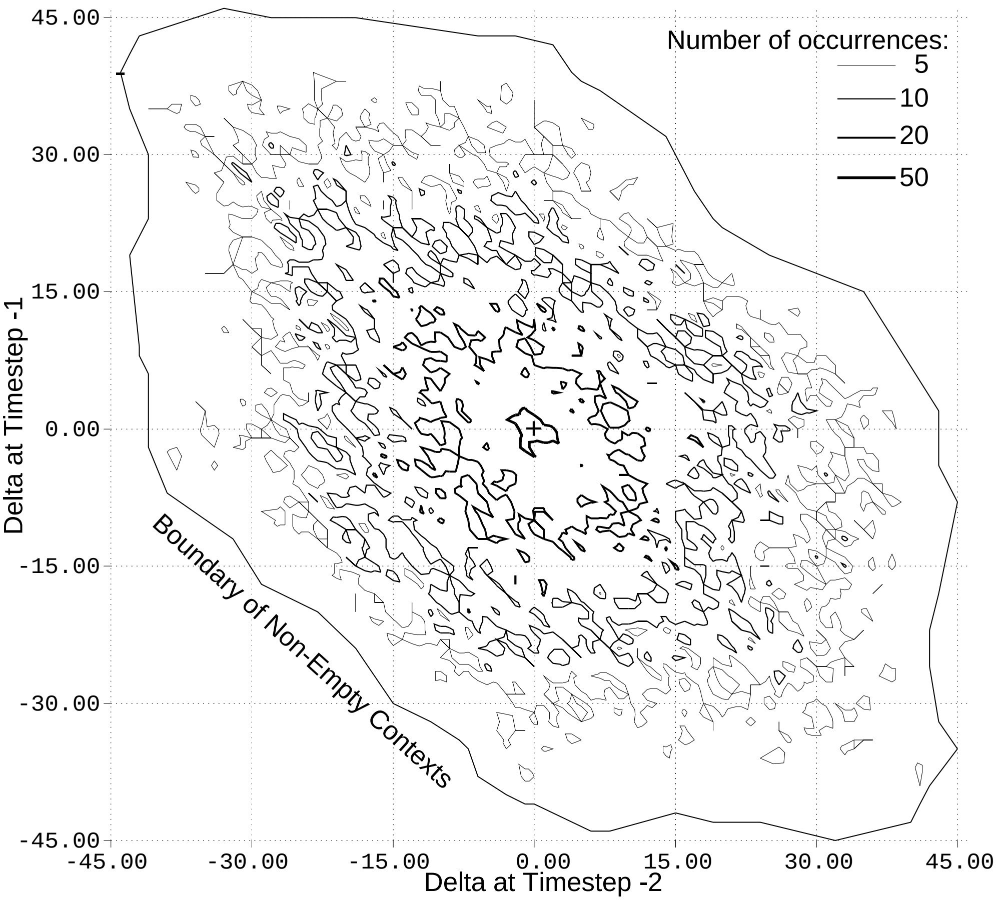
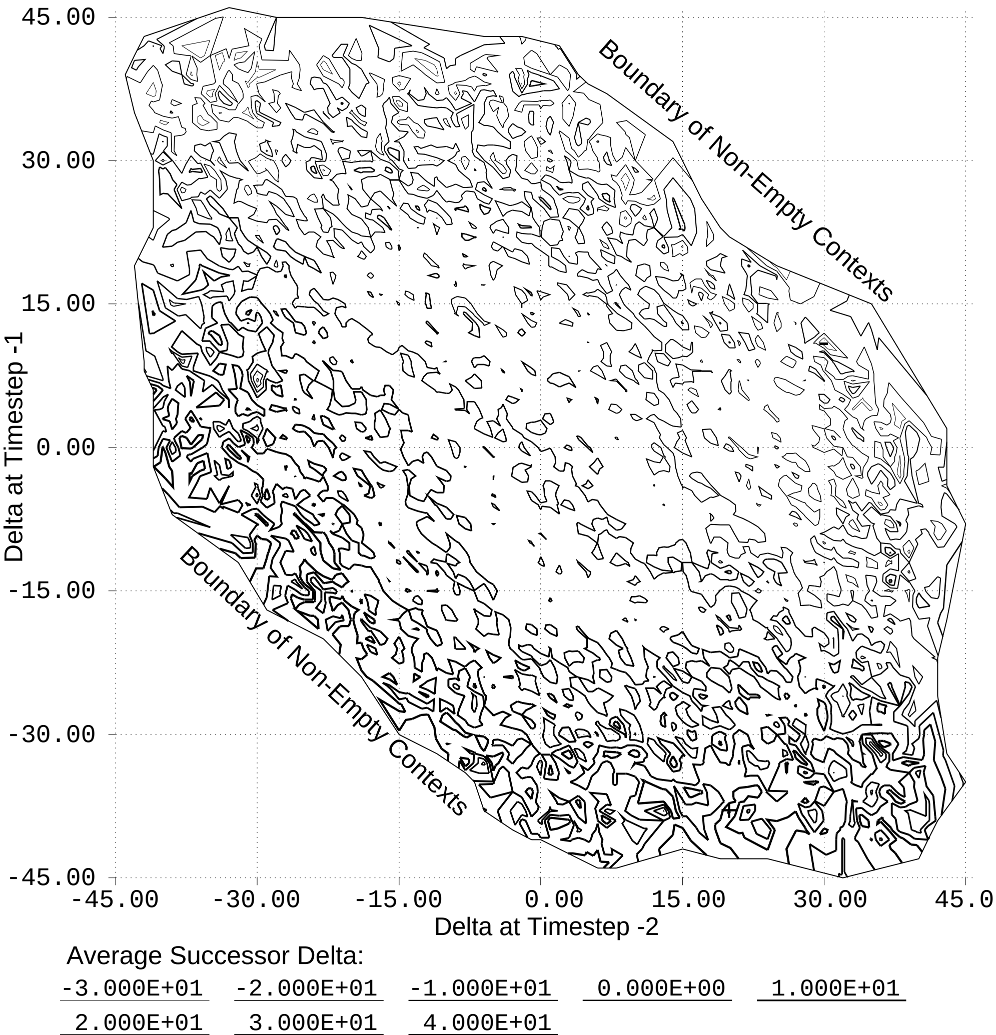


Figure 20: *Mean successor delta of order-2 contexts for raw temperature signal*



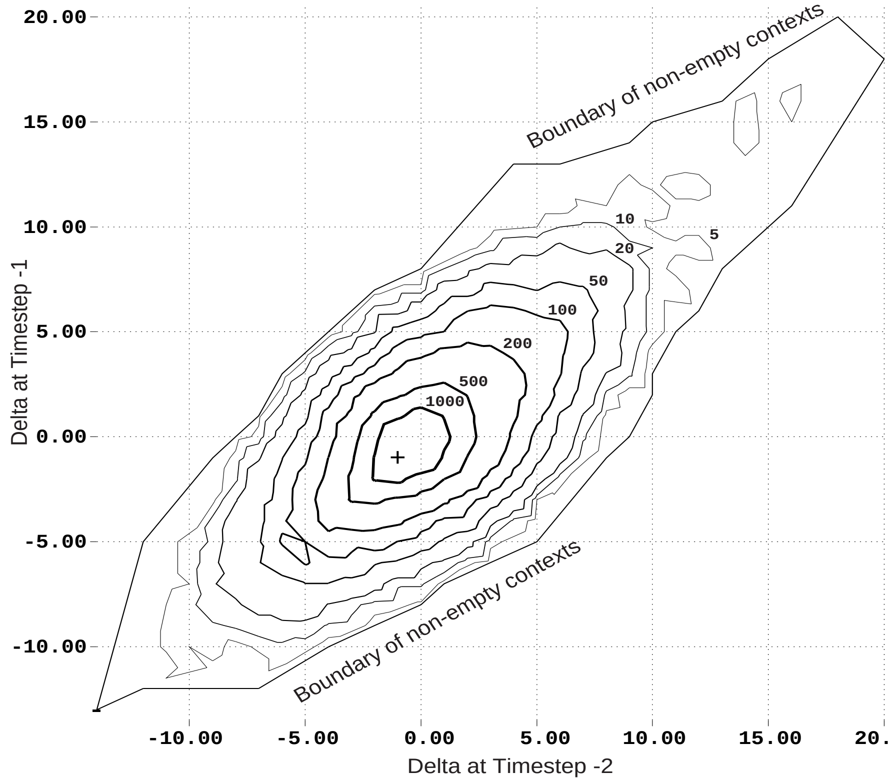


Figure 21: *Occurrence counts of order-2 contexts after application of filter (2,2)*

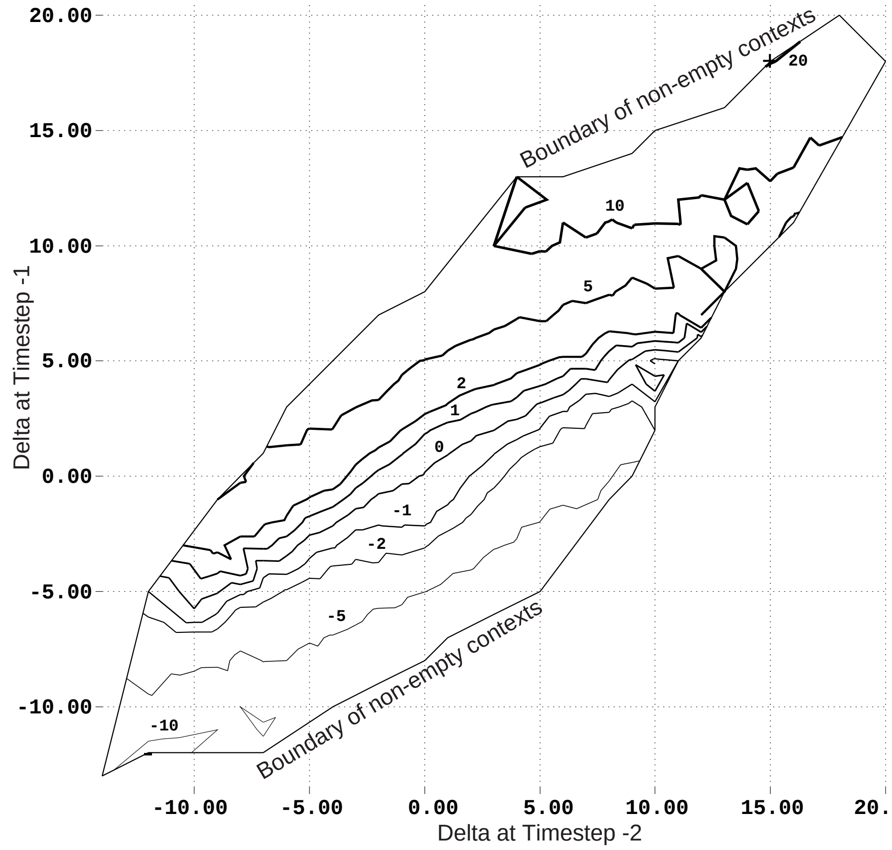


Figure 22*:Mean successor delta of order-2 contexts after application of filter (2,2)*

during the collection of the data to code each new value exactly according to the accumulated knowledge about its probability at the time when the parameter is arriving. To give an idea about the sort of information used for statistical coding, two pairs of figures, Figs.19 and 20 for the raw signal (Fig.13) and Figs. 21 and 22 for the same data after filtering (Fig.14) are shown. Figs.19 and 21 show, how many times successive combinations of delta values occur during the whole flight (the total number of data points is ≈42000). For the raw signal the picture is rather discouraging, showing only a marginal increase of counts around the origin and the boundary of nonzero contexts covering a large area. The contexts of the filtered signal are well centered and there are no outliers.

Figs. 20 and 22 show the average successor delta value after the occurrence of two preceding deltas. Whereas the picture for the raw signal is again not easy to read (contexts below the ≈140° meandering line through the origin are followed by positive successors on the average), the contour plot for the filtered signal gives a well structured picture (e.g. a pair of two deltas of 5 predicts an average next delta of 2). A coder can benefit from that type of information by adjusting its predictions accordingly.

There is an enormous literature on efficient data compression methods. Beating the compression efficiency of the best available general purpose coders requires very sophisticated tailoring with respect to a special data type. If the delta values for engine or flight data are limited to the 8-bit

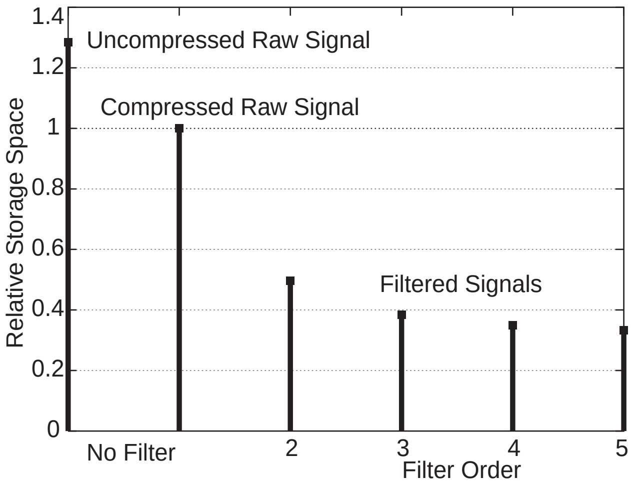


Figure 23: *Influence of filters on the compressibility of delta coded turbine blade signals*

range, arbitrary text coders [BCW90] can be used for compressing those data. Because a comparison of the performance of some of the best coders applied to delta coded engine signals showed only marginal differences in compression rates, the well documented and freely available "ARITH-N" program in [NG96] was used to determine the compression rates in the present investigation. Figure 23 gives some results for the TBT signal example, demonstrating the reduction of required storage space that can be accomplished without any adverse influence on the useful information content of the stored signal. A comparison was made with the memory occupied by the TBT signal on the new flight data recorder of the German Tornado aircraft [SS95], that uses a RLE method with an adjustable threshold for ignoring small parameter changes. The 42000 data in Fig.13 consume 67.6k byte of storage space, whereas the application of filter (5,5) plus statistical coding of the delta signal squeeze the data volume down to 10.7 k byte, a factor > 6.

A Simplified Model for LUM

To assess the fatigue life usage of rotating engine parts, a mathematical model has to be developed, that is simple enough to be either executed in an on-board LUM system or to process large amounts of stored flight data after they have been downloaded and stored. A short outline of the method is presented here. A more rigorous treatment is given in [Gra00]. During the design and development testing of a new or modified engine component finite element calculations of the thermal and mechanical behavior are performed to verify that the requirements on safety and durability are fulfilled. During this process some high stress areas are identified on the component, that are considered to be candidates for cracking. Because the stress at those areas also comprises thermal stresses a model for the temperature development within the component is also needed. This model calculates metal temperatures at selected points, some of them coinciding with the critically stressed areas. Fig. 24 shows a compressor disk with 6 temperature points, 4 of them critical areas.

Figure 25 outlines the most important steps of the calculation: The engine inlet temperature T_{in} and the spool speed N are used as input into the temperature calculation. Both parameters are used to determine the local gas temperatures, that influence heating or cooling of the component. Heat transfer between gas and metal is a nonlinear function of the engine operating point. There is a mutual influence of the temperatures in the component via heat conduction, indicated in Figure 26. The temperatures at one time point n are influenced by all other temperatures at the previous time point n-1. The updated temperatures and the spool speed are used to compute the stresses at the critical areas:

$$S(n) = a + b \cdot (N(n))^2 + \sum_{i=1}^{l} g_i T_i(n)$$

with a, b, g_i coefficients specific for each area.

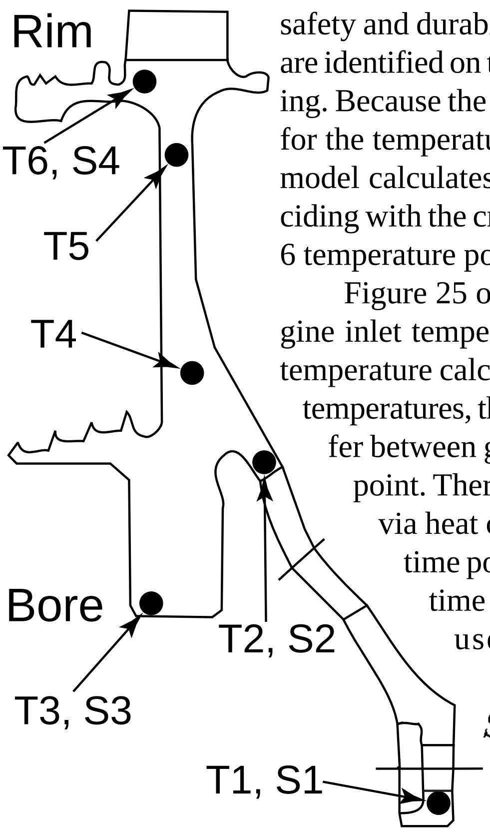


Figure 24: *Temperature and stress areas for compressor disk*

The process is similar for all critical areas. The metal temperature at the critical area is then used to compute a temperature dependent material strength (i.e. UTS(T)) which usually drops with increasing T. $S_{Norm}(n) = \frac{S(n)}{\mathrm{UTS}(T(n))}$ is then used as input into a cycle extraction process, which sorts the stress extremes into closed hysteresis loops. By using S_{Norm} instead of S the well known fact is considered, that for two equal stress cycles the one with higher T will contribute a higher damage. The cycle extraction produces cycles (n_1,n_2), (n_3,n_4), … Each cycle is then converted into an equivalent 0-max tension cycle with assumed equal damage. The well known Goodman correction is used:

$$S_{0-max}(n_{max}, n_{min}) = \frac{\mathrm{UTS}(T(n_{max}))\cdot(S(n_{max})-S(n_{min}))}{\mathrm{UTS}(T(n_{max}))-S(n_{min})}.$$

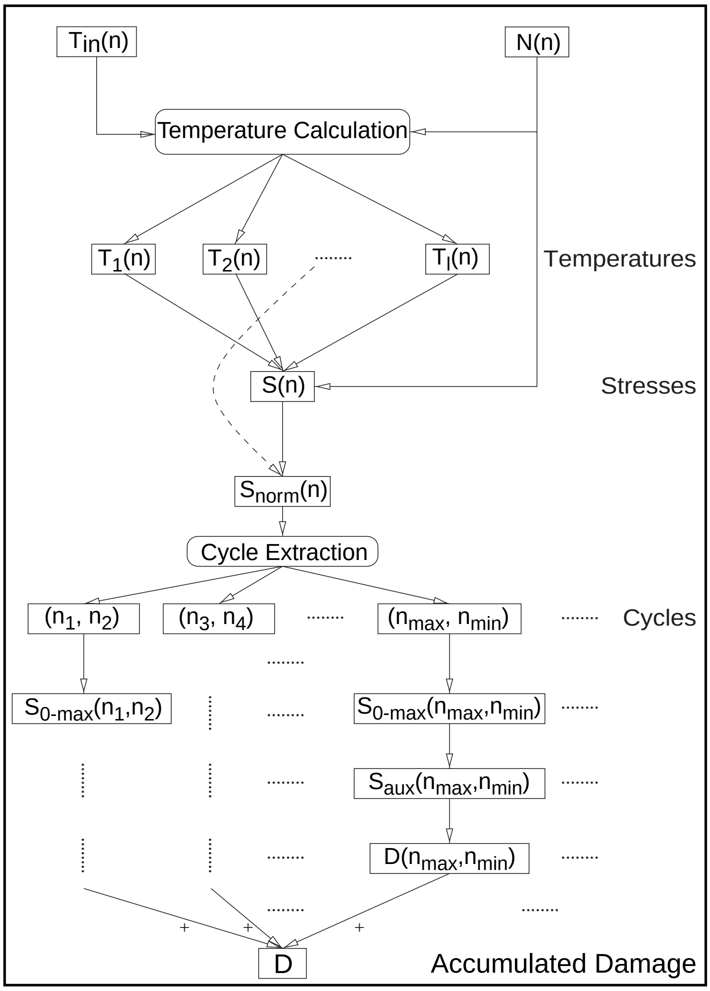


Figure 25: *Overview of cyclic fatigue life usage calculation*

The Goodman formula usually has to be amended by further corrections (e.g. precautions for the denominator becoming 0 or negative). Next the specific stress concentration factor k_t for the local geometry of the critical area and the material's infinite cyclic life threshold FCUT are used to calculate a damage parameter:

$$S_{aux}(n_{max}, n_{min}) = \frac{S_{0-max}(n_{max}, n_{min})\cdot k_t}{S_{cut}(n_{max})} - FCUT$$

S_{cut} is a temperature dependent threshold stress value. In the present investigation $S_{cut}(n_{max}) = FCUT \cdot \mathrm{UTS}\ (T(n_{max}))$ was used. To normalize all damages at a critical area to the damage of the largest stress cycle of the design mission, the Goodman-corrected $S_{0\text{-max}}$ of this cycle is used to calculate

$$S_{ref} = \frac{S_{0-max,design}\cdot k_t}{S_{cut}} - FCUT.$$

Here $S_{cut} = FCUT \cdot \mathrm{UTS}\ (T_{ref})$ is assumed, where T_{ref} is a characteristic high temperature at this area, usually that corresponding to the maximum stress of the design mission. If S_{aux} is calculated from $S_{0\text{-max}}$ of an arbitrary cycle, only positive values will contribute a damage increment:

$$D(n_{max}, n_{min}) = \left(\frac{S_{aux}(n_{max}, n_{min})}{S_{ref}}\right)^{ESN}$$

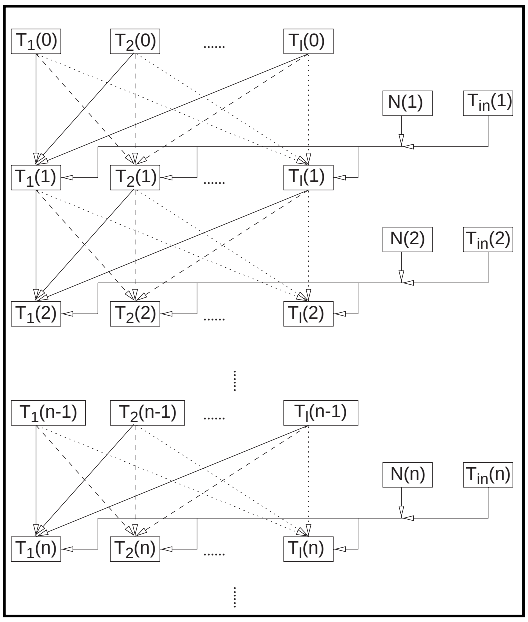


Figure 26: *Calculation of metal temperatures*

Other cycles are said to fall below the fatigue cutoff. By virtue of the definition of S_{ref}, D=1 will be computed for the largest cycle of the design mission. Life releases for a particular component type of an engine usually are expressed as multiples of this value.

ESN is the exponent in the so-called S/N curve, which expresses the number of survived stress cycles as a function of the cycle's magnitude for a certain material. This parameter determines, how fast the damage grows with increasing stress level. For large values of ESN even small changes in the computed stresses will have a considerable influence on the computed damage results. This parameter plays a decisive role in the determination of the required accuracy for the storage of the spool speed signals.

This is easily seen for a critical area with low thermal stresses, where the total stress varies $\sim N^2$. If additionally no fatigue cutoff would be present, the damage would vary $\sim N^{2 \cdot ESN}$. Since ESN may assume values as high as 4, a small change in the spool speed signal might be amplified by a factor of 8 in the damage calculation. If fatigue cutoff is present, this effect becomes even worse. For the life limiting critical area of the compressor disk in Fig.24 (Area 2), which has ESN=2.5, the presence of fatigue cutoff would amplify a spool speed increment of 1% at design conditions into an increase in damage of 6.9%, assuming that the total stress is only dependent on N^2.

Another effect immediately influencing the sensitivity of calculated damage to small variations of the spool speed signal is the ratio of thermal stresses and centrifugal stresses in the stress law. Stresses at areas with a high portion of thermal stresses tend to react to spool speed changes with some delay thus de-coupling the highest stress events from “natural” occurrences of the highest spool speeds (e.g. at the start of the aircraft).

Influences of Filtering

As preliminary tests with various types of flight data indicated some potential benefit of applying lowpass filters before trying to store the data, and no generally applicable rule was found, how to select the filters and their parameters, a decision was made to perform some systematic tests with the maybe most important single parameter entering the LUM calculation, the spool speed signal of the HP spool. An available calculation model for the compressor disk shown in Fig. 24 was used, because of its representative nature and its relative simplicity involving various types of critical areas.

Because of the plan, to track the propagation of data modifications throughout the whole calculation process outlined in the previous chapter, it was necessary to limit the number of flights to 24, also due to limitations in available data storage capacity for a student’s project. The data were carefully selected from an existing pool of recorded flights from all 3 owner nations of the Tornado aircraft.

Investigation Method

The data from both engines in an aircraft were used. An in-house program of the MTU stress department for the life consumption analysis of rotating components (Mission Analysis Program) was used to compute all relevant data with the two input parameters T_{in} and N taken from the flight data recordings. Temperatures and stresses and the normalized stresses were computed and stored for every time point in the flights for all areas indicated in Fig.24. All cycles found by the cycle extraction, together with the damage at each area were also stored . The results of the unfiltered data were used as a reference. The same computation was repeated with the spool speed signal replaced by the delay-corrected output of each of the 25 lowpass filters. The inlet temperature was not filtered. 1200 sets of files (24 recorded flights · 2 engines · 25 filters) were stored to enable a statistical analysis of the deviations between the reference using the raw data and the results with the filtered data. The differences between computed temperatures, stresses, normalized stresses and damages with respect to the values based on the original flight data were computed. Histograms of the deviations were used to get an idea about the resulting distributions. Basic statistical parameters, as expected values, medians, variances, ranges and quantiles were computed for all difference data. A detailed description and comprehensive results are given in [Gra00].

The original and filtered spool speed data were converted into delta values and were fed into the „ARITH-N“ file compression program [NG96] with model orders set to 1,2,3. The resulting output sizes were recorded. Note the difference between filter order and coding order (the maximum number of predecessor values used in the coder’s internal statistics tables). Summarizing the coding results, it turned out, that coding with order-1 prediction was always optimal for the raw spool speed values and in most cases also for the data with filters (1,1),(1,2) and sometimes (2,1). For all other filtered data order-2 coding resulted in best compression, whereas order-3 was nearly never better than order-2, with single exceptions for filter (5,5).

The median value of storage space reduction by the statistical arithmetic coder was 3.7 for the raw N data relative to the storage of uncompressed 8 bit delta values. This means, that the average data volume for the 2Hz spool speed signal for 1 hour of flight time was 1944 bytes. These data permit a lossless reconstruction of the original signal, which had a 11 bit accuracy. The additional gain, that can be reached by filtering the data is shown in Fig. 27. In contrast to the previously shown results for the noisy TBT signal (Fig.23), the compression gain is now only a function of the cutoff frequency and nearly independent of the filter order with the exception of order 1 filters, which behave different due the lack of overshoot in their step response. The 4 filters with cutoff frequency > 0.2 even cause a deterioration of compression rates due to their tendency to amplify overshoots in the signal.

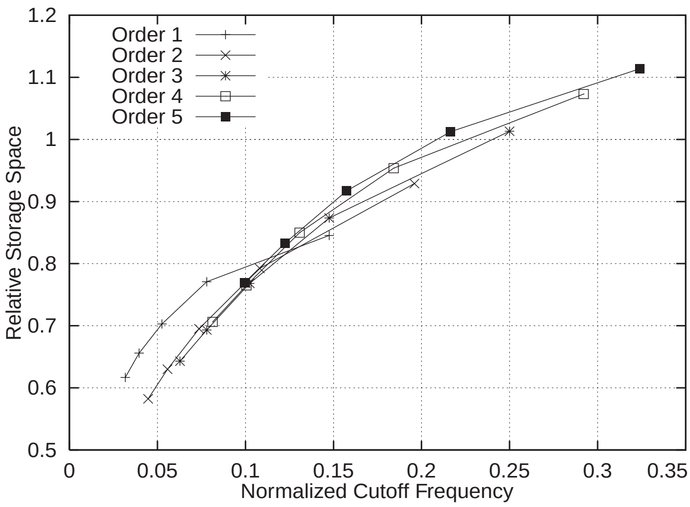


Figure 27: *Size reduction factors for filtered N signals*

Accuracy Loss of Computed LUM Results

Omitting a detailed discussion about the various effects of the input signal filtering on temperature and stress development, a summary of the effects on the final result, i.e. the computed fatigue life consumption

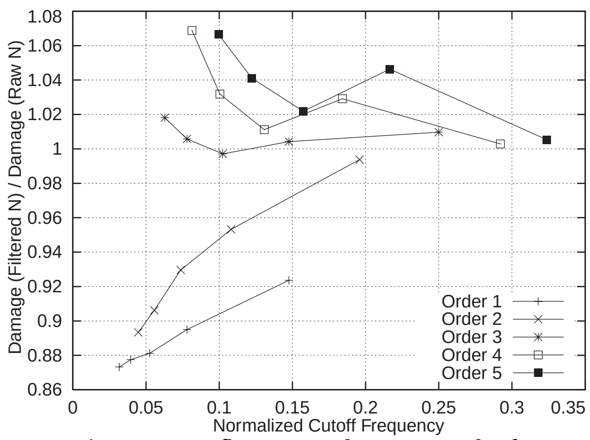


Figure 28: *Influence on damage results for area 1*

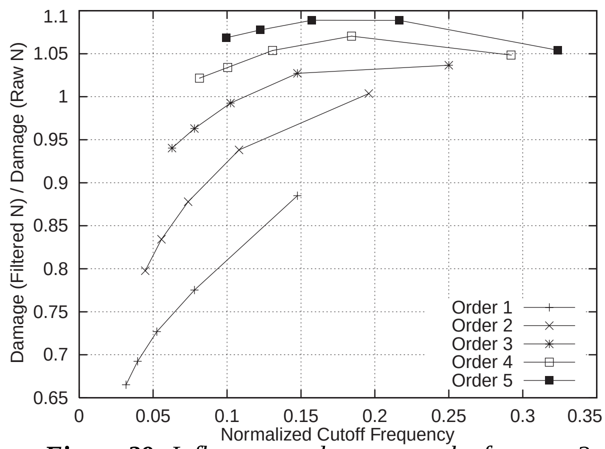


Figure 29: *Influence on damage results for area 2*

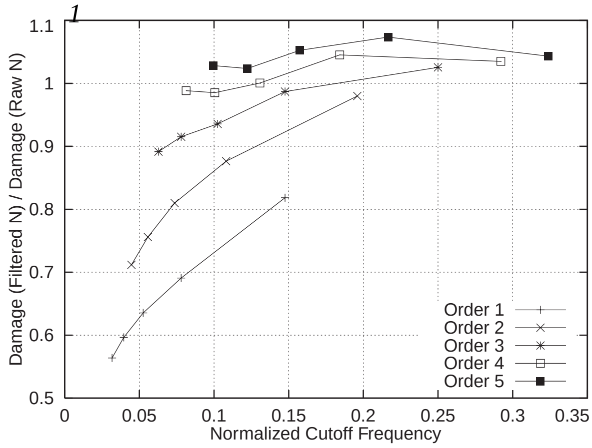


Figure 30: *Influence on damage results for area 3*

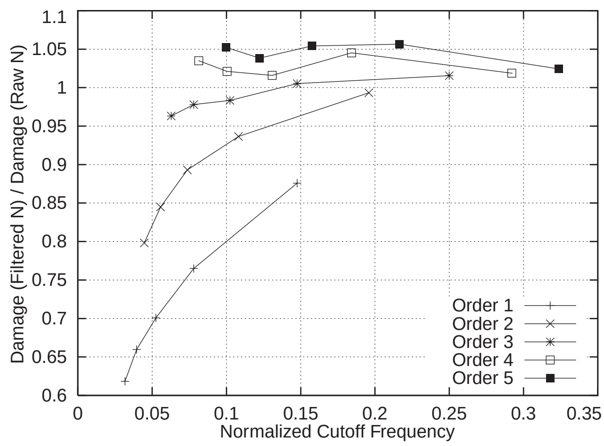


Figure 31: *Influence on damage results for area 4*

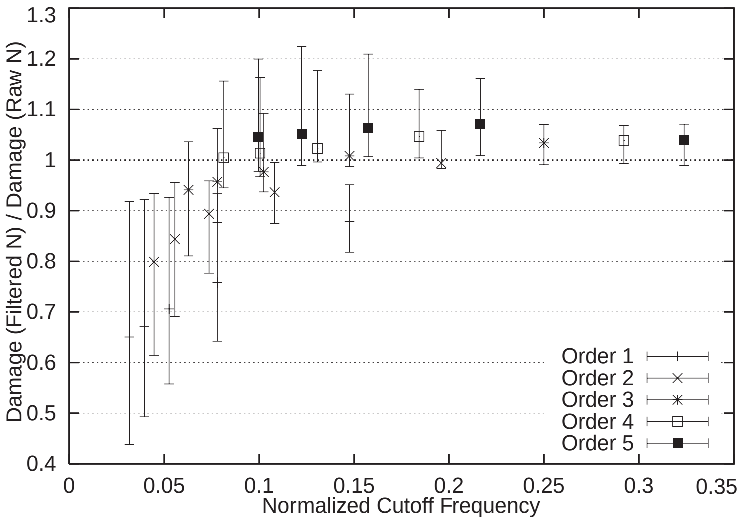


Figure 32: *Scatter bands of computed damage for area 2*

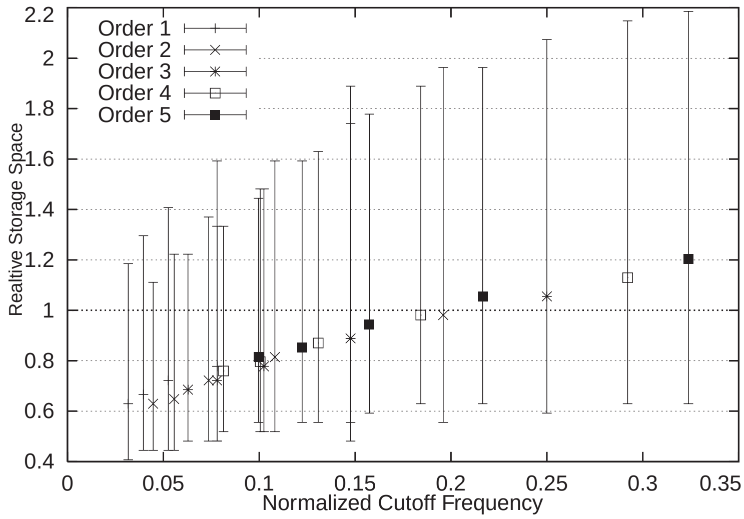


Figure 33: *Scatter bands of storage requirements after application of filter and compression*

at the critical areas is now given. The results are shown in Figures 28 - 31, each corresponding to one critical area. Some common features are easily visible: The application of order 1 filters always leads to a severe underestimation of computed damage results. Most of the order 2 filters also produced results with too low damage. Filter (2,1) performed remarkably well with damage ratios close to 1.0 for all areas. The damage ratios of the 5 filters with order 3 were grouped around the desired value 1.0 with best results for the filters (3,2) and (3,3). The filters with order 4 and 5 generally yielded an overestimation of damage results. For area 3, the filters (4,3), (4,4), (4,5) and (3,2) yielded better results than all other filters.

These results are average results over all investigated flights. They do not preclude much higher deviations of the damage results for a particular flight. Figure 32 shows the scatter bands of the damage results at area 2 for all combinations of flights and filters. Since there was at least one result, that was considered as an outlier due to problems in the recording process, it was decided to omit one result from either end of the scatter band in the plot. The scatter is lowest for filters with high cutoff frequencies, because their output tends to follow closely the (physically correct) overshoots of the spool speed signal. With decreasing filter cutoff there is not only an average under-prediction of damage, but also an increase in the scatter. One remarkable effect is the clustering of scatter bandwidths for filters with equal group delay, i.e. the results for filters (5,1), (4,1), (3,1), …, form one group, the filters (5,2), (4,2), … the next one, and so on.

Figure 33 shows the ranges of required storage space. Again one result from each the lower and upper end of the scatter band has been omitted from the plot for reasons of consistency. The symbols inside the range bars are at the same locations and the same scaling is used as in Fig.27, that means the storage space values are divided by the median of the storage space required by the compressed raw signals. The relative scatter varies only slightly with a minor increase towards the „less-intrusively“ filtered signals (filters (5,1),(4,1),..with cutoff frequencies > 0.2). As a rule of thumb, a factor of 2 may be assumed for the maximum deviation of the storage space from the median for a single flight for an arbitrary choice of filters. The largest scatter [0.52,2.26] occurs, as expected, for the raw signals, whose scatter band is thus exceeding slightly the worst case range of the filtered data. The variability in compression ratio would have to be taken into account for the design of a recording system, because it determines the extra memory to be allocated to avoid memory overflows and corresponding data loss, if the readouts are to be performed after a fixed number of flights or after a fixed engine operating time.

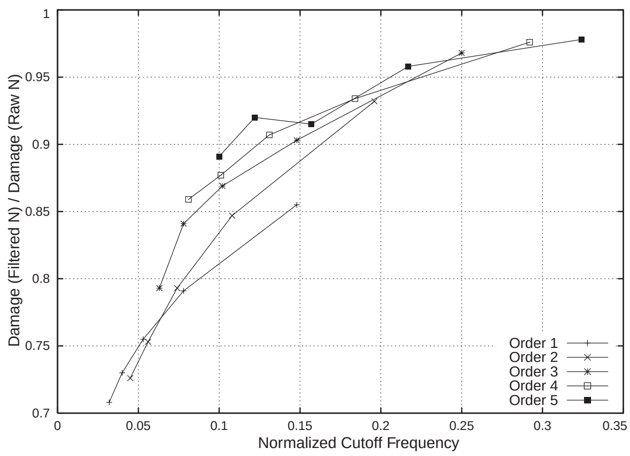


Figure 34: *Influence of filter application on computed damage for critical area in HP compressor*

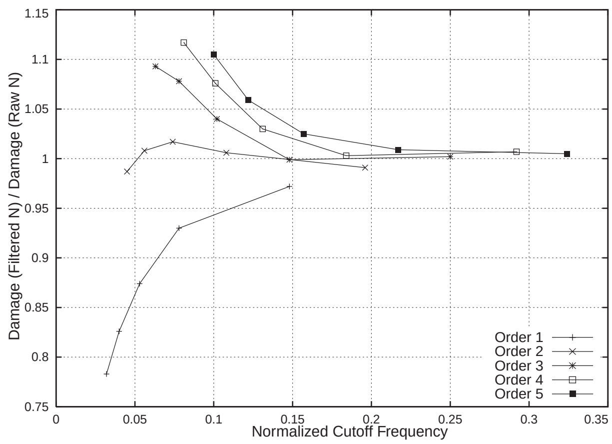


Figure 35: *Influence of filter application on computed damage for critical area in HP turbine*

The dependence of accuracy on filter parameters shown in Figures 28-31 does not allow to derive a simple strategy for the filter selection. Some selected results of an independent study with the same filter family applied to the same signal type (HP spool speed), but only to a single flight are shown in figures 34 and 35. In this study all 34 areas influenced by the HP spool speed of the RB199 were included and checked for deviation from the damage results calculated with the original signals. Some of the obtained dependencies were found to be very similar to those shown in the figures 28-31, but also new types of dependencies occurred. In Fig.34 the damage results almost entirely depended on filter cutoff frequency, with only weak influence of filter order, whereas Fig. 35 shows a completely different behavior, with an increased over-prediction of damage with decreasing cutoff frequency for all filter orders > 1.

Conclusions

The influence of filter application on the results of a fatigue life usage calculation is strongly dependent on the type and the noise content of its input signals. For temperature signals and other signals contributing to damage only via integrating algorithms, suitable filter parameters can be derived from a spectral analysis of the signal's autocorrelation function. Filter cutoff frequencies can be selected to remove all noise components found in the spectrum, leaving a smoothed filter output with great potential for efficient data compression.

Properly acquired spool speed signals have a very low noise content. Even small changes to this signal may have an immediate influence on the computed life consumption results. The dependence of the results on the filter parameters is very complex and it is neither possible to provide a generally applicable law nor to predict the magnitude of the deviations, if more than one critical area or even different engine components are affected.

Only modest additional gains in data compression rates can be accomplished by filtering signals with low noise content, unless considerable information loss is accepted. By applying delta coding together with available statistical data compression methods impressive compression rates can be achieved already for the original signals. Therefore it is recommended to avoid filter application to spool speed signals.

Using appropriate filtering and compression methods to each signal type would result in very low storage requirements. To give an order of magnitude: 10k bytes of memory will be sufficient to store one hour of engine operation without any adverse influence on the accuracy. A recording system entirely dedicated to the data needed for engine LUM calculation could overcome some of the disadvantages of existing on-board engine monitoring systems. It could be integrated as a separate task in an existing monitoring or engine control system.

References

[BCW90] T.C.Bell, J.G.Cleary, I.H.Witten: Text Compression.
Prentice Hall, Englewood Cliffs, N.J., 1990
[Ber94] J.Beran: Statistics for Long-Memory Processes. Chapman & Hall, New York 1994
[BP97] J.Broede, H.Pfoertner: OLMOS in GAF MRCA Tornado - 10 Years of Experience with On-Board Life Usage Monitoring.
33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Seattle 1997
[Bra00] B.Bradford, S.Rose, B.Bloomfield, R.Greenwood: Engine Review. FY99 Engine-Related Mishaps. United States Air Force Flying Safety Magazine, January/February 2000
[Gra00] S.Graf: Filterung und Datenkompression von Flugdaten zur Unterstuetzung der Zustands- und Lebensdauerueberwachung von Triebwerken. Diplomarbeit im Fachgebiet Mathematik, Universitaet Passau, Fakultaet fuer Mathematik und Informatik, 2000
[Gro99] D.R.Grossi: Aviation Recorder Overview. International Symposium On Transportation Recorders, Arlington, Virginia, May 3-5, 1999
[HK99] R.A'Harrah, G.Kaseote: A Case for Higher Data Rates. International Symposium On Transportation Recorders, Arlington, Virginia, May 3-5, 1999
[NG96] M.Nelson, J.-L.Gailly: The Data Compression Book, 2nd Edition.
M&T Books, New York, 1996
[PB87] T.W.Parks, C.S.Burrus: Digital Filter Design. John Wiley & Sons, New York, 1987
[PR95] H. Pfoertner, C.Ross: Preparing Life Usage Monitoring for the Next Decade.
18th International Symposium AIMS, Stuttgart 1995
[OS89] A.V.Oppenheim, R.W.Schafer: Discrete-Time Signal Processing.
Prentice Hall, Englewood Cliffs, N.J., 1989
[SS95] A.Schick, U.Schulz: Technical Logistic Support with Modern Solid State Flight Data Recording System, 18th International Symposium AIMS, Stuttgart 1995
[SM89] M.E.Sigrist, A.N.Muggli, SWISSAIR Studies on the Compressibility of Aircraft Data.
15th Symposium AIMS, Aachen 1989
[Smi97] S.W.Smith: The Scientist and Engineer's Guide to Digital Signal Processing.
California Technical Publishing, 1997
[SD93] S.D.Stearns, R.A.David: Signal Processing Algorithms using Fortran and C.
Prentice Hall, Englewood Cliffs, N.J., 1993

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List of Acronyms

ESN …	Exponent in S/N Curve
FCUT…	Fatigue Cutoff
FDR …	Flight Data Recorder
HP …	High Pressure
LUM …	Fatigue Life Usage Monitoring
MAP …	Mission Analysis Program
n …	Number of Time Point
N …	Rotational Spool Speed
OLMOS	On-board Life Monitoring System
RLE …	Run Length Encoding
TBT …	Turbine Blade Temperature
UTS …	Ultimate Tensile Strength

MONITORING OF CRITICAL INLET AIR CONDITIONS

FOR HELICOPTER GAS TURBINES

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ABSTRACT

Turbo-shaft engines of helicopters are often subjected to harsh environmental conditions. Of inherent importance for a safe engine operation is the quality of the airflow entering the gas turbine. This paper briefly discusses the different forms of critical inlet air conditions and takes a closer look on the ingestion of contaminated air. In an experimental setup the injection of carbon dioxide gas into the inlet of a test bed engine is realized. Methods to detect such situations are developed and requirements for control systems are derived.

1. INTRODUCTION

The ability of helicopters to operate independently of airfields and fixed based infrastructure strongly influences their use. Unlike fixed-wing aircraft, which mainly fly point to point for passenger or cargo transportation, helicopters are mostly used for special assignments. These missions, either military or civil, often demand flights into remote or confined areas and frequently occur under marginal meteorological conditions or during night time. Typical missions include:

- Air rescue, air ambulance
- Aerial fire fighting
- Military operations, including the launch of missiles

The flight profile of these missions is dominated by low altitudes. The combination of low altitude and the helicopter's low lift-to-drag ratio are key factors for relatively high accident rates. The fatal accident rate in Germany was 1.3 accidents per 100.000 flight hours for helicopters compared to 0.17 accidents for fixed-wing aircraft (MTOW >5.7 t) [1]. A study showed that most of the fatal accidents were weather related (28.2%) closely followed by mechanical causes (25.9%). Among the mechanical failures, engine related loss of power was the most frequent accident cause (31.8%) [2]. One reason for the high number of engine failures is the harsh environment the engines are operated in. Unlike jet engines of commercial transport aircraft that usually operate at constant power in high altitudes, the helicopter's turbo-shaft power-plants experience frequent changes in power setting and sometimes distorted inlet conditions. The inlet distortions include water or snow intake, e.g. when flying in heavy rain, hovering over lakes to take on water for fire fighting missions or during marine air rescue missions, inlet pressure distortions from abrupt flight maneuvers or due to the own rotor down-wash. A possible inlet anomaly is also the intake of polluted air, which typically is some sort of exhaust gas from a combustion process. The intake gas then represents a combination of high temperature and low partial oxygen pressure, both of which have a negative effect on engine performance. Conditions like these can be found during aerial fire fighting flights, when flying or

hovering over industrial plants and possibly shortly after a missile is launched. For example, some years ago, a helicopter suffered an engine flame out when hovering over a factory chimney that inadvertently was not shut down. Unfortunately the combination of zero forward speed, low altitude and external load did not allow a safe autorotational landing and the helicopter crashed into factory buildings [3].
The following chapters describe the effort to simulate the intake of contaminated air and develop methods to reliably detect and counteract such situations.

2. CLASSIFICATION OF INLET CONDITIONS

The quality of the air flow condition at the engine's inlet is of inherent importance for a safe and reliable operation of gas turbine engines. Unfortunately, engine installation in helicopter airframes often is a compromise of many disciplines and thus is less favorable for the engine compared to turbofan engines installed in commercial aircraft. In addition to the main function of an inlet design, providing unrestricted and turbulence-free air to the compressor, inlets for helicopters often include special centrifugal extractors to prevent foreign object damage to the rotating parts. Three major intake disturbances can be identified.

2.1 Pressure Distortion

Resulting from the design of the intake and the integration of the engine into the airframe, a distorted pressure distribution at the compressor intake plain is experienced.

$$DC_\varphi = \frac{\overline{p}_t - \overline{p}_{t,\min,\varphi}}{\overline{q}} \tag{2.1}$$

The distortion coefficient DC_φ describes the total pressure distortion in a φ-deg sector over the compressor circumference. A strong distortion not only reduces the available engine power, but also invokes a shift in the compressor surge line and increases the danger of compressor stalls [4].

2.2 Temperature Distortion

With increasing intake air temperature, the thermodynamic efficiency of the engine worsens and thus the available engine power, limited by hot section temperatures, decreases. Furthermore, similar to the effect of pressure distortion (2.1), an uneven temperature distribution at the intake adversely affects the compressor's efficiency.

2.3 Contamination

A critical form of inlet disturbance is the intake of gases with low oxygen content. The engine not only immediately looses power, but is also in the danger of suffering a flame out. The occurrences of such intake conditions are, for example, when due to maneuvering the engine's exhaust gases are recirculated. Other situations can be found e.g. when the helicopter is flying in the vicinity of active chimneys or closely over open wildfires, typically during aerial fire fighting missions. The combination of low airspeed, low altitude and high pilot workload (e.g. maneuvering with external loads) increases the probability of a hard forced landing in case of an engine flame out.
A second form of inlet disturbance is the intake of water. While the ingestion of smaller amounts of water improves the engine's efficiency and increases power output due to improved thermodynamic cycle efficiency and higher mass flows, from a certain amount of water on, the danger of an engine flame out exists. If the water content increases excessively, not all of the water can evaporate during the

compression and thus threatens to extinguish the combustion. In recent experiments at the Department of Flight Propulsion it was shown, that excessive water intake could be detected with a supplemental temperature probe located at the outlet of the compressor. With the countermeasure of fully opening the compressor's inter-stage bleed valve and engaging the ignition, an engine flame out could be avoided over a vastly enlarged operating range [5].
If the ingested water is in the state of aggregation of ice, there is a potential danger of mechanically damaging the rotating parts of the first compressor stages. Larger particles of ice can appear if de-icing devices are activated too late. Ice that already has built up at structures (e.g. inlet or leading edges) breaks away after the inlet de-icing is engaged and is ingested into the engine. It is necessary to detect icing flight conditions and automatically operate the de-icing equipment in time.

3. TEST FACILITIES

3.1 Test Bed Engine

Among the facilities operated by the Department of Flight Propulsion at the Technical University of Munich are two test cells for turbo-shaft engines of up to 1000 kW shaft power. While one cell is awaiting the installation of a MTR390 engine, the other one is equipped with an Allison 250 engine. This engine, which was donated by the German Armed Forces, has been used in research work for several years. Although the basic design is more than 30 years old, the engine is still in production at Rolls-Royce Allison for several applications, mainly light helicopters. The department's engine is a series C20B model that was installed in an Army BO105 helicopter.
Despite its age, the engine remains a valuable test bed for any kind of research work. The engine's unconventional design allows easy access to most parts, not only for inspection but also for modification and installation of supplemental sensors.
The engine, which is rated at 300 kW continuous power, uses a combined axial/radial flow compressor. While the first 6 stages are axial flow stages, the 7^{th} and final stage is a centrifugal one. The design of the engine provides access to each of the axial stages for the installation of sensors with only minor modifications to the compressor casing. The compressed air is led by two ducts into a reversing combustion chamber where the fuel is injected. The uncooled 2-stage high pressure turbine drives the compressor directly and the auxiliary systems via a gearbox. A 2-stage axial flow power turbine expands the exhaust gas to ambient pressure. The power turbine drives with an intermediate gearbox the engine's output shaft. Its rotational speed is around 6000 RPM.

3.2 Fuel Metering Unit

The production engine is equipped with a hydro-mechanical fuel flow controller and a mechanically driven fuel pump. Additionally, an electronically controlled fuel metering unit was integrated into the test bed engine. Supplied by an electric fuel pump, a metering valve is commanded by an experimental FADEC system [6]. An assembly of electromagnetic valves allows the transition between the two supply systems as the operator demands. With this setup the original controller acts as a safety backup to the digital system.

3.3 ECB Control

The eddy current brake (ECB) used to dissipate the generated energy is controlled by a digital control system that ensures a quick response time of the brake and thus allows a simulation of load profiles as they appear in typical helicopter missions.

3.4 Standard Engine Controller

The standard engine controller is a hydro-mechanical Bendix type. It comprises start and output shaft speed control, but includes no other limitation or protection functions except for overspeed. An altitude compensation function is included. In combination with a scheduled bleed valve, a surge-free acceleration is provided, however at the cost of reduced economics and engine dynamics.

3.5 Bleed Air System

A major modification to the engine was the fitting of an electronically operated bleed air valve. While the production engine is equipped with a pneumatically engaged valve allowing only a scheduled operation in dependence of the compressor outlet pressure, the electronic design allows an independent control of the bleed air flow. Because of its fast response time, the valve can be used to terminate the onset of engine surge [4]. The valve consists of a cylindrical body that is rotated by a servo motor. A LVDT is used for a position feedback, so that an accurate movement is granted under all operating conditions.

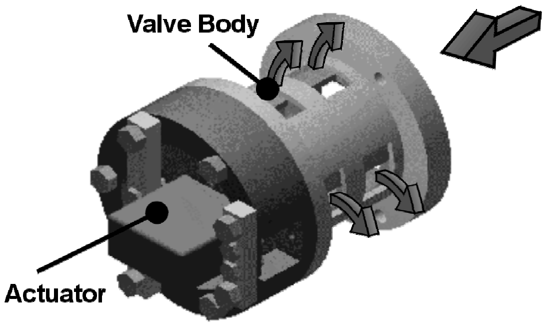


FIGURE 3.1: ELECTRONIC BLEED VALVE

3.6 Experimental FADEC

A fully operational full authority digital engine control (FADEC) was developed [6,7] and successfully tested. It allows safe engine operation over the entire operating range and significant performance improvements compared to the hydro-mechanical system are gained. It features non-standard temperature and pressure compensation. Also included are various protection and limitation functions to ensure engine integrity and to reduce pilot workload. The monitored parameters include gas generator rotational speed (N_1), high pressure turbine exit gas temperature (TOT) and fuel-air ratio. A special detection logic is included to recognize the onset of a compressor stall [4,5]. The developed algorithm detects stalls after a very short time and a possible engine surge is at worst terminated after one surge cycle by a significant control input of the FADEC. The measures taken are mainly opening of the bleed valve and a reduction of the fuel flow.

Furthermore, ignition and starter engagement are FADEC controlled so starting and transition from ground idle to a flight condition (output shaft speed control) is done fully automatic as selected by the pilot/operator by the means of pressing a button.

3.7 Special Control Algorithms

A number of special control algorithms are included into the digital control system. The algorithms offer supplemental functions for test and measurement purposes, e.g. engine operation at constant corrected gas generator speeds for steady state measurements, fixed fuel flow profiles, indirect operation by influencing the hydro-mechanical controller levers, etc. [7].

4. Integrated Software Environment

The concept of the test cell is based on an integrated software and hardware environment to minimize the effort in handling and analysis of the test data. To achieve this goal a number of basic tools and operating systems were chosen:

- MATLAB as standard tool for data analysis and control software development
- LabVIEW for graphical online display of data
- Windows NT network
- VME Bus hardware

4.1 Data Acquisition

The installed test bed engine is equipped with a large number of sensors, far more than used in production engines. Mainly temperature and pressure probes are located throughout the entire engine, so that a good knowledge of all components can be gathered.

Data acquisition is realized by two independent systems based on VME Bus hardware:

- High resolution AD-boards with 96 channels at 16 bits and 35 µs conversion time are used to provide high accuracy data for steady state measurements and continuous monitoring of all signals.
- The high speed data acquisition system is used to record transient operations at very short sample times with a 12 bit resolution and a conversion time of 4 µs. Immediately after the measurement the data is available in MATLAB format for viewing or further analysis.

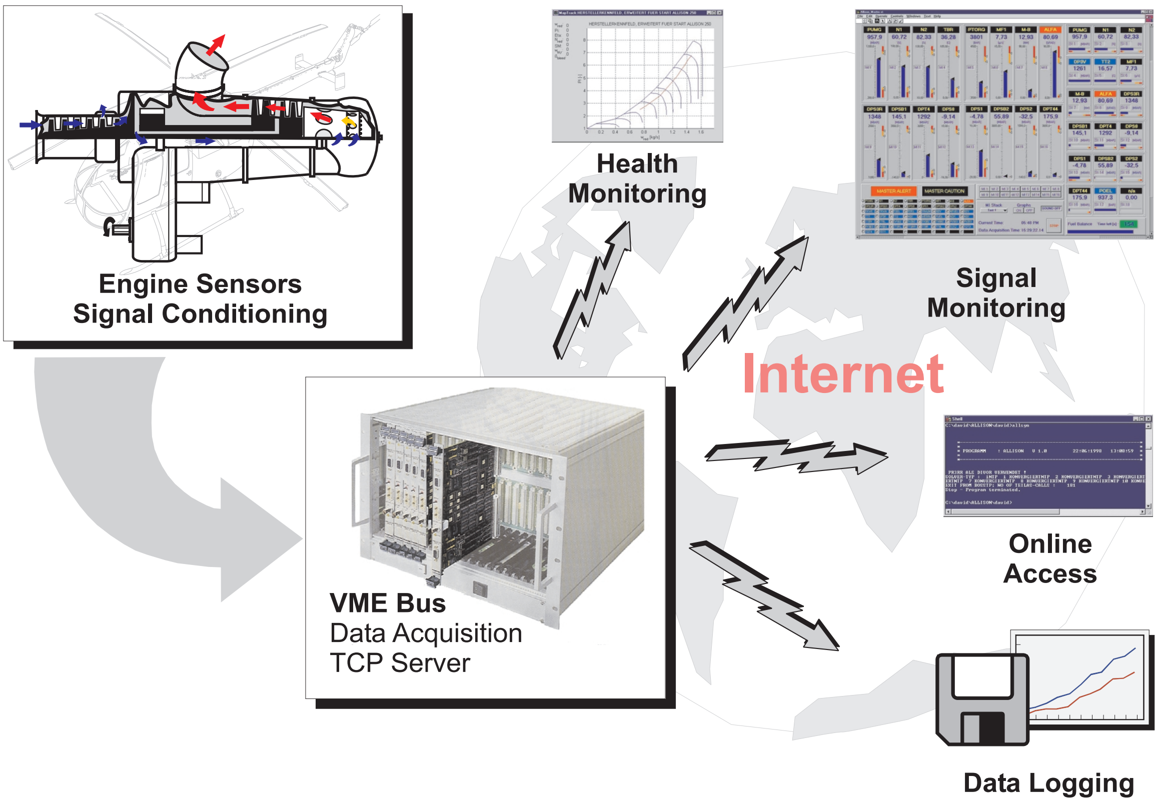


FIGURE 4.1: THE DATA ACQUISITION SYSTEM

Whereas the dynamics data is gathered upon a user command in the host controller's memory and then streamed directly to disk, the data of the high resolution acquisition system is sampled continuously. If a steady state measurement is requested, the data is collected over a specified period of time (e.g. 60s), then averaged and stored to disk. Additionally to the capability of running measurements, the sampled data is constantly held in memory and sent to clients on request by an Ethernet link. The online access via internet allows the setup of numerous distributed clients that run demanding applications without consuming CPU time of the data acquisition system. Standard interfaces are realized for MATLAB, LabVIEW and C/C++ use, and the actual signals are available on all workstations or PCs that have internet access. Among the tools that use internet data during experiments are (Figure 4.1):

- A data logging program that records all available signals at low sampling rates (e.g. 1 Hz) and stores them to disk. This data can be easily accessed in MATLAB or EXCEL for a later documentation of the test run (peak temperatures, speeds etc.).
- An extensive monitoring screen displays a selection of signals to the test engineer. The display can be configured during the test run. Additionally all signals are checked against the exceeding of specified limits. For each signal four values that represent upper and lower limits, can be stored in a central database. The thresholds are divided into warnings and alerts which are presented optically and acoustically.
- A MATLAB routine running on a separate PC workstation constantly computes the thermodynamic cycle based on the actual measurement. With this information an online display of e.g. component efficiencies is possible. The online tracking of the actual operating point in the component maps allows a fast and reliable visual inspection of the engine's health.

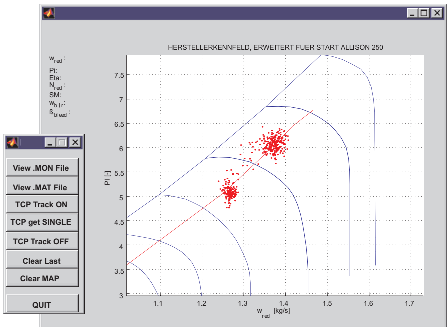


FIGURE 4.2: ONLINE COMPONENT MAPS

4.2 Full Authority Digital Engine Control (FADEC)

The FADEC Software is developed in MATLAB using the graphical programming extension Simulink. With this tool, based on the rapid prototyping philosophy, it is possible to create real-time applications without the need of programming C-Code. The C-Code is automatically generated by Simulink and with the use of a cross compiler downloaded to the real-time hardware. The hardware consists of a dSPACE multiprocessor system equipped with a DEC alpha CPU and a TI C40 DSP. The high computing performance, especially of the alpha CPU, allows the execution of sophisticated control algorithms using extremely short sample times, e.g. tests for surge detection and recovery were run at 6kHz controller sample rate [4]. With the enormous advances in computer technology comparable flight approved on-board hardware should become available within few years.

Along with the graphical programming, several tools are provided to control the application in real-time. A graphical user interface (GUI) allows online access to all FADEC parameters and visualization of the controller's inputs- and outputs and serves as operator control panel. A separate data acquisition system allows the recording of internal parameters of the control algorithms. This data can be used for off-line simulation in the development process of new controllers.

5. Experimental Setup: Carbon Dioxide Injection

To investigate the effects of contaminated air, a practical setup was developed to simulate the ingestion of gases with low oxygen content. In order to partly displace the intake air with inert gas, an assembly fed by a high-pressure CO_2 bottle was installed. CO_2 was chosen because it is the residual of most combustion processes, furthermore, it is inexpensively available and easy to handle. The special construction of the bottles allows the extraction of CO_2 either gaseous or in the liquid state.

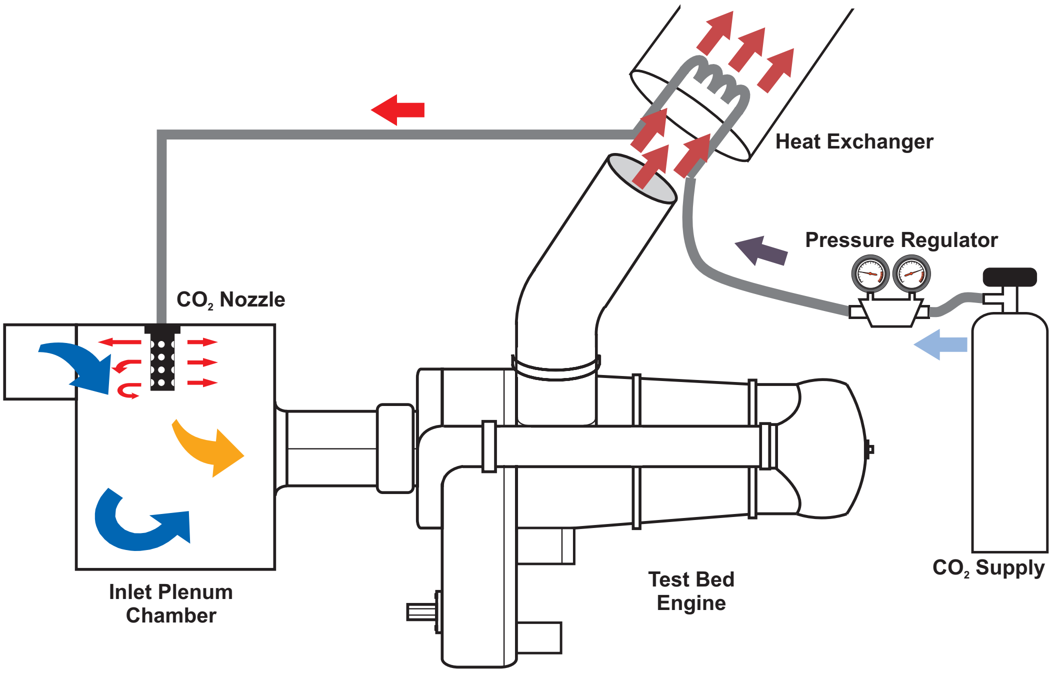


Figure 5.1: CO_2 Injection into the test bed engine

With the help of a pressure drop regulator the injected mass flow can be controlled at a constant level in the gaseous state. If liquid CO_2 is injected, the mass flow is limited by the orifice area of the spray nozzle. The spray nozzle is installed in a plenum chamber located upstream the compressor intake and the flow is radially injected into the main air mass flow, so that a good mixing of the gases is achieved.
However, the injected gas has a very low temperature due to the strong expansion. The colder intake mass flow enhances engine performance and partially offsets the negative effect of the reduced oxygen content. To overcome this problem and create a more realistic simulation environment - ingested exhaust gases usually are at higher temperatures than the ambient air - a simple heat exchanger was inserted into the CO_2 supply line between the pressure regulator and the nozzle. It is located downstream of the engine's exhaust ducts, inside the test cell's chimney. Due to the high exhaust gas temperature of the engine, a rise in the CO_2 temperature is achieved and the injected gas is significantly warmer than the intake air.
The developed setup allows the simulation of contaminated intake air with remaining oxygen contents from 21 vol.-% (standard) down to approximately 10 % at high temperatures for different engine power settings (i.e. intake mass flows). A further reduction can be realized if the pressure drop regulator is by-passed and a spray nozzle with a larger orifice diameter is installed. With this setup the mass flow cannot be controlled exactly due to the pressure drop inside the CO_2 supply bottle. However, test results showed, that the engine will immediately flame out and a relight cannot be achieved under such low O_2 conditions.

6. EXPERIMENTS

Several experiments were conducted to study the effects of contaminated air on engine performance and operability. The aim was to collect data for a detailed thermodynamic analysis and as a basis for developing algorithms to detect and distinguish such intake conditions. The tests comprised injection of CO_2 at various power settings (idle to max. power) including flame outs.

6.1 Flame Out

If the oxygen content is reduced too far, the combustion chamber will immediately suffer a flame out. If no corrective action is taken, either by the operator or by the control system, the combustion will not relight again and the engine will spool down. The following figures 6.1 through 6.4 show the effects and the reaction of the standard control system to a flame out.

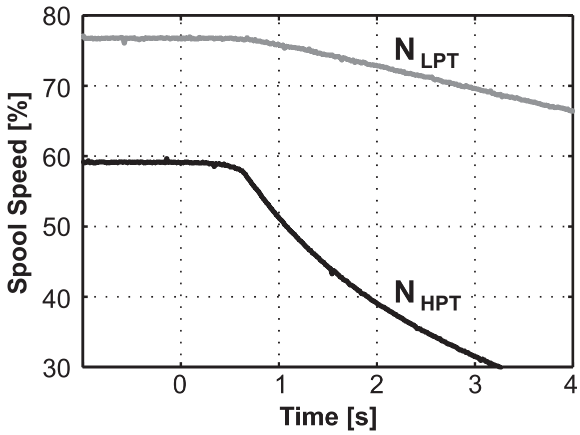


FIG. 6.1: SPOOL SPEEDS

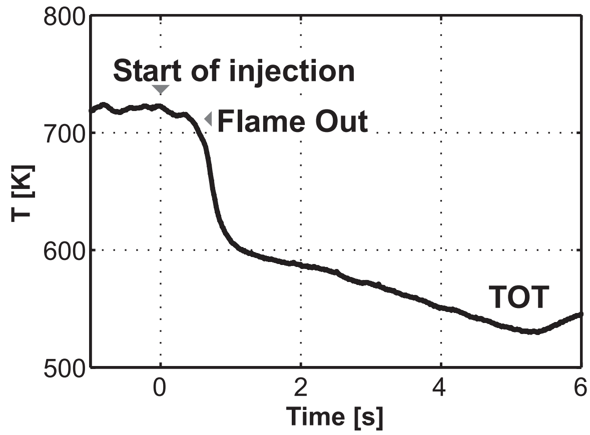


FIG. 6.2: TOT DURING ENGINE FLAME OUT

Figures 6.1 and 6.2 depict the flame out after a CO_2 injection is initiated with the engine running at idle power. At t=0.6 s the combustion becomes unstable and consequently stops. As a result of the ended combustion, the compressor outlet pressure p_C instantly drops (figure 6.3). The control system increases the fuel flow to counteract the decreasing gas generator speed, however, without the engagement of the ignition the combustion will not relight. At t=0.93 s the peak fuel flow is reached before the fuel-air ratio limiter restricts the metered fuel flow to prevent surge during an eventual acceleration. At t=5.0 s the fuel flow is stopped manually. This experiment, though not spectacular, shows the need for the control system to detect a flame out and take appropriate measures, i.e. engage the ignition.

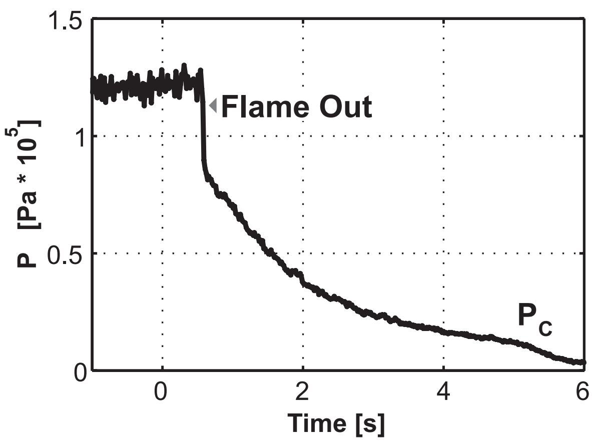


FIG. 6.3: COMPRESSOR EXIT PRESSURE

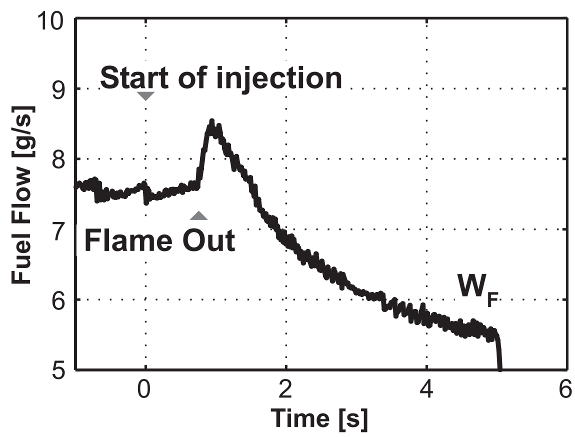


FIG. 6.4: COMMANDED FUEL FLOW

6.2 Carbon Dioxide Injection

The following figures show the impact of the injection of smaller amounts of heated CO_2 (approximately 50 g/s) into the engine's intake. Two test runs are shown at a low power setting and near full power respectively (delivered shaft torque 100 Nm and 300 Nm). During the tests the engine's FADEC is active and tries to maintain a constant output shaft speed of 100%. The controller successfully keeps the speed, so the delivered shaft power during the injection is nearly constant. CO_2 is injected into the plenum chamber for a duration of 5 seconds (see figure 5.1). The relevant time span is shaded in the following figures.

Figure 6.5 depicts the temperature in the compressor's intake plain. Although the amount of injected CO_2 is constant for both power settings, the gas temperature for the increased power setting is higher. This effect is caused by the setup of the heat exchanger (ref. figure 5.1) which is subjected to the engine's exhaust gas temperature and consequently the temperature of the injection gas correlates with the power setting.

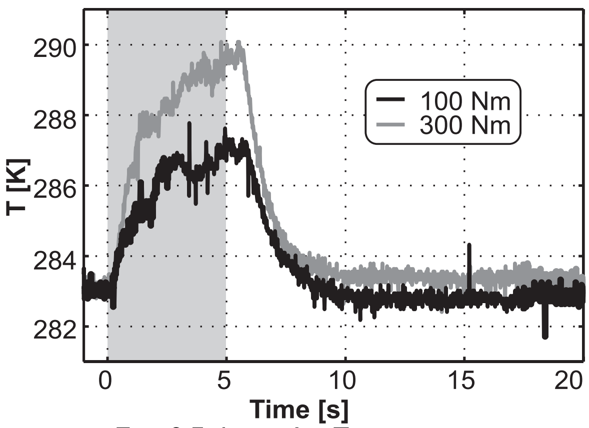


FIG. 6.5: INLET AIR TEMPERATURE

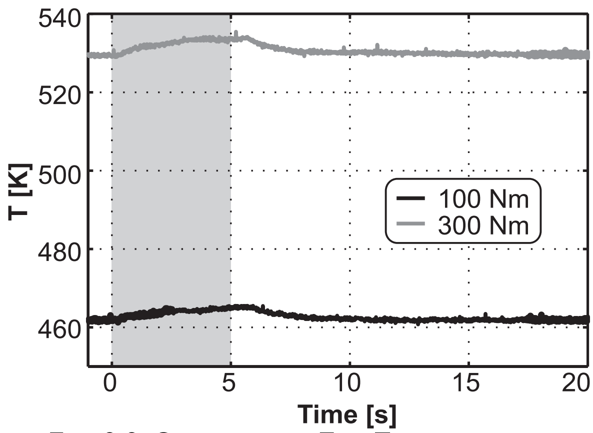


FIG. 6.6: COMPRESSOR EXIT TEMPERATURE

Figure 6.6 shows the compressor exit temperature. Although a small rise can be seen, if corrected to the changed intake condition and in combination with a basically unchanged compressor outlet pressure (Figure 6.8), a slight improvement in the compressor's efficiency can be observed. This is due to the changed thermodynamic properties of the intake gas. Inherent with this improvement is a marginal reduction in the compressor's spool speed (Figure 6.7).

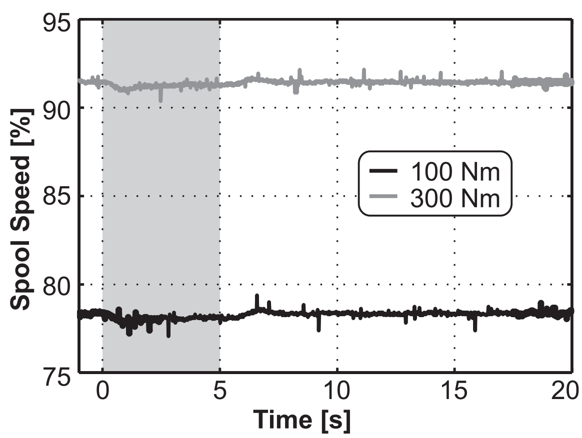


FIG. 6.7: GASGENERATOR SPOOL SPEED (N_{HPT})

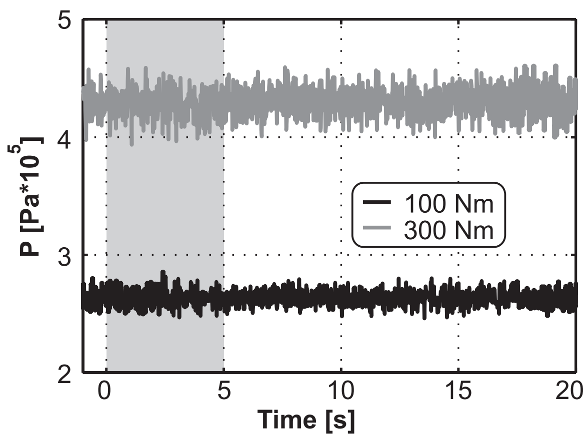


FIG. 6.8: COMPRESSOR EXIT PRESSURE

The following figure 6.9 depicts the control system's reaction to the injection. Fuel flow is increased as a reaction to the increased inlet temperature on the one hand, and to offset a slight reduction in the output spool speed on the other hand. The spool speed remains within 0.3% of its demand value, however, the offset cannot be fully compensated by the control system, what clearly indicates the need of an adaptation of the control laws.

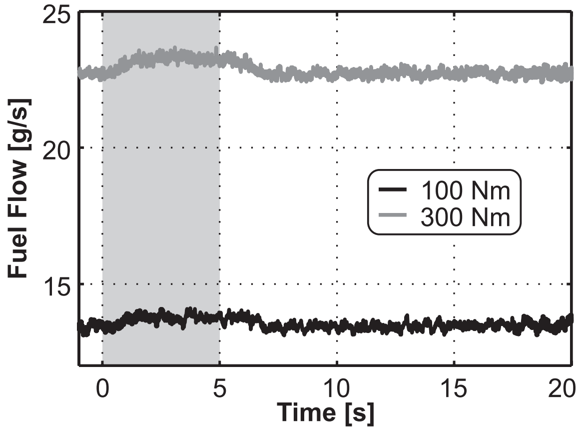


FIG. 6.9: FUEL FLOW

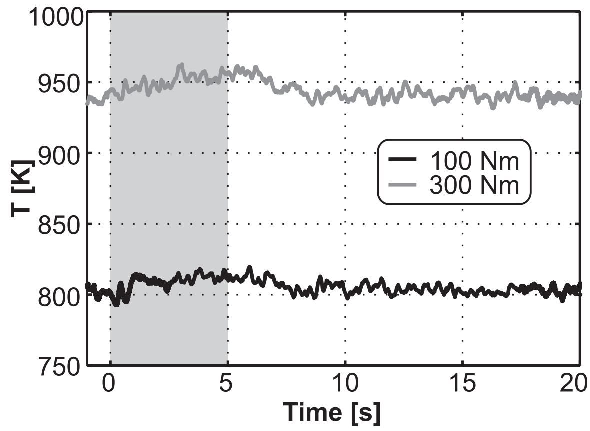


FIG. 6.10: TURBINE OUTLET TEMPERATURE (TOT)

The lapse of the turbine outlet temperature (TOT, Figure 6.10) shows a marginal increase, but if corrected to the changed intake temperature, it remains fairly constant, indicating a constant operation of both turbines.

If the metered fuel flow is plotted against its mean value of the time span excluding the CO_2 injection, a rise of 2.5% to 4.5% can be seen. This rise can be attributed mainly to a decreasing combustion chamber efficiency due to the reduced partial oxygen content. These findings are matched by the results of the engine observer model (see figure 7.9) as shown in the next chapter.

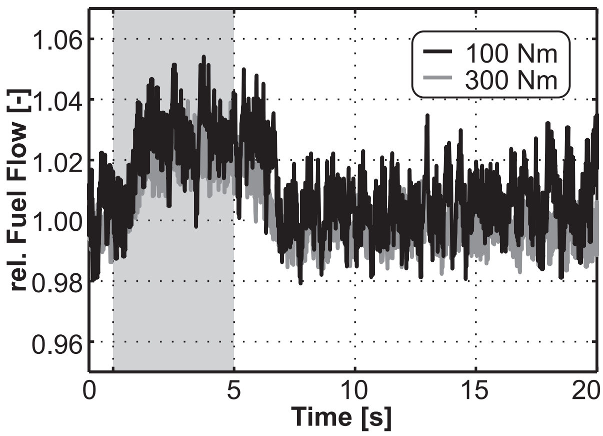


FIG. 6.11: RELATIVE CHANGE IN FUEL FLOW

7. OBSERVER MODEL

7.1 Introduction

For online detection of the contaminated air injection as described in section 6 the use of a parameter identifying state observer, including a non-linear engine model, is evaluated. The whole design and testing process is done within the MATLAB/Simulink environment. After a brief overview of the underlying theory, the design of a non-linear observer is described and first results of evaluations, using both simulated and measured data, are shown.

7.2 Theoretical Background

7.2.1 Extending the State Space System

State observers are originally used to determine non-measurable state variables of dynamic systems from known measurement and input variables, to make them available for state feedback controllers. There is, however, the possibility to treat parameters, which shall be identified, as additional states of the dynamic system. This adds dimensions to the system's state space. Consider the following linear system in state space representation, where $\mathbf{u_u}$ denotes unknown parameters or disturbances to be identified:

$$\dot{\mathbf{x}} = \mathbf{Ax} + \mathbf{Bu} + \mathbf{B_u u_u} \tag{7.1}$$

$$\mathbf{y} = \mathbf{Cx} + \mathbf{Du} + \mathbf{D_u u_u} \tag{7.2}$$

To be able to design an observer for this system that is capable of identifying the unknown parameters $\mathbf{u_u}$, the first step is to build a disturbance model for these parameters:

$$\dot{\mathbf{x}}_\mathbf{d} = \mathbf{A_d x_d} + \mathbf{B_d u_d} \tag{7.3}$$

$$\mathbf{y_d} = \mathbf{C_d x_d} + \mathbf{D_d u_d} \tag{7.4}$$

$$\mathbf{u_u} = \mathbf{y_d}$$

For step-wise varying parameters, the corresponding matrix coefficients would be

$$\mathbf{A_d} = \mathbf{0} \quad \mathbf{B_d} = \mathbf{I} \quad \mathbf{C_d} = \mathbf{I} \quad \mathbf{D_d} = \mathbf{I}\,.$$

After including the disturbance model into the system, the state space equations become

$$\dot{\mathbf{x}}_\mathbf{e} = \mathbf{A_e x_e} + \mathbf{B_e u_e} \tag{7.5}$$

$$\mathbf{y_e} = \mathbf{C_e x_e} + \mathbf{D_e u_e} \tag{7.6}$$

with

$$\dot{\mathbf{x}}_\mathbf{e} = \begin{bmatrix} \mathbf{x} \\ \mathbf{x_D} \end{bmatrix} \quad \mathbf{u_e} = \mathbf{u} \quad \mathbf{y_e} = \mathbf{y}$$

$$\mathbf{A_e} = \begin{bmatrix} \mathbf{A} & \mathbf{B_u C_d} \\ \mathbf{0} & \mathbf{A_d} \end{bmatrix} \quad \mathbf{B_e} = \begin{bmatrix} \mathbf{B} \\ \mathbf{0} \end{bmatrix}$$

$$\mathbf{C_e} = \begin{bmatrix} \mathbf{C} & \mathbf{D_u C_d} \end{bmatrix} \quad \mathbf{D_e} = \mathbf{D}\ .$$

For this extended system, the observability can be checked and an appropriate gain matrix can be designed.

7.2.2 Observability Checks

According to Kalman [8] the extended linear system is observable only if the rank of its observability matrix

$$\mathbf{Q_O} = \begin{bmatrix} \mathbf{C_e} & \mathbf{C_e A_e} & \cdots & \mathbf{C_e A_e^{n_e - 1}} \end{bmatrix}^{\mathrm{T}}$$

equals the number of system states n_e. This test, however, only leads to a yes-no-type answer to the question of observability. There are other methods that can be used to quantify the degree of a given system's observability. The method used here is based on a geometrical point of view of the problem. The idea is to find the vector that is most orthogonal to the row vectors of the normalized observability matrix $\mathbf{Q_{O,n}}$. The more orthogonal this vector is to the row vectors, the less observable is the system and the found vector points into the direction in the system's state space that is least observable [9]. It can be shown that the task of finding this vector is equivalent to finding the smallest eigenvalues of the matrix $\mathbf{Q_{O,n}^T Q_{O,n}}$. The smaller this eigenvalue, the less observable is the system. The corresponding eigenvector is the sought vector and points into the direction of the worst observability.

7.2.3 Observer Gain Design

After the observability issues have been clarified, a suitable gain matrix can be shaped. The equations describing a linear observer using the gain matrix $\mathbf{L}$ are

$$\dot{\hat{\mathbf{x}}}_\mathbf{e} = \mathbf{A_e}\hat{\mathbf{x}}_\mathbf{e} + \mathbf{B_e u_e} + \mathbf{L}\left(\mathbf{y_e} - \hat{\mathbf{y}}_\mathbf{e}\right) \tag{7.7}$$

$$\hat{\mathbf{y}}_\mathbf{e} = \mathbf{C_e}\hat{\mathbf{x}}_\mathbf{e} + \mathbf{D_e u_e} \tag{7.8}$$

where the hatted values denote estimated quantities. It can be seen that the observer is basically a model of the system plus a feedback of the estimation error $\mathbf{y_e} - \hat{\mathbf{y}}_\mathbf{e}$ times the gain matrix $\mathbf{L}$. This shows that the gain matrix determines the degree to which measured data is taken into account when generating estimates for state and output variables. The design of this gain matrix is crucial for the observer to work correctly. In general, higher gains lead to a faster dynamic behavior of the observer, but cause an increased sensitivity to measurement noise.

There are two common ways of designing the gain matrix: pole placement or a solution of the Riccati matrix differential equation. When using pole placement techniques, the closed loop observer poles have to be chosen appropriately and certain algorithms are used to produce the corresponding gains. By solving the non-linear matrix Riccati differential equation, using information about system and measurement noise covariances, the prediction error can be minimized. The resulting observer is an optimum filter or Kalman filter.

7.2.4 Linear and Non-Linear Observers

As can be seen in equations (7.7) and (7.8), the observer consists of a model of the system plus the feedback of the estimation errors. The model used, however, does not necessarily have to be a linear model in state space representation. All types of engine models (state space, performance synthesis, ...) can be used. The general setup is shown in figure 7.1.

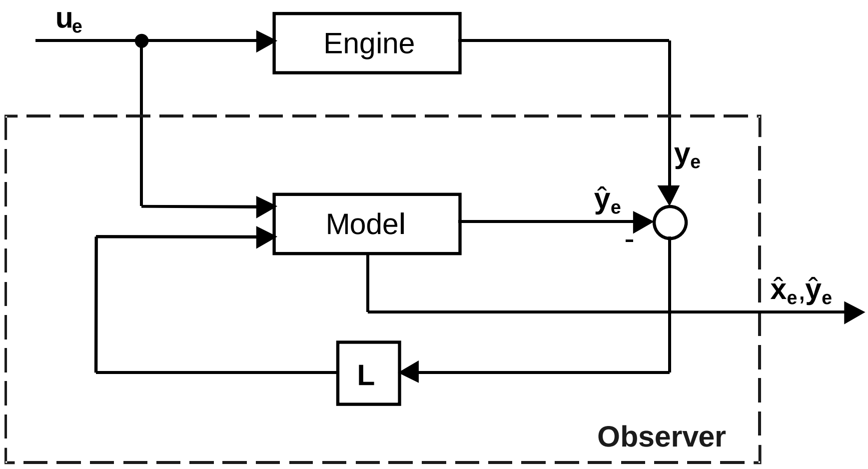


FIG. 7.1: ENGINE AND OBSERVER

7.3 Results

7.3.1 Observer and Gain Design

To observe the critical inlet conditions using an on-board observer, an observer has to be set up to identify the drop of combustion chamber efficiency and, if possible, the simultaneous changes in compressor and turbine efficiency. The engine model used here in the observer is a non-linear combination of ten linear state space models obtained at different power settings [10]. The model itself contains five system states, the high pressure spool speed and four component metal temperatures ($\mathrm{N1} =$ high pressure shaft speed, $\mathrm{TC} =$metal temperatures, $\mathrm{HPC} =$ compressor, $\mathrm{HPT} =$ high pressure turbine, $\mathrm{LPT} =$ low pressure turbine, $\mathrm{CC} =$ combustion chamber):

$$\mathbf{x} = \left[\mathrm{N1} \quad \mathrm{TC\_HPC} \quad \mathrm{TC\_CC} \quad \mathrm{TC\_HPT} \quad \mathrm{TC\_LPT}\right]^{\mathrm{T}}$$

The system has the following control input, unknown parameter input and measured output variables ($\mathrm{WF} =$ fuel flow, $\mathrm{N2} =$ low pressure shaft speed, $\mathrm{DE} =$ changes in efficiency, $\mathrm{PC} =$ compressor outlet pressure, $\mathrm{TOT} =$ turbine outlet temperature):

$$\mathbf{u} = \left[\mathrm{WF} \quad \mathrm{N2} \quad \mathrm{Customer\_Bleed}\right]^{\mathrm{T}}$$

$$\mathbf{u_u} = \left[\mathrm{DE\_HPC} \quad \mathrm{DE\_HPT} \quad \mathrm{DE\_CC}\right]^{\mathrm{T}}$$

$$\mathbf{y} = \left[\mathrm{N1} \quad \mathrm{PC} \quad \mathrm{TOT}\right]^{\mathrm{T}}$$

To be able to detect the changes in the component efficiencies $\mathrm{DE\_HPC}$, $\mathrm{DE\_HPT}$, and $\mathrm{DE\_CC}$, these variables have to be treated as additional system states and the resulting extended system has to be observable, using the three measurements $\mathrm{N1}$, PC, and TOT. Since there is no *a-priori* knowledge about the efficiency changes, a disturbance model for step disturbances is used in accordance with equations (7.3) and (7.4).

The next step is to check the observability of the extended system using different configurations of parameters to be identified and different sets of measured variables. Applying the method described in section 7.2.2, the resulting smallest eigenvalues of the normed matrix $\mathbf{Q}_{\mathbf{O,n}}^{\mathbf{T}}\mathbf{Q}_{\mathbf{O,n}}$ are shown in table 7.1. Since the efficiency adder of the combustion chamber $\mathrm{DE\_CC}$ is considered most important for this case, a first approach is to only detect changes in this parameter. Because of the high noise level and the slow sensor dynamics (compared to $\mathrm{N1}$ and PC sensors) of the TOT measurement, only $\mathrm{N1}$ and PC are used for these first tests, even though the degree of observability then drops significantly (compare table 7.1).

Available measurements	Identified parameters	Smallest eigenvalue	Least observable direction
N1 PC TOT	DE_HPC DE_HPT DE_CC	$1.119 \cdot 10^{-7}$	DE_HPC DE_HPT
N1 PC	DE_HPC DE_HPT DE_CC	$7.773 \cdot 10^{-16}$ (not observable!)	DE_HPC DE_HPT
N1 PC TOT	DE_CC	$1.127 \cdot 10^{-5}$	TC_HPC TC_LPT DE_CC
N1 PC	DE_CC	$3.690 \cdot 10^{-7}$	TC_HPC TC_LPT DE_CC

TABLE 7.1: OBSERVABILITY OF DIFFERENT CONFIGURATIONS OF MEASURED VARIABLES AND IDENTIFIED PARAMETERS

After building the extended system using the state space matrices of a linearization, obtained near the engine design point (which is believed to be the most representable), a gain matrix can be designed. A prerequisite of designing a Kalman filter gain is the disturbability of the system, which is not fulfilled here due to the chosen disturbance model for step disturbances. To overcome this problem, a sub-optimum solution to the Riccati matrix equation can be found by using the modified system matrix

$$\mathbf{A}_{\mathbf{e,S}} = \mathbf{A}_{\mathbf{e}} + \alpha \mathbf{I}$$

for all states added during the extension of the system, where α is a positive scalar quantity, determining the stability margin of the closed loop observer system. The covariance matrices chosen correspond to a full scale noise of the unknown inputs $\mathrm{DE\_HPC}$ and $\mathrm{DE\_LPT}$ (matrix $\mathbf{Q}_{\mathbf{n}}$), a 1% of full scale noise on the sensor signals $\mathrm{N1}$ and PC (matrix $\mathbf{R}_{\mathbf{n}}$) and no covariance between process and measurement noise (matrix $\mathbf{N}_{\mathbf{n}}$):

$$\mathbf{Q}_{\mathbf{n}} = \begin{bmatrix} 1 & 0 \\ 0 & 1 \end{bmatrix} \quad \mathbf{R}_{\mathbf{n}} = \begin{bmatrix} 0.01 & 0 \\ 0 & 0.01 \end{bmatrix} \quad \mathbf{N}_{\mathbf{n}} = \begin{bmatrix} 0 & 0 \\ 0 & 0 \end{bmatrix}$$

The stability margin α is chosen to be 0.5, thus representing a reasonable compromise between fast enough observation dynamics and good enough sensor noise rejection. The calculated gain matrix $\mathbf{L}$ leads to the following closed loop observer poles λ_O:

$$\lambda_O = \begin{bmatrix} -2.003 & -0.527 & -0.288 & -0.121 & -0.080 & -0.500 \end{bmatrix}^T$$

Compared to the original system pole locations at

$$\lambda = \begin{bmatrix} -1.068 & -0.517 & -0.373 & -0.120 & -0.077 \end{bmatrix}^T$$

it can be seen that the first pole, which according to its eigenvector corresponds mainly with the spool speed $\mathrm{N1}$, is shifted towards faster dynamics. The other poles basically remain at their original locations and the additional observer pole introduced by including $\mathrm{DE\_CC}$ lies at the given stability margin of $\alpha = 0.5$.

Since the engine dynamics changes from idle to the highest power setting, the observer gain also varies. However, [11] suggests to use a constant gain matrix for all power settings, which only introduces small changes in the observer's dynamic behavior at different engine power settings.

7.3.2 Results using Simulated Data

First evaluations are carried out using simulated data. An engine model is used to provide "measurements" to the observer, facilitating experiments excluding modeling errors and measurement errors. The overall MATLAB/Simulink setup can be seen in figure 7.2, which is similar to the basic setup shown in figure 7.1.

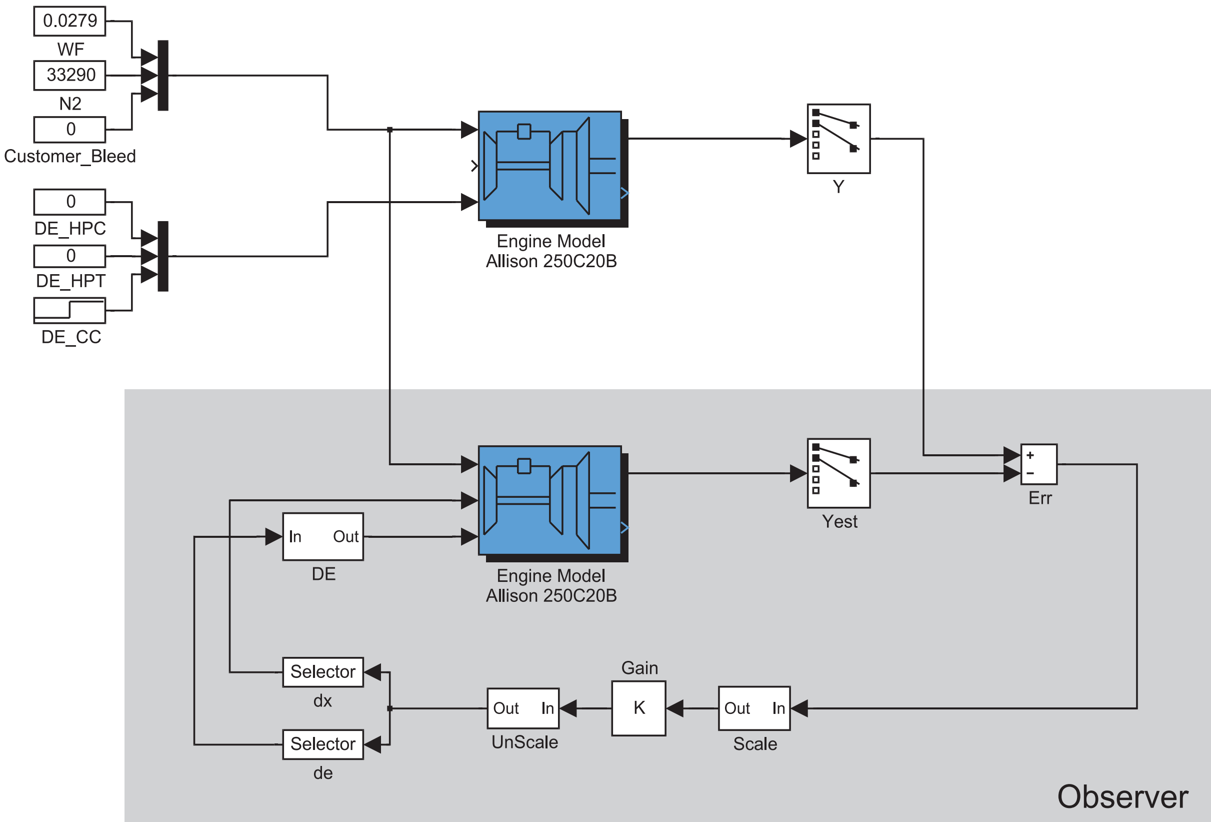


FIG. 7.2: SIMULINK SETUP

Figure 7.3 shows the reaction of the observer to a step change in $\mathrm{DE\_CC}$. The change is correctly detected after approximately five seconds, with a small overshoot. This corresponds to a time constant of approximately 2 seconds and thus is in accordance with the observer pole at -0.5. Figure 7.4 shows the "measured" and observed spool speed N1. Figures 7.5 and 7.6 show the same change in DE_CC, but during a simultaneous acceleration of the engine.

Due to the fact that only changes in combustion efficiency were considered during the design of the observer, it is quite sensible to changes in compressor or turbine efficiency. Figures 7.7 and 7.8 show an erroneous estimation of the combustion efficiency adder due to a step drop in compressor efficiency adder of two percent. It can be seen that the observer tries to match the "measured" and estimated outputs for N1 and PC as good as possible by changing DE_CC (see high pressure spool speed N1 in figure 7.8, for example). This behavior could be improved by adding other efficiency changes like DE_HPC and DE_HPT to the list of identified parameters, at an increased demand of measured variables, however, or by implementing a surveillance logic that validates the identified values by checking the residual estimation error, which would sacrifice estimation speed.

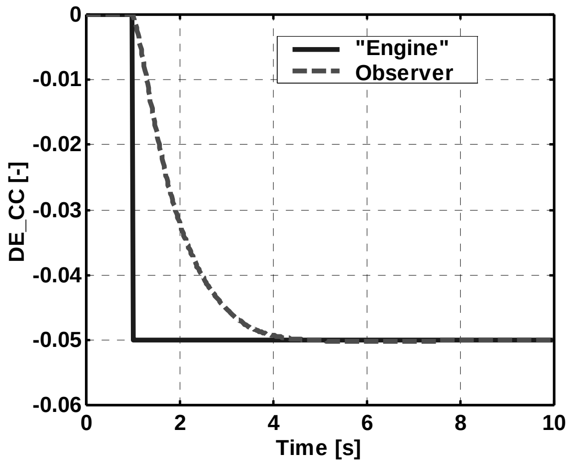


Fig. 7.3: Simulated change in DE_CC: Identified and "real“ DE_CC

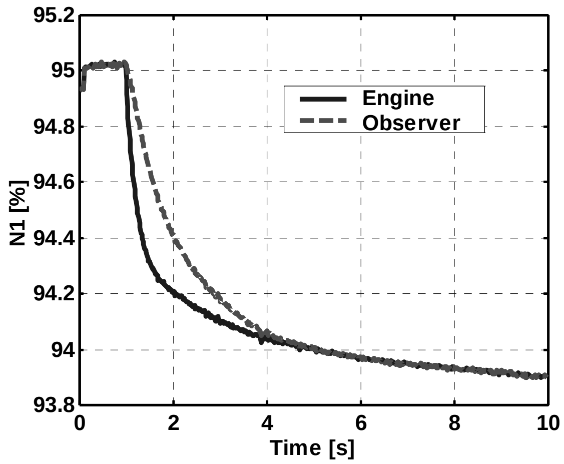


Fig. 7.4: Simulated change in DE_CC: Identified and "real" N1

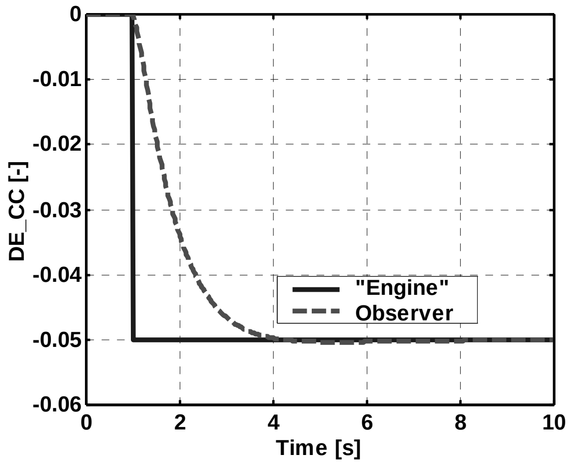


Fig. 7.5: Simulated change in DE_CC during engine acceleration: Identified and "real" DE_CC

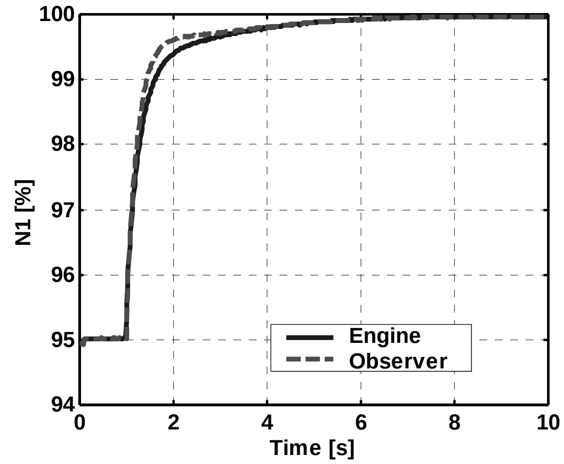
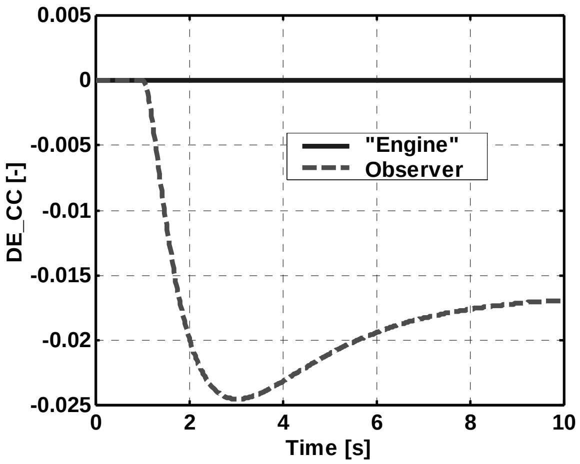


Fig. 7.6: Simulated change in DE_CC during engine acceleration: Identified and "real" N1



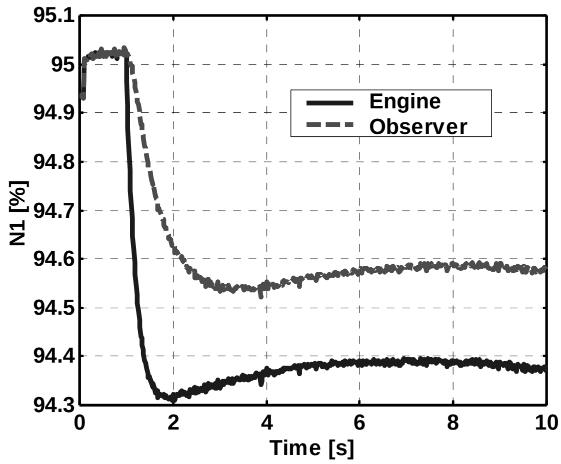
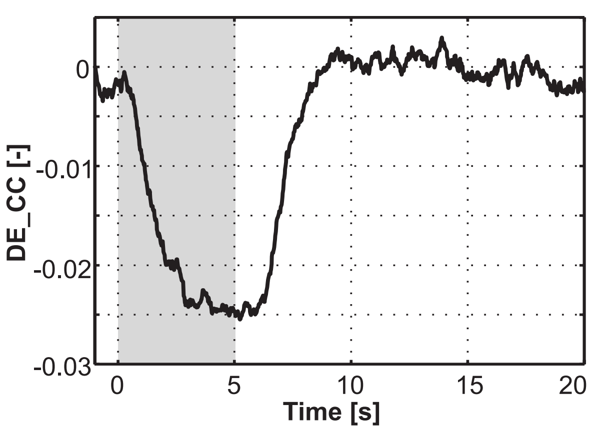


FIG. 7.7: SIMULATED CHANGE IN DE_HPC: IDENTIFIED AND "REAL" DE_CC

FIG. 7.8: SIMULATED CHANGE IN DE_HPC: IDENTIFIED AND "REAL" N1

7.3.3 Results using Measured Data

After the test runs with simulated data, the observer is evaluated using measured engine data from the rig experiments described in section 6. First the data of the carbon dioxide injection at a torque of 100 Nm (see section 6.2 and figures 6.5 to 6.11) is used. The model included in the observer was adapted to compensate measurement or modeling bias at this operating point of the engine. Figure 7.9 shows the detection of a 2.5% drop in combustion efficiency during the injection, where the shaded area shows the duration of the injection. This drop of combustion efficiency is in accordance with the increase in metered fuel flow shown in figure 6.11. Figure 7.10 exemplary demonstrates how the observer filters the measurements and tries to match its output variables and the measurements (the figure shows the high pressure spool speed, N1).



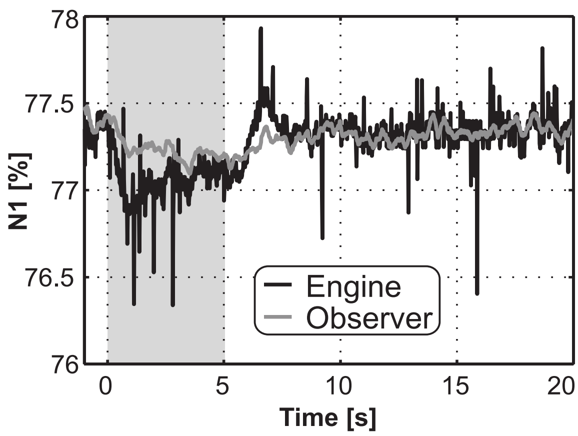


FIG. 7.9: CARBON DIOXIDE INJECTION: IDENTIFIED DE_CC

FIG. 7.10: CARBON DIOXIDE INJECTION: IDENTIFIED AND MEASURED N1

Since observers are known to be quite sensitive to measurement bias, measurements from an acceleration from $N1 = 72\%$ to $N1 = 78\%$ are fed into the observer (corresponding fuel flow signal shown in figure 7.11). Figure 7.12 shows that due to modeling and / or measurement errors, a positive change in combustion efficiency (2%) is detected and after the acceleration there is still a bias of -1% left. This demonstrates how crucial an accurate engine model is for a successful parameter estimation. A possibility to avoid the effects introduced by measurement or modeling bias is to implement a bias detection / compensation algorithm that corrects the simulated values according to the measurements before feeding them into the observer, which would sacrifice identification speed, however. Since it is only the relative change in combustion efficiency that is of interest here, not its absolute value, the measurement or modeling bias is not considered in the last two test cases. Finally, the measurements obtained by two carbon dioxide injections performed at idle power are fed into the observer. Figure 7.13 shows an identified drop of 10% during the carbon dioxide injection, figure 7.14 a drop of 60% following a flame out in the combustion chamber (refer section 6.1, figures 6.1 to 6.4).

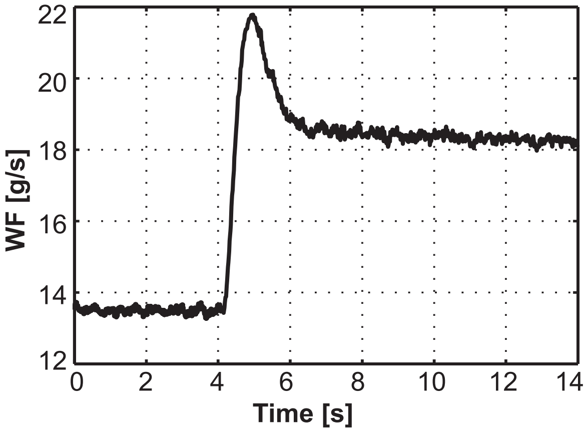


FIG. 7.11: ENGINE ACCELERATION: FUEL FLOW

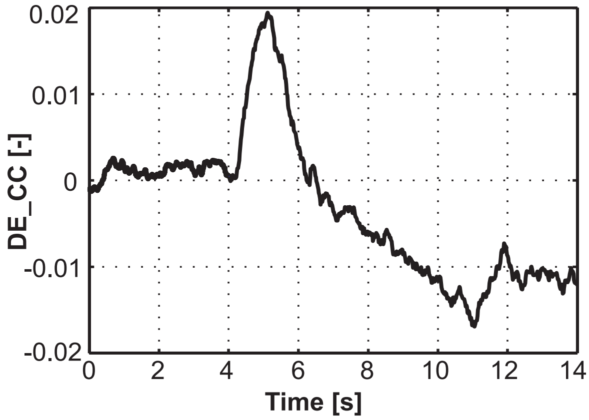


FIG. 7.12: ENGINE ACCELERATION: IDENTIFIED DE_CC

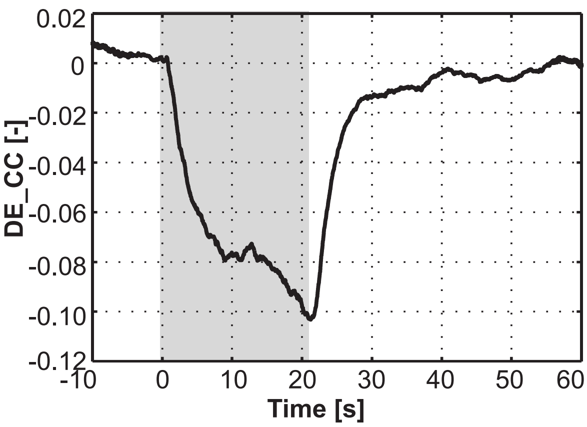


FIG. 7.13: CARBON DIOXIDE INJECTION WITHOUT ENGINE FLAME-OUT

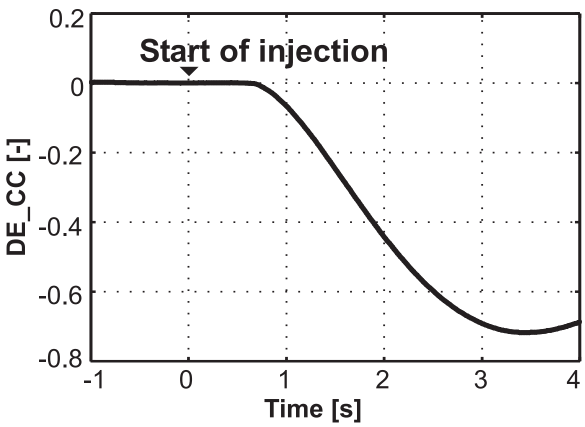


FIG. 7.14: CARBON DIOXIDE INJECTION LEADING TO FLAME-OUT

8. CONCLUSION AND OUTLOOK

In the previous chapters, a practical setup was presented to simulate the intake of contaminated air into a helicopter turbo-shaft engine. For a detection of the contaminated air injection into the inlet, tests were made using a state observer identifying the change in combustion efficiency. Experiments with simulated data showed a fully satisfactory behavior of the observer in absence of parameter changes not included in its design. The experiments conducted using data obtained by rig experiments revealed problems arising through modeling and/or measurement errors. With the main problem items identified, further investigations will be made if and how this type of detection could be exploited for an on-board system.

Requirements for Engine Control Systems

The investigation of helicopter accidents revealed that the restoration of engine power is of eminent importance. Loss of power in low flight altitudes almost always leads to a forced landing. Autorotational landings, although physically possible, often end in accidents due to the high pilot workload and the absence of a suitable landing terrain. The restoration of engine power, even if only partially, will increase the chances of safe continuation of the mission. However, due to the high workload of the pilot and other possible distractions (e.g. choice of landing terrain, jettisoning of external loads, etc.) the restoration of engine power has to be realized automatically by the control system. Unless the pilot commands an engine shut-down, the FADEC system should always try to maintain engine power.

Furthermore, in the advent of a possible loss of power due to environmental conditions, the pilot should be given an easily readable warning in the cockpit, rather than relying on the interpretation of gauges.

The feasibility of this depends on the integration of adaptive on-board engine models into the control system to identify changed environmental conditions and/or changes in the engine's performance due to degradation.

With this adaptive control, engine operation not only becomes safer in marginal or critical conditions, but engine response and fuel economy will also improve under normal operating conditions.

9. REFERENCES

[1] Bundestelle für Flugunfalluntersuchung: Unfallstatistik, Braunschweig, 1996

[2] Helicopter Safety: Fatal Turbine-Helicopter Accidents, Flight Safety Foundation, 1995 (based on NTSB and FAA Data)

[3] Luftfahrt-Bundesamt: Flugunfallinformation V67, Braunschweig, 1987

[4] Gabler R.: Betriebsverhalten von Wellenleistungsgasturbinen bei Verdichterinstabilitäten und Methoden zur Restabilisierung, Dissertation, Lehrstuhl für Flugantriebe, München, 1998

[5] Erhard W., Gabler R., Preiss A., Rick H.: Monitoring and Control of Helicopter Engines at Abnormal Operating Conditions, RTO Meeting Proceedings 8: Design Principles and Methods for Aircraft Gas Turbine Engines, 1999

[6] Preiss A.: Entwicklung eines Labor FADEC Systems zur Regelung von Gasturbinen, DGLR Jahrestagung, München, 1997

[7] Auer, E.J:.Modellbildung, Simulation und Entwicklung eines digitalen Reglers für eine Hubschraubergasturbine,VDI Fortschrittsbericht, Reihe 8/581, VDI Verlag Düsseldorf, 1996

[8] Kalman, R.: On the general Theory of Control Systems. Proceedings of the 1st IFAC-Congress, Vol. 1, Moscow, 1960

[9] Brown, R.G.: Not Just Observable but How Observable?. Proceedings of the National Electronics Conference, Vol. 22, Chicago, 1966

[10] Gabler R., Auer E., Erhard W., Rick H.: Turboshaft Engine Monitoring System including Contingency Operation, 5th European Propulsion Forum, EPF95-17, Pisa, 1995

[11] Sugiyama, N.: On-Line System Identification of Jet Engine, AIAA, 1996

ONLINE-MONITORING FOR AIRCRAFT ENGINES – DEVELOPMENT AND APPLICATION

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1 INTRODUCTION

For modern aircraft engines high standards in terms of reduced emission and higher efficiency have to be met whilst development cost and time to market continuously have to be reduced. In this environment availability and analysis of engine test results during the development phase are crucial for development and therefore market success. New technologies have been developed to provide quick analyses of engine and component tests and to ensure safe and reliable testing even during critical maneuvers.

This paper describes the development and practical utilization of an Online-monitoring system for vibration and other dynamic data which has been developed in cooperation of Rolls-Royce Deutschland and Müller-BBM VibroAkustik Systeme. The Online-monitoring system is now widely being used for a number of different tests and applications. Two examples for employing online-monitoring are presented and described in more detail.

For the development of the online signal monitoring the integration of the system in the analysis process for vibration data is crucial. The aim of safely monitoring vibration data is related to the objective of ensuring shorter analysis responses and early adjustments of analysis parameters.

2 ONLINE-MONITORING TASKS IN AIRCRAFT ENGINE DEVELOPMENT

The development of modern and efficient aircraft engines demands a development test program to be carried out for the whole engine and its components. Rig tests are performed at the beginning of the development process for engine components as compressor, turbine and combustion chamber. Development engines are built as test vehicles to perform extensive tests with several operating conditions to be examined and verified. In figure 2.1 elements of a test program during the development process are shown.

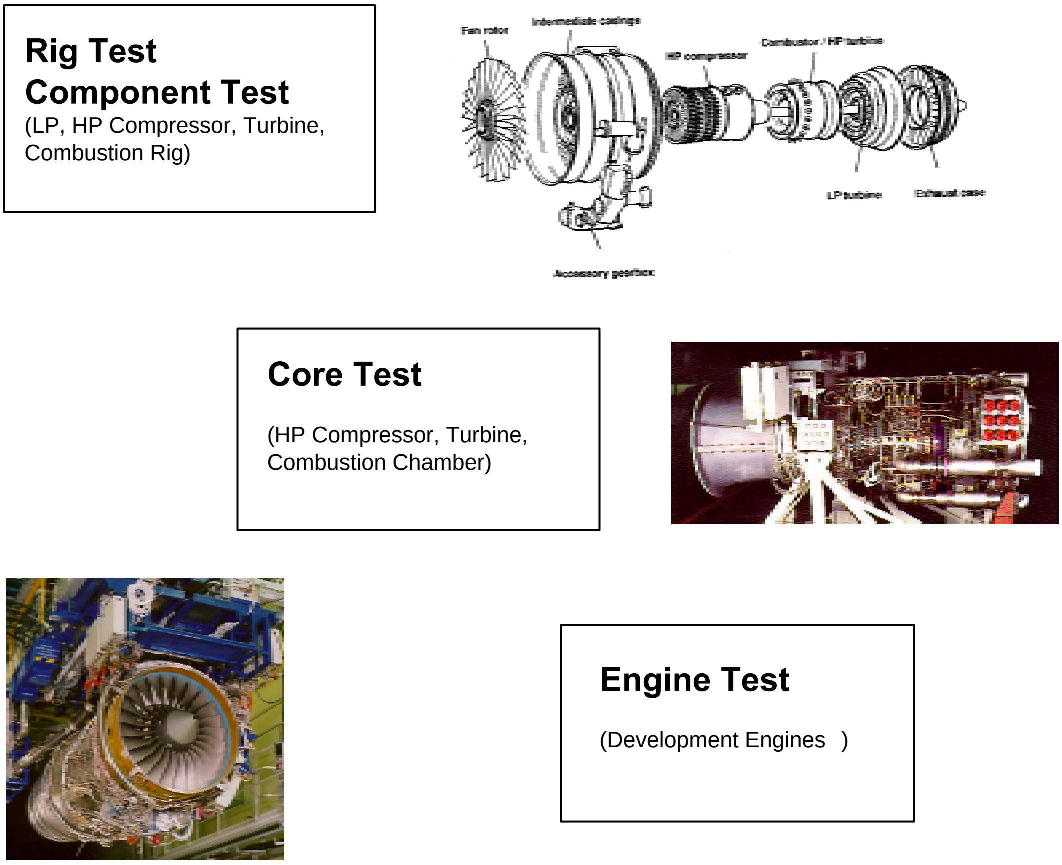

Figure 2.1: *Elements of the test program in the development process of aircraft engines*

Rig tests of the engine components are used to investigate basic aero-thermodynamic performance and to validate the basic structural integrity. These rigs are well-equipped with different measurement sensors because most parts are still accessible. In many cases real engine conditions can only be approximated and design parameters are simulated. Signal monitoring is required to allow for a safe testing and to observe whether the measured parameters, e.g. pressure fluctuations and structural vibrations, meet the predictions. Feedback during the test program is important to prove that experiments have been performed successfully and that the required instrumentation is working and giving expected results through the whole testing. With an efficient online signal analysis and monitoring test parameters can be changed quickly and experiments adapted during the test program.

Development engines are assigned to perform specific tests for compliance purposes either with the customer or the airworthiness authorities' requirements. Some certification tests require the engine to be operated at conditions equivalent to normal engine flight conditions for a defined number of hours. Furthermore critical engine tests are carried out where the engine is operated at its boundaries of aerodynamic and structural stability. This is necessary to verify that newly developed engines meet the high standards of safety demands. It has to be proved that the structural integrity of all engine parts along with the engine performance characteristics are ensured even for critical engine conditions. For all those tests with the engine operating at conditions, which cannot be fully predicted, the structural integrity and life endurance of the engine parts have to be evaluated and verified on-line. An efficient and reliable signal monitoring is essential during such tests, because design and build are of extensive cost. Quick analysis response helps to ensure test success and is necessary for reducing development times.

3 SYSTEM REQUIREMENTS

The task of on-line monitoring vibration data during the test program is integrated into the acquisition and analysis process of dynamic data at Rolls-Royce Deutschland. The integrated process of vibration measurement, data recording and analysis as a basis for data validation is shown in figure 3.1. Based on an intended integration of the online monitoring into the system environment at Rolls-Royce Deutschland the system requirements were defined.

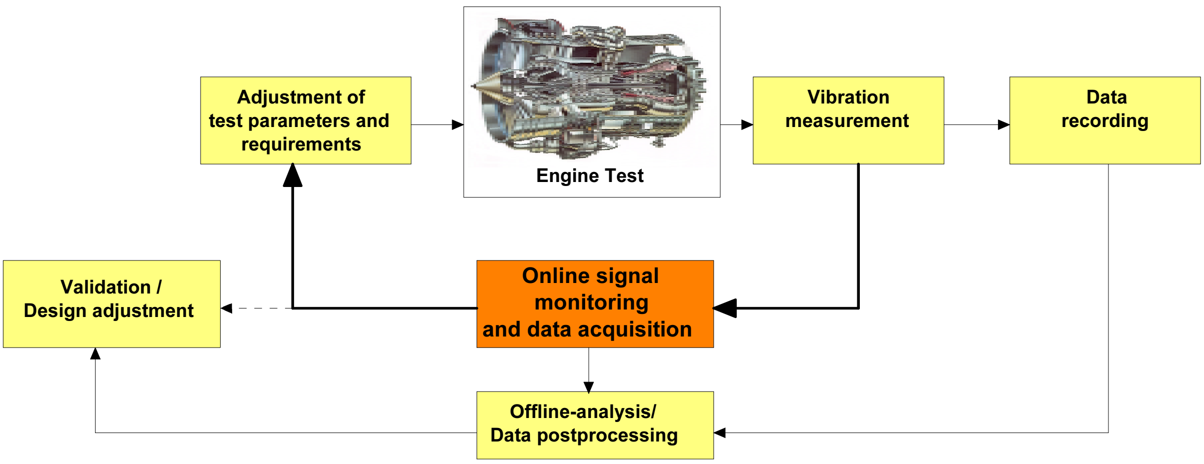

Figure 3.1: *Online-monitoring system as used at Rolls-Royce Deutschland*

The basic requirements for the development of an Online-monitoring system were defined as follows:

- Simultaneous monitoring of a minimum of 32 dynamic measurement channels.
- Simultaneous monitoring of different measurement parameters and types
- Combination of frequency analysis and limit monitoring
- Storage of frequency spectre and related information for later offline analysis
- Open system structure to allow for a future system extension

Generally the system is used for two categories of engine tests. For engine validation tests, where several engine maneuvers are carried out to validate the structural characteristics, it is important that unpredictable events and limit exceedances are protocolled and frequency spectre are stored. Furthermore the engineer should have an option for changing limit predictions if the expected characteristics deviate. For other long-term engine tests the vibration behavior of specific components have to be observed over the whole engine test. For changing vibrations characteristics of critical parts a safety monitoring without permanent engineering support is required.

System specifications

- *Definition of up to ten bandpass and order limits for each channel (based on typical blade vibration characteristics)*
- *Definition of warning and alarm limits for each bandpass and order*
- *Issue of a warning and alarm signal*
- *250 msec loop-time (20 kHz frequency range) between frequency spectre to allow for an evaluation of parameters during quasi stationary tests*
- *Spectre averaging for minimizing statistical influence*
- *Storage of APS data , bandpass and order levels and protocol information*

Figure 3.2: *Technical specifications for the Online-monitoring system*

The specifications for data storage and monitoring capacity are listed in figure 3.2. The specifications were based on the most comprehensive tasks of monitoring compressor or turbine blade vibrations. There usually up to ten natural frequencies with specific stress or strain limits have to be observed. Other applications for rotordynamic investigations demand order signals to be tracked based on excitations by the engine orders of LP and HP system.

4 SYSTEM REALIZATION

For the measurement hardware VXI technology, which is designed for complex measurement purposes, had to be selected to be compatible with existing systems. In figure 4.1 the measurement system used for online-monitoring, equipped with a VXI-mainframe with four 8-channel measurement cards, is shown. Data pre-processing is performed already on the measurement cards and includes the calculation of the frequency spectre. Thus the required loop-time of 250 msec (for a frequency range of 20 kHz) is achieved for the calculated bandpass and order levels. The data transfer rate between the VXI-mainframe and the workstation is limited by the MXI-bus interface to 1.2 - 1.4 MByte/sec. Therefore the autopower spectre are transferred to the workstation where bandpass and order levels are extracted and compared to the defined limits.

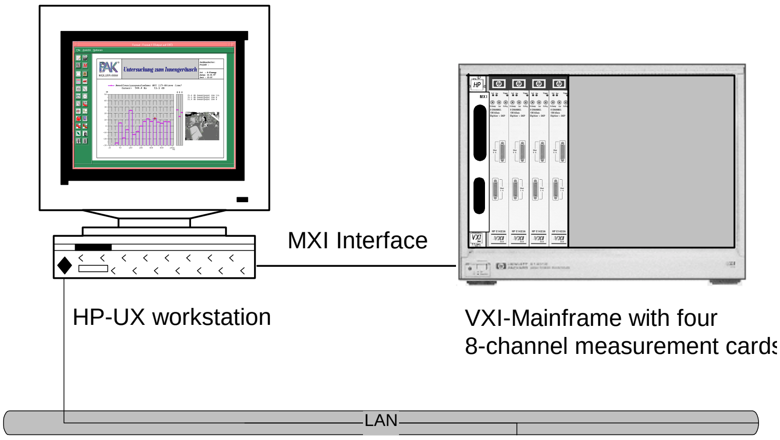

Figure 4.1: *Measurement setup for monitoring purpose*

The monitoring system was accomplished based on the PAK measurement system of Müller-BBM VibroAkustik Systeme GmbH. Extended postprocessing and monitoring tools are included in the modular software structure of PAK for the use as monitoring system.

For a complete data assessment and evaluation during the test a powerful graphical user-interface is provided. The user-interface as described in figure 4.2 includes a channel overview, where limit exceedances of all measured signals can be recognized. The limit overview directs the user to all monitored limits for each parameter. A bar graph allows for readings of measured peak and current values of the observed limits. The online-graphic features autopower spectre for two parameters, speed and history graphs. Here the coarses of the tracked bandpass and order limits are displayed in a time frame of up to 60 minutes.

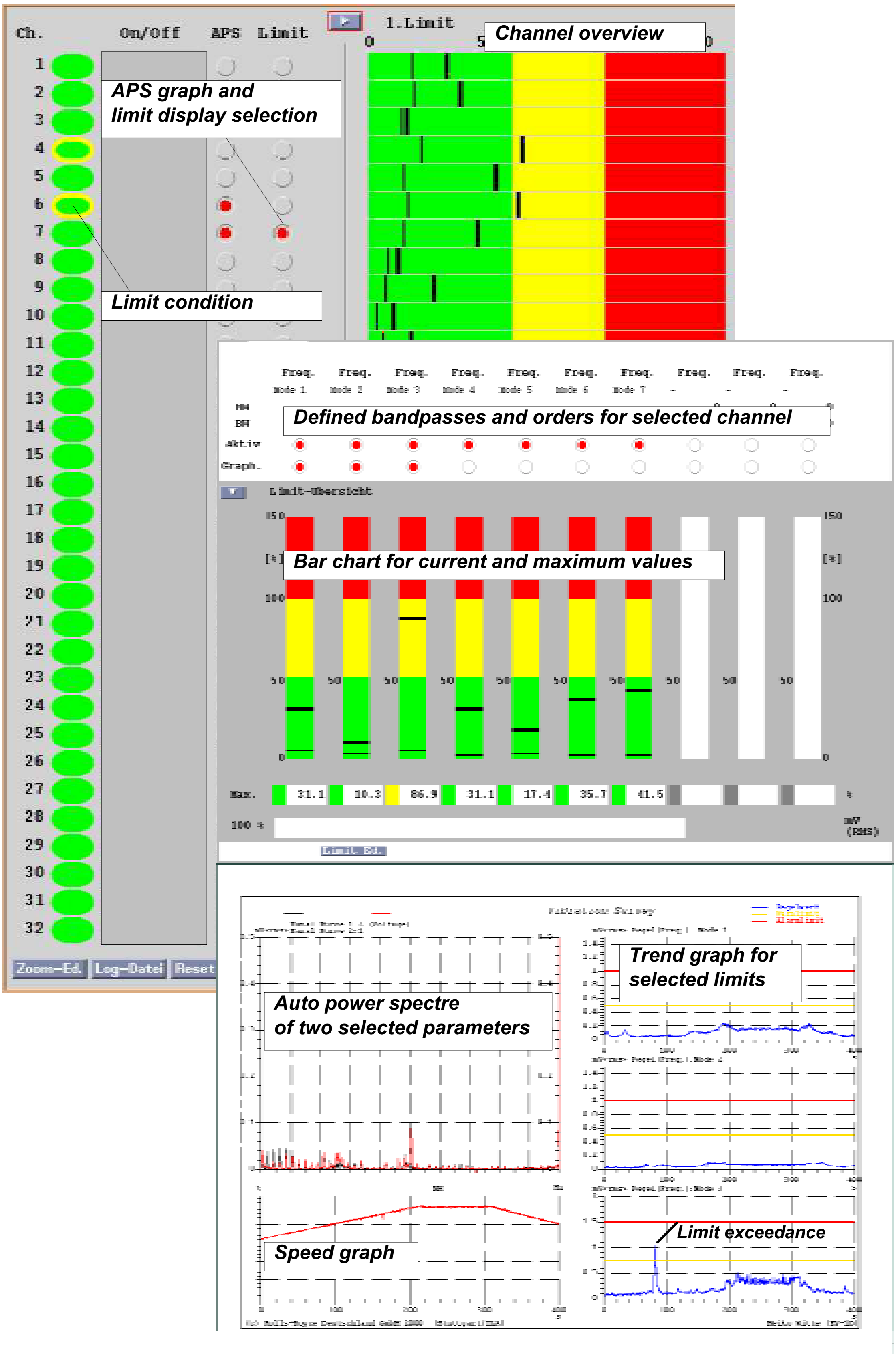

Figure 4.2: *Graphical user-interface of the Online-monitoring system*

Limit exceedances and vibration resonances can be evaluated and correlated with these information on-line during the test. The parameters displayed are selected interactively via the channel and limit overview. All necessary information and parameters are defined within multiple editors for channel selection, limit definition and test specifications prior to the test. Data storage is variable and includes a protocol function with a list of limit exceedances and peak hold values for each limit, and further an optional storage of autopower spectre and tracked limit data. The open software structure allows for a future implementation of multiple sampling rates and an extension of the channel number.

5 ONLINE-MONITORING IN USE

5.1 Monitoring applications

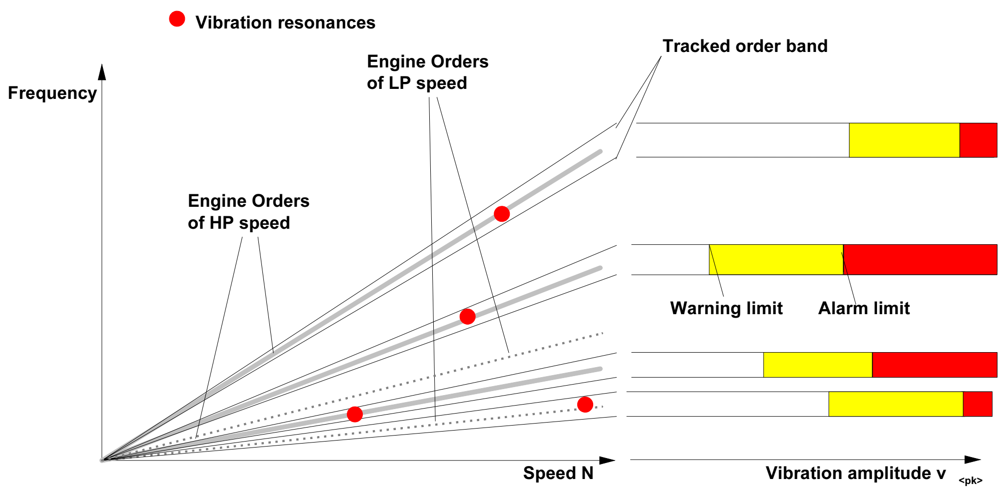

Figure 5.1: *Order band limit definition for online-monitoring*

The Online-monitoring system is being used for a multitude of different engine tests and experiments. The specter of monitoring tasks to be supported is wide which explains the needed flexibility of the system. As described in section 5.2 fan flutter monitoring requires specific order bands to be tracked from dynamic pressure transducer signals. Engine order tracking is necessary also for rotordynamic applications where specific resonances occur at multiples of speed frequency. In figure 5.1 the definition of limits for such tasks is shown. Different warning and alarm limits are set for a number of engine orders related to either low pressure or high pressure system speed for two shaft engines.

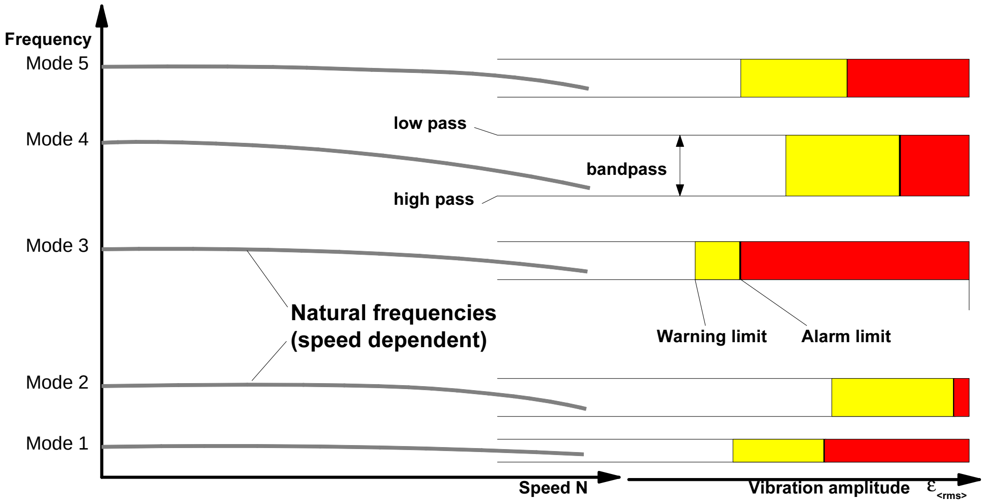

Figure 5.2: *Bandpass limit definition for online-monitoring*

For monitoring of structural characteristics of engine parts as blades, vanes, pipes or accessories bandpass limits are defined to observe vibrations in the natural frequencies. Because natural frequencies may shift according to speed and temperature a bandpass is defined which has to include all relevant resonances during transient maneuvers. For each vibration mode different warning and alarm limits are set respectively as shown in figure 5.2. By that means critical structural resonances can be tracked during long-term engine running. The system may run semi-automatically and will issue an alarm sound when defined bandpass limits are exceeded. An example is further described in section 5.3.

4.2 Monitoring of fan flutter

Development of the BR710 engine for the British marine elucidation airplane Nimrod demanded a compliance test to prove that the engine performance does not decrease significantly due to salt corrosion impact. As part of this testing movements of the fan working line and changes to the stability margin had to be investigated before and after corrosion impact. Also changes to the fan stability margin had to be shown to be in pre-defined limits by testing before and after the corrosion part.

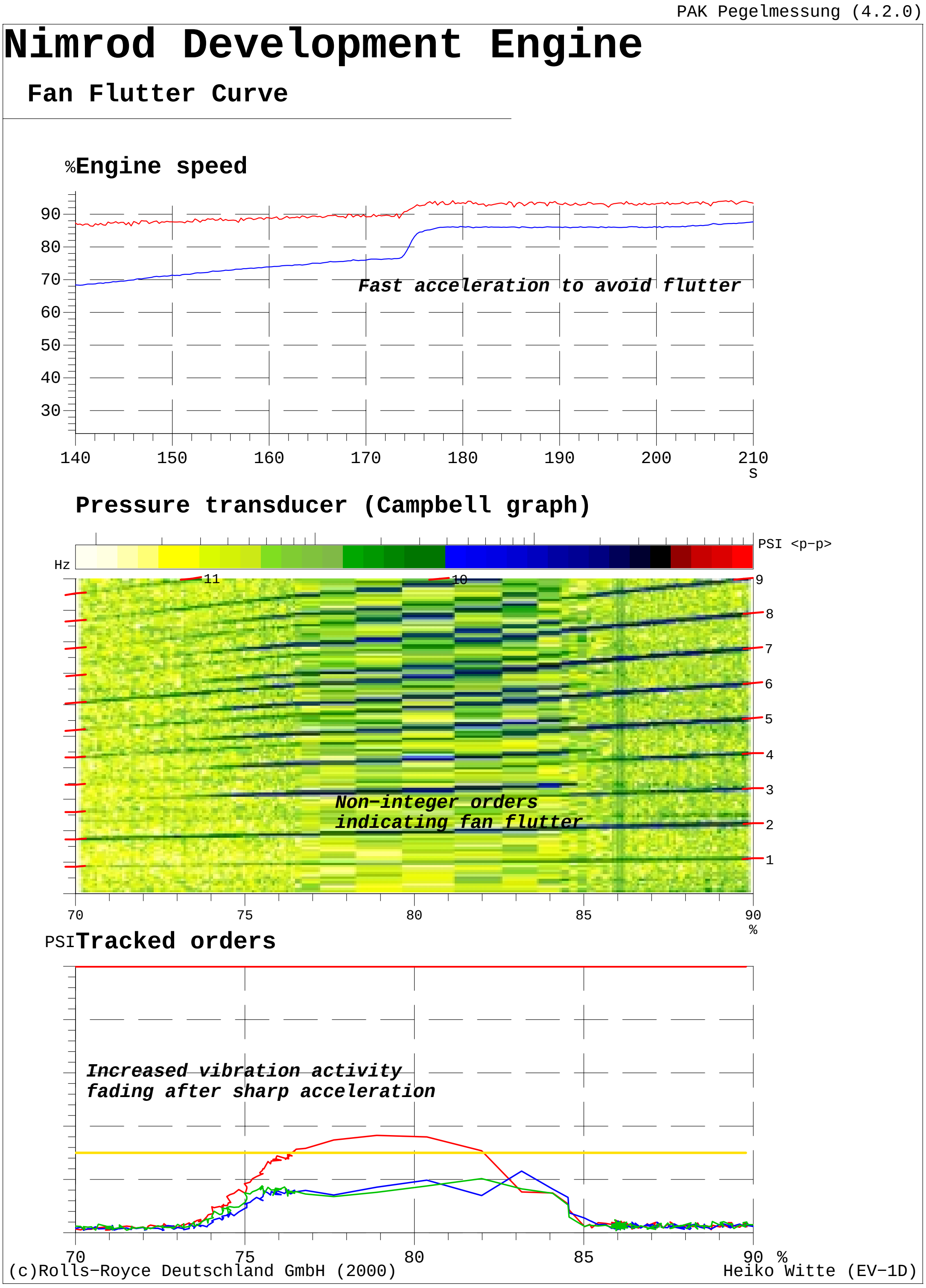

Figure 5.3: *Data analysis showing flutter occurrence on dynamic pressure transducers*

For this testing online-monitoring was required to identify commencing flutter vibrations and to ensure the structural integrity of the fan when the engine is operating near the fan flutter region. High instrumentation costs for strain gauge application and telemetry urged a more efficient measurement method. Thus pressure transducers were applied which react on flow pressure changes in the flow field induced by fan blades vibrating in resonance. The pressure transducers fitted in the fan casing measure increasing pressure fluctuations at specific frequencies related to speed harmonics based on the Doppler-effect. For fan flutter small amplitudes of fluctuations occur in between the harmonic engine orders and signify increasing blade vibrations of the fan blades.

For the monitoring task these small pressure amplitudes which are low compared to the harmonic engine orders had to be monitored and compared to sensitive limits. The limits were defined in a narrow order band in between the harmonic engine orders. Dynamic pressure responses are measured for a number of order bands because of several natural frequencies of the fan which are excited by flutter. During the test it was crucial for engine safety that for exceeding limits the test crew had to be alarmed immediately. Uncontrolled flutter vibrations are avoided by fast accelerating or decelerating the engine through the critical speed band of the flutter region. For redundant and comparable data three pressure transducers were monitored. Each parameter included limit definitions for four different order bands. In figure 5.3 an example is shown for an engine maneuver where fan flutter was detected. Within seconds the observed order bands increase from noise levels to significant values. While in the Campbell-diagram (figure 5.3) those responses do not seem to be highly significant they represent strong flutter vibrations of the fan. Vibration levels increase rapidly within a few seconds. The speed graph (see figure 5.3) shows increased acceleration starting immediately after the warning limit was exceeded. Flutter responses still remain strong but do not increase further. After passing the flutter speed band pressure fluctuations due to fan flutter fade.

As described above the monitoring system enables to detect the fan flutter region and to determine exactly its location in the operating map. With the information stored in the protocol file an immediate statement in regard of flutter onset and related speed information can be given. A quick data analysis can be achieved using online stored frequency data.

5.3 Monitoring vibration data during engine endurance tests

Within any engine development program cyclic endurance tests are mandatory. The test consists of a high number of in-service-maintenance cycles. Endurance tests are part of the certification of a new engine or modifications and may also be required by the customer. Here the task of online-monitoring includes the evaluation of structural vibration data of these accessories. Casing vibrations are monitored at several positions at the engine to give information about engine health and safety. Thus critical imbalances and malfunctions of the engine can be recognized and avoided. Characteristics of the components in regard of the LP and HP system are extracted. Trend graphs based on monitoring results help to recognize aggravations of engine parts or to evaluate reasons for any occurrences during engine running.

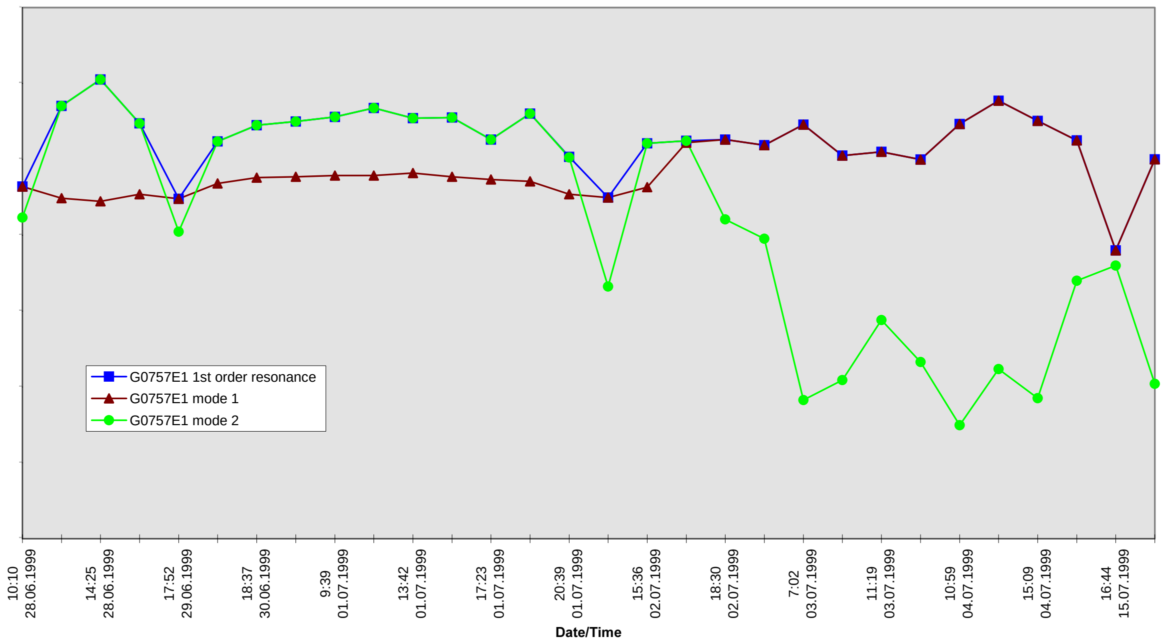

Figure 5.4: *Trend chart for fuel pipe vibrations based on monitoring results*

For the validation of a new fuel pipe standard strain gauge instrumentation had to be monitored during the cyclic endurance test. The Online-monitoring system provided an effective solution to observe natural frequencies with critical vibration levels when crossing first engine orders. As described in figure 5.1 bandpass limits were set to calculate the vibration energy within the natural frequencies. For the required continuous monitoring of the applied strain gauges during the whole test human monitoring of vibration signals was not necessary. The use of online-monitoring provided to be reliable and inexpensive. Warning and alarm limits have been set and exceeded limits triggered an alarm. The alarm signal would trigger the test engineer either to ask for vibration analysis support or to an immediate engine shutdown. To reduce the data amount only tracked bandpass and order levels were stored. A constant safety recording system where time data are stored is available if more comprehensive frequency information becomes relevant. Based on peak-hold values saved in the protocol files trend charts, as shown in figure 5.4, are delivered. Thus a safe and reliable engine testing and efficient data analysis is achieved.

6 CONCLUSION AND WAY FORWARD

The Online-monitoring system which has been developed and implemented at Rolls-Royce Deutschland provides a powerful mean for monitoring and analyzing measurement parameters. It is employed for different engine tests and requirements. It has been shown that complex measurement tasks are supported where data assessment and evaluation is required during the test. The combination between monitoring and analysis tasks is useful to combine limit monitoring and data evaluation to issue interim test results immediately after test completion. For simpler monitoring tasks an efficient semi-automatic operation of the system, where exceeded limits are signalled by an alarm sound, is applicable.

For the employment in aircraft engine monitoring the system provides powerful tools to be applied for all kinds of dynamic measurement transducers. The technical specifications are based on comprehensive measurement tasks in engine development while the graphical user-interface facilitates data assessment and interpretation.

An integration of an even wider channel number and the implementation of additional analysis features as mixed sampling rates will be further developments for the monitoring system. An effective cooperation between software development at Müller-BBM VibroAkustik Systeme and system application and definition at Rolls-Royce Deutschland will allow for user-oriented system adaptation.

7 REFERENCES

/1/ Ziegenhagen, St., Witte, H., Fischer, W.: Echtzeit-Signalüberwachung von Flugtriebwerken in der Erprobung; VDI-Berichte Nr. 1466(1999).

ENGINE VIBRATION MONITORING SYSTEMS

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Presented at the 20th International Symposium on Aircraft Integrated Monitoring Systems (AIMS) 23…25 May 2000 in Garmisch-Partenkirchen.

ENGINE VIBRATION MONITORING SYSTEMS

SUMMARY

The paper provides analysis of the evolution and state-of-the-art of the Engine Vibration Monitoring (EVM) systems for monitoring of the vibration of the aero engines.

The EVM systems evolved from mandatory monitoring systems, that were designed to monitor imbalance of the engine's rotating parts to the systems that have been combination of the monitoring and diagnostics/maintenance oriented system. Engine vibration spectrum is given with assignment of vibration sources. Typical structure of the EVM system includes vibration sensor(s), low noise vibration cable assemblies and signal processing electronic unit. Each of the hardware components and its position in the system is described in more detail.

Further are analysed signal processing techniques utilised in the existing vibration monitoring systems and supplementary functions implemented in modern digital vibration monitoring systems – cold rotor trim balancing, vibration acquisition for vibration data trending...

The functions and position of the future EVM systems in the Aircraft Integrated Monitoring Systems are analysed together with the advanced vibration analysis techniques and engine diagnostic function they can perform.

Combination of vibration analysis techniques with other engine condition monitoring functions for engine monitoring, diagnostics and prognostics is described.

Presented at the 20th International Symposium on Aircraft Integrated Monitoring Systems (AIMS) 23…25 May 2000 in Garmisch-Partenkirchen.

LIST OF CONTENTS

LIST OF SYMBOLS AND ABBREVIATIONS

1/REV	1 per Revolution
AC	Alternate Current
ACC	Accelerometer
ACMS	Aircraft Condition Monitoring System
AIMS	Aircraft Integrated Monitoring System
BB	Broad Band
BITE	Built-In Test Equipment
CFTB	Cold Fan Trim Balancing
CRTB	Cold Rotor Trim Balancing
DC	Direct Current

ECM	Engine Condition Monitoring
EICAS	Engine Indication and Crew Alerting System
EIVMU	Engine Interface and Vibration Monitoring Unit
EMU	Engine Monitoring Unit
EVM	Engine Vibration Monitoring
EVMS	Engine Vibration Monitoring system
FTB	Fan Trim Balancing
HP	High Pass
IEVM	Integrated Engine Vibration Monitor
LP	Low Pass
LRU	Line Replaceable Unit
MAU	Modular Avionics Unit
MCDU	Multi-Purpose Control and Display Unit
NB	Narrow Band
N1	Low Pressure Rotor speed
N2	High Pressure Rotor speed
N3	Intermediate Pressure Rotor speed
NVM	Non Volatile Memory
PM	Processing Module
RCC	Remote Charge Converter
RPM	Revolution Per Minute
RTB	Rotor Trim Balance

1. EVOLUTION OF THE ENGINE VIBRATION MONITORING SYSTEMS

According to U. S. Airworthiness Standard for Transport Category Airplanes, FAR Part 25.1305(d)(3) engine rotor unbalance monitoring is mandatory in commercial transports since 1974 as “an indicator to indicate rotor system unbalance” for turbojet engine powered airplanes [1]. Engine Vibration Monitoring Systems (EVMS) are practically exclusively used for such an engine unbalance monitoring.

A vibration of the engine is generated not only by engine rotor unbalance. There are also other sources of engine vibration. A typical vibration acceleration spectrum from an engine-mounted transducer would be as in Figure 1. It is clear that the information in this form can not be used for cockpit display and hence signal processing is required to extract the rotors unbalance data defined as one-per-revolution (1/REV) components of the vibration spectrum.

Early EVM systems have been available since late 1940s and used mostly analog circuitry for measurement of the broadband (BB) vibration of the engine.

EVM systems were later improved and allowed to separate rotor vibration. Such EVM systems were developed for aircraft L-1011, DC-10-30, B 727 and A300.

Significant improvement came with introduction of the first analog tracking filter system on B747 in 1978. This was the first system that approached the objective of the FAA regulations and introduced also the possibility of gathering Fan Trim Balance (FTB) data, on wing, from revenue flight. FTB was later implemented on A310 in 1984 and since that date on most other aircraft types.

In 1981 a digital version of the tracking filter was introduced for the B767.

At this point it was evident that a “generic” system approach was needed and such a system was used on the GIV, A320, F100 and almost all Boeing aircraft models. This generic approach has been recently expanded to the EVM unit that can cover all family of the Boeing aircraft except of B777. The unit can recognize type of the aircraft and engine and adjust the performance accordingly.

By 1986, the concept of a relatively sophisticated rotor imbalance monitoring system and its economical benefits had been largely recognized in the industry and it was realized that the inherent computing power of an effective vibration system could usefully handle other parameters as well.

This led Airbus in 1989 to the concept of the Engine Interface and Vibration Monitoring Unit (EIVMU) for the A330/A340. In addition to advanced vibration processing, this unit effectively functions as a multi parameter processor making the choice of engine type transparent to the rest the avionics suite. This generic concept allows use the same hardware platform for interface with different types of the engine on the same aircraft or family of aircraft with the same architecture of the avionics. The hardware platform is adapted to the specific engine by means of the software personality module.

Most of the modern EVM systems combine two functions in one unit - vibration monitoring and output of the engine vibration or alert signals for cockpit display and recording and acquisition of the maintenance information for trending of engine parameters and for rotor trim balancing (RTB). This separates the safety (flight crew) indications from maintenance data.

Recent trend in the aircraft avionics is integration of the different functional systems in the integrated modular avionics. Such Integrated Engine Vibration Monitoring (IEVM) modules have been developed for integration in the Primus EPIC™ Modular Avionics Unit (MAU) from Honeywell.

The technologies required for processing vibration signals over appropriate frequency bands and applying suitable algorithms to perform cockpit and maintenance related advisory functions have been refined to a cost effective level. In some instances a degree of diagnostic ability has been incorporated. Virtually no prognostic ability other than simply advising of a potentially damaging limit exceedance has been implemented.

2. STRUCTURE OF THE ENGINE VIBRATION MONITORING SYSTEM

Typical structure of the complete EVM system is described on the Figure 2 [1]. This paper addresses mainly on-board portion of the EVM system.

Typical EVM system comprises signal source, signal transmission, signal processing and interfaces with other on-board and off aircraft systems. Engine vibration is detected in most cases by two transducers installed on engine. The signals from transducers are transmitted to the processing electronics.

The processing electronics is in most EVM systems in the standalone electronic unit installed in the aircraft electronic bay.

2.1. Signal Source

Most important issues that have to be addressed during definition of the vibration signal source are transducer location, transducer mounting and transducer characteristics.

2.1.1. Transducers Location

Most EVM systems on the narrow body and wide body aircraft have two vibration channels. Transducer locations depend on engine type and there are two basic configurations of transducers on the engine.

The first configuration utilizes one transducers located inside the engine near the number 1 rotor bearing and the second transducer located on the housing (flange) of the turbine. This approach allows performing two plane balancing of the engine and distinguishing if the engine unbalance is generated by compressor or turbine modules of the engine.

The second configuration utilizes two collocated transducers or one dual output transducer located on the compressor case. This allows compare both channels and to check if the difference between them is within certain limit. Such an approach increases credibility of the vibration measurement, makes however more difficult the identification of the engine module that generates increased vibration.

Smaller engines for business or regional jets have mostly one transducer located on the fan/compressor or turbine case.

The mounting location of the transducer is extremely important and shall be established during engine development and verified during engine ground runs on the aircraft and during aircraft flight tests.

2.1.2. Transducer Mounting

Best solution for transducer mounting is a mounting provision – mounting surface on the engine. Special mounting brackets and attachment points are often used as well. The installed transducer should have first resonance at least three times the highest frequency of interest.

The mounting interface should be clean with metal-to-metal surfaces. Flatness should be within 0.0005 in/in. For higher frequencies measurement a roughness of the surface should be better than 0.032 mils [1].

2.1.3. Transducers Characteristics

Displacement and velocity transducers are practically not used in the modern EVM systems. Accelerometers are most common sources of the vibration signal.

Piezo-resistive accelerometers or their equivalent (circuited piezo-electric) are inherently limited in temperature and frequency response so are generally used only in suitable specific applications.

Piezo-electric accelerometer (further accelerometer) has high reliability (no moving parts) and high temperature capability (up to 1200°F). Accelerometers are the most widely used type of vibration transducer in the EVM systems.

The design of the accelerometer is usually customized for certain engine installation. Examples of the customized design of the piezo-electric accelerometers - surface mounted accelerometers with connectors, surface mounted accelerometers with integral cables, internal mounted accelerometer with integral cable, accelerometers with built in electronics, collocated / dual output accelerometers are on the Figure 3.

Important accelerometer characteristics have to be considered during selection and evaluated during development testing – sensitivity, resonance frequency, temperature response, transverse resonance frequency, operational temperature…

Experience proved that a "good" sensor installation is not necessarily transferable to a similar derivative engine. Even minor changes in operating speeds, mechanical design or even engine mounting can significantly change the total system dynamic response.

2.2. Signal Transmission

The driving factor of the signal transmission is that generally, the sensors are on the engine and the signal processing is performed in the aircraft electronics bay.

Accelerometer outputs very high impedance and low level charge signal. Hence special care must be taken to transmit this signal to the first stage of processing electronics. The most important issues of the signal transmission are shielding (single or double), grounding concept, cable properties, cable routing and clamping, connector shielding, connector strain relief, connector sealing and connector contacts. Cable with special low noise treatment is used for signal transmission between accelerometer and the first stage of electronics. Industry standard is to use differential piezoelectric accelerometer that has fully floating isolated ground and uses two connector pins for signal transmission.

The difficulties of the low level signal transmission can be avoided by placing the first stage of signal conditioning closer to accelerometer – inside accelerometer, integrally attach to accelerometer or in the standalone unit. The standalone unit is usually called Remote Charge Converter (RCC), is mounted in the pylon or on engine in order to convert the accelerometer signal to a less sensitive form as early as possible and to eliminate the need for low noise wiring in the aircraft. RCC can contain in one unit two to three channels. Remote installation of the signal conditioning electronics is limited by temperature limits of available electronic components.

2.3. Signal Processing

Accelerometers generate on the output an electric charge signal that is proportional in amplitude, frequency, and phase to the acceleration applied to the accelerometer basis in the sensitive axis. This charge signal has to be conditioned in the differential charge amplifier. Depending on the required frequency range on the BB maintenance output the vibration signal passes a low pass (LP) and high pass (HP) filters on the charge amplifier input. This will eliminate the saturation of the charge amplifier by the high frequency high amplitude signals generated by blade passage, gear mesh and also by low frequency effects. Good practice is to avoid any switching of the vibration signal before charge amplification.

The vibration signal is after charge amplification further filtered (anti-aliasing and HP filters), integrated to velocity and analog to digital converted. The further digital processing in the EVM unit is tracking filtering or spectral analysis. A measurement of the BB vibration is performed in the most of the digital EVM units in addition to tracking filtering or spectral analysis. This allows to continue monitoring of the engine vibration in case of loss of the rotor speed signal and also to correlate the narrow band (NB) and BB vibration in case of doubts the measurements are correct.

Tracking filtering provides synchronous narrow bandpass filtering. Central frequency of the tracking filter is controlled – slaved by engine rotor speed signal. Rotor speed signals are generated by engine tachometers. The tacho ratio between rotor speed and the frequency output from tachometer shall be considered.

Spectral analysis of vibration signal, in combination with rotors speed signal, allows extract from vibration the same information as tracking filtering. In addition spectrum provides lot of non-rotor related diagnostic information that can be used for detection and prediction of engine or components malfunction.

The vibration data acquired and conditioned by EVM are further used to perform two different task on the aircraft:

a) Monitor and output continuously vibration for cockpit display and recording in the FDR and trigger a cockpit alert,
b) Acquire and store vibration for maintenance functions.

The EVM units have also extensive BITE.

2.3.1. Monitoring functions.

The vibration signal, measured by accelerometer in terms of acceleration, is in the processing electronics integrated from acceleration to velocity in order to produce an approximately “flat” response with engine speed. Indications to the flight deck is further transformed by application of “normalization curves” for N1 and N2 (N3) which further adapted the velocity signals to the actual engine response characteristics. The objective is to produce an indication to the pilot in “Scalar Units” that had the same meaning on all aircraft types. The EVM unit output formats are usually AC or DC analog outputs or labels on digital buses like ARINC429, ARINC629, RS232 or MIL-STD-1553B.

2.3.2. Maintenance functions

These functions are usually not visible to flight crew and are stored during flight for post-flight analysis by maintenance personnel. The purpose of the information is to get economic benefits from On-Condition Maintenance (OCM), Engine Rotor Trim Balance [2], Predictive Analysis and reduction of the maintenance action by detailed diagnostics.

2.3.2.1. Engine Rotor Trim Balancing

Most of the new engines have incorporated provisions for placing balancing weights in one or two balancing planes. First balancing plane is usually in the Fan area; second balancing plane is in the Low Pressure Turbine area. The balancing weights for fan balancing are typically bolts and for LPT balancing special clips that can be clipped to turbine blades. The method is called Rotor Trim Balancing. When only Fan has provisions for such a balancing the method is called a Fan Trim Balancing.

Early methods required special on ground test of the engine and ground support equipment for acquisition of the required vibration information for RTB. Most of the state-of–the art EVM units use for rotor balancing method called, depending on the number of the balancing planes, Cold Rotor Trim Balancing (CRTB) or Cold Fan Trim Balancing (CFTB). This method allows collect data required for RTB during normal revenue flight and perform calculations of the required trim balancing on ground. This brings to the operator following advantages:

- The necessity of balancing is continuously monitored without additional operating costs,
- The balancing solution is more accurate. Data for CRTB solution are obtained in conditions of the real engine operation during flight and do not require other verification than evaluation of vibration measured during next revenue flight.

The CRTB requires information about rotor imbalance – value of the imbalance and angle position of the imbalance on the rotor in each balancing plane.

The value of the imbalance can be calculated from N1 tracked vibration signal.

The angle position of the imbalance on the rotor can be measured as phase between peak of the N1 tracked vibration signal and a reference point on the rotor defined by 1/REV reference signal. The 1/REV signal can be extracted from the speed signal if one tooth of the phonic wheel is higher or lower or the engine can have tachometer that generates independent 1/REV signal. Phase measurement has peculiarities related to the lead and lag phase and representation of the vibration signal [1].

The balancing algorithm further requires definition of the trim balance parts, engine trim balance locations (number and position of the holes and blades), engine shaft rotation direction, position of reference blade (hole) and sets of influence coefficients. All this information is specific for each engine/aircraft type.

2.3.2.1.1. Rotor Trim Balancing Data Collection

The EVM collects engine vibration and speed data at number of default, engine specific N1 target speed ranges for each engine. The speed ranges for triggering vibration data acquisition are normally defined by airframe/engine manufacturer. There are provisions implemented in the EVM that allow modification of the default triggering limits by aircraft operator.

There is number of engine balance flight histories stored within the non-volatile memory (NVM) of the EVM unit. For each new flight, the unit computes temporary accumulative averages for the engine vibration data. All measurement shall be made during a time window being at least 5 seconds in length and stability criteria are to be fulfilled. Typical stability criteria are: N1 speed is stable within ±1%, vibration amplitudes are stable within ±0,1 mils and vibration phase is stable within ±10 degrees [1].

After computing each value, the unit records new value if the new N1 speed is closer to the mid-speed of the target speed range than the previously recorded value for the N1 speed.

The Engine Balance Flight History can be read or erased either via cockpit MCDU or via front panel display and keys of the unit.

2.3.2.1.2. Rotor Trim Balance Computations

The EVM unit performs all computations necessary to determine balance corrections for one and two plane engine trim balance (Fan or Fan + LPT). In the case of accelerometer failure, the trim balancing will be performed using the available valid data. The unit uses only the data stored in the Engine Balance Flight History.

The EVM unit stores in NVM generic Fan and LPT balance coefficients for more than 10 engine N1 speeds, for 2 accelerometers and for 2 planes.

For both one plane trim balancing (FAN only) and two planes trim balancing (FAN and LPT) the EVM unit has the provision to specify installation of new balance weights and instructs the user to install additional trim balance weights on the FAN and/or on the LPT. If so required by the operator, then the Engine Balance Flight History for the specific engine can be erased. The unit keeps track about added trim balance weights and checks if the balancing solution does not exceed maximum allowed weight for Fan and LPT. When the limit weight limit is exceeded the unit warns about exceedance of the weight limit.

2.3.2.2. Vibration Data Acquisition for Maintenance Purposes

In addition to the collection of the data for CRTB the EVM unit usually stores up to 30 flight histories (1 history for each leg, i.e. for a total of up to 30 legs) within the non-volatile memory. For each new flight the following information is stored when maximum tracked vibration is detected per accelerometer for each engine separately: N1 tracked vibration, N2 tracked vibration, N3 tracked vibration, BB vibration, N1 speed, N2 speed, N3 speed, time elapsed. The storage of these parameters into the non-volatile memory is performed at the end of the flight.

2.3.3. BITE and Self Test

The EVM unit performs usually the BITE/Self Test operations in different conditions:

a) Power-up BITE,
b) Cyclic BITE,
c) Self-test (Manual BITE),
d) Continuous / functional tests.

As a rule a status word, containing error flags, is periodically transmitted on digital bus of the EVM unit. Each item tested during a BITE procedure is declared "FAILED" only after confirmation of the failure in n consecutive test routines.

There is also a circular buffer to store N failures within the unit NVM memory for repair/troubleshooting purpose only. There are provisions for reading and erasing the fault history buffer.

New EVM units should have implemented provision for on-board download of the new software version. This will allow continuous evolution and improvement of the EVM system.

3. FUTURE ENGINE VIBRATION MONITORING SYSTEMS

Increasing requirements to the reliability of the engines generate requirements to implement on the new engine programs or retrofit on existing engines more sophisticated analysis and processing of the vibration signals. These new requirements address in general:

- acquisition of harmonic and non harmonic vibration amplitudes and phases (other than fundamental harmonic vibration amplitude and phase of propeller/low pressure or high pressure rotor),
- acquisition of vibration signatures at both steady state and/or transients modes of engine,
- acquisition of more vibration diagnostics parameters at defined triggers for long term trending and prognostics,
- implementation of advanced signal-processing techniques that will provide bearings and gearboxes monitoring and diagnostics.

For meeting this requirements the part of the vibration spectrum, that is outside the frequency range of existing EVM systems and which is normally filtered for the reasons given earlier, has to be used and advanced methods of the vibration signal processing shall be implemented.

There is great potential for combining traditional unbalance monitoring used in aviation with the well known high frequency techniques required for bearing and gear diagnostics that are already widely used in industrial gas turbine applications.

Many of the advanced diagnostic techniques have also been recently evaluated in the Health and Usage Monitoring Systems (HUMS) for diagnostics of the engines, main gearbox and transmissions diagnostics of the rotorcraft.

4. MONITORING OF OTHER ENGINE PARAMETERS.

High computation power of the modern processing electronics allows integrate with vibration processing also monitoring of other engine parameters. Typical engine parameters and quantity of the parameters that are being processed in Engine Monitoring Unit (EMU) are in the following table:

No.	Parameter	Signal	Quantity
1	Rotor Speed	Frequency Signal	4
2	Temperature (thermocouple)	DC voltage	1…2
3	Temperature	RTD	5
4	Angle	RVDT sensor	3
5	Pressure Type I	AC voltage	8
6	Pressure Type II	DC current	1
7	Pressure Type III	DC voltage	6
8	Fuel flow	Time lag	1
9	Fire	Resistance	8
10	Oil Quantity	Potentiometer	1
11	Potentiometer	Voltage ratio	1
12	Vibration	Charge	2
13	Discrete Signal Input	Discrete	35…40
14	Digital Input Bus	ARINC-429	8
15	Discrete Signal Output	Discrete	15
16	Digital Output Bus	ARINC-429	2
17	RS232	Code	1

Such a combination of the EVM unit and EMU provides very interesting opportunity to combine different methods of engine condition monitoring within processing module of one monitoring unit. Most interesting is a combination of vibration analysis and fluid monitoring techniques. Correlation of these two independent methods should increase credibility of the diagnostics of the bearings and gearboxes.

Another example is the addition of temperature sensors directly in the accelerometer. Such a combination has provided an effective means for monitoring grease lubricated bearings where oil debris monitoring cannot be applied.

The principle of processing multiple level parameters within the same LRU has been proven and opens the way to combined parameter analyses to enhance the accuracy of traditional functions and/or to expand the number of relevant engine conditions that can be detected.

Close cooperation between the engine manufacturer and a system supplier having wide experience in complete system design will be an essential requirement in the implementation of the combination of advanced diagnostics methods of engine condition monitoring.

5. RELATION BETWEEN ENGINE VIBRATION MONITORING SYSTEM AND ENGINE CONDITION MONITORING SYSTEM

Engine manufacturers are more and more interested in the ECM that allows operate engine “on condition” and also helps to sell engines to airlines on the power by the hour basis. The architecture and location of the ECM systems on the aircraft are however not clear so far.

The ECM functions will be most probably implemented in the

- standalone ECM unit in the electronic bay of the aircraft, or
- module(s) in the Aircraft Condition Monitoring System (ACMS), or
- standalone ECM unit located on the engine, or
- module(s) in the Electronic Engine Control (EEC) unit or Full Authority Digital Engine Control (FADEC) on the engine, or
- standalone units for different techniques, located on the engine and in the electronic bay, or
- combination of all above mentioned options.

Each of the solutions has advantages and disadvantages and most probably the evolution of ECM will witness different architectures depending on the objectives of the ECM system.

There will be, most probably, on one side willingness of the engine manufacturers to develop an ECM system that will allow use the same hardware platform for the whole family of the engines or on all engine types of the engine manufacturer. The goal will be to use such an ECM system as a standard fit or option for OEM deliveries and also as a retrofit system for those engines that are already in operation.

On the other side the airframe manufacturers will, again most probably, prefer to have the same interface for all the engines on the same family of the aircraft. An example of such a successful solution is EIVMU that can be installed on two different types of aircraft with five different types of the engines.

Important issue for the future ECM systems will be relation between EEC unit and the ACMS. It is logical that in addition to its own specific inputs, an efficient ECM system will require access to parameters and data that are available from EEC or other engine or aircraft systems. Similarly, for the ECM system to be fully effective there will be a need to transmit processed engine condition data to the FADEC or other engine or aircraft systems.

6. CONCLUSIONS

Mechanical design and location of the transducer, mounting features and cabling have to be considered as an important, integral part of engine design and the sensor response has to be monitored during engine development testing. Provisions for the 1/REV signal are essential for all new engines if the Cold Rotor Trim Balancing has to be implemented.

There is new area of the advanced vibration monitoring and diagnostics that can provide further cost benefits to both engine manufacturers and aircraft operators. There is sufficient technology to implement such advanced methods on-board.

There is a challenge to expand ECM capabilities using combination of the various condition monitoring techniques within one ECM unit.

7. REFERENCES

[1] SAE E-32 Committee — AIR1839 A Guide to Aircraft Turbine Engine Vibration Monitoring System

[2] D. Burnell — Engine Condition Monitoring – Engine Systems and Aircraft Operational Effects, Proceedings of the 18th Symposium AIMS 1995

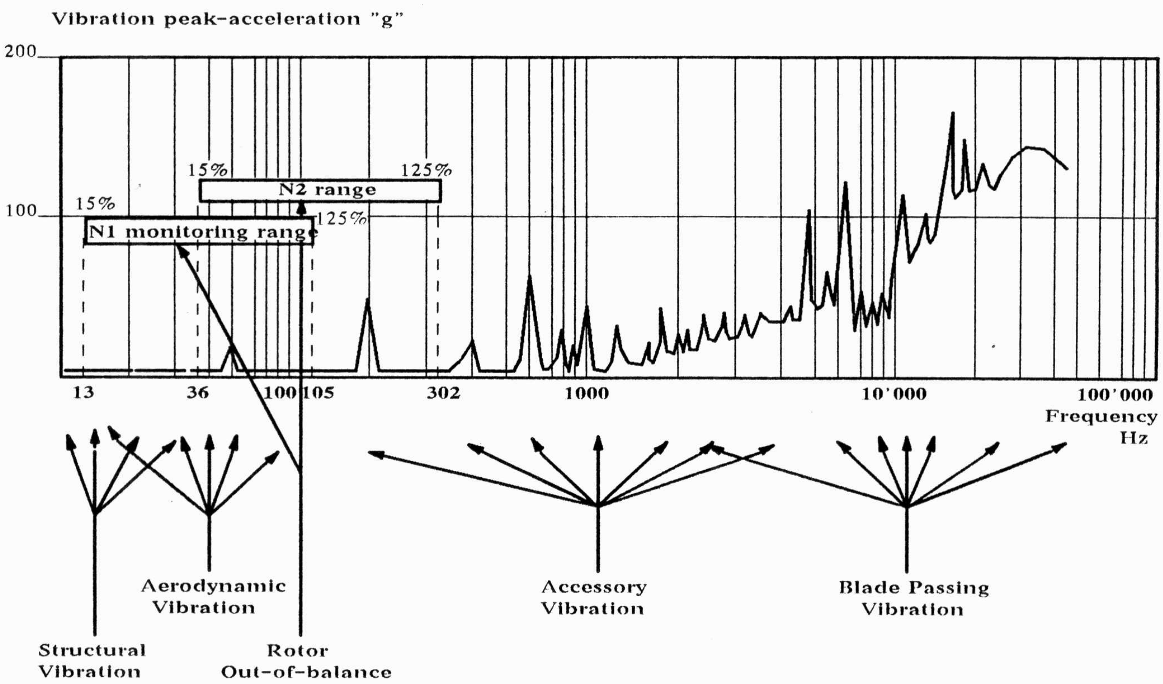

Figure 1: Typical Engine Vibration Spectrum

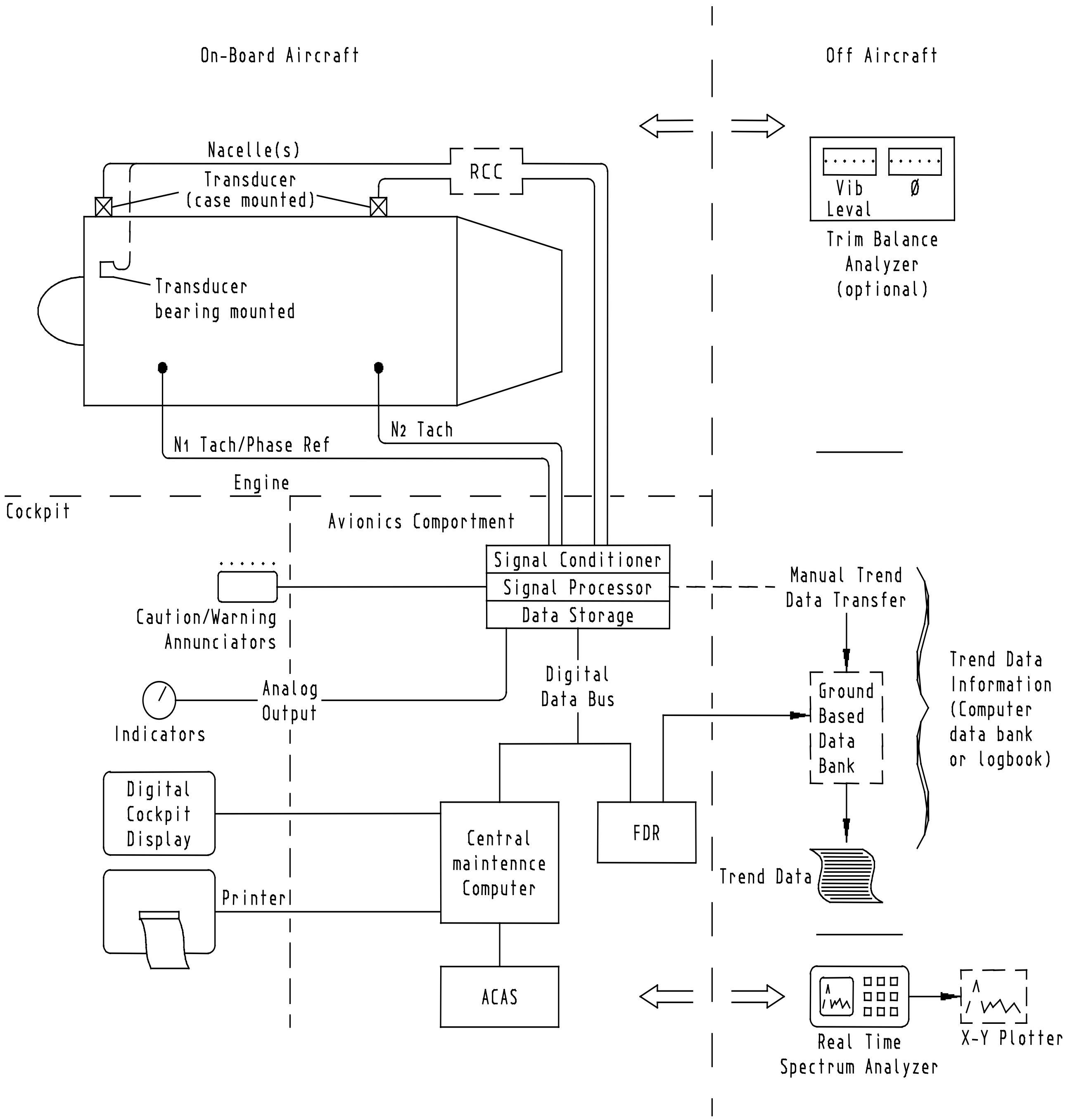

Figure 2: Typical Structure of the complete EVM System

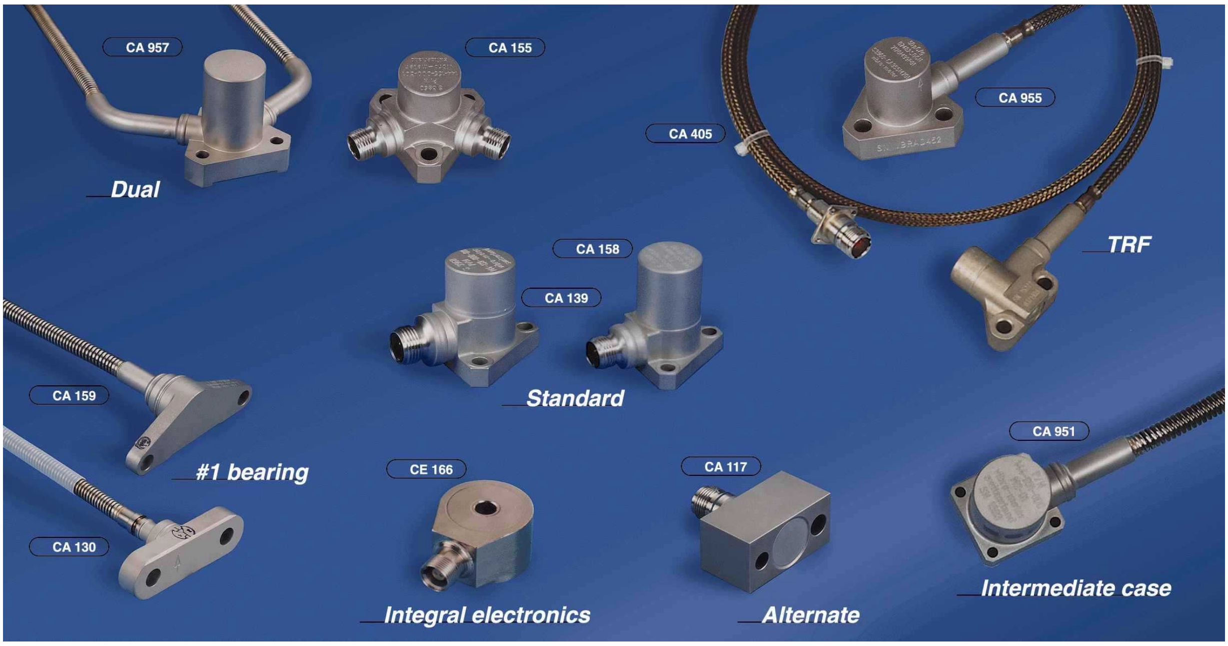

Figure 3: Accelerometers for Engine Vibration Monitoring

Figure 4: Engine Vibration Monitoring units

AN INTEGRATED CONDITION-MONITORING SYSTEM FOR GAS TURBINE ENGINES

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ABSTRACT

Hungarian Air Force initiated a long period Large Engine Monitoring System. The developed integrated monitoring system includes diagnostics based on thermo-gasdynamic parameters, vibration diagnostics and lube oil analysis.

The paper deals with the mathematical model based diagnostic system, developed for aircraft engines and installed on gas turbine power plants, too. The system – by monitoring a few parameters of the engine – gives opportunity to determine the trend of change of parameters and helps to localise the possible damage.

The vibration diagnostic system controls the engine in ready to flight situation, short duration test and evaluation. The paper presents some typical failure situations and results of trend analysis.

The lubrication system monitoring includes a day-by-day engine oil analysis and data collection for all aircraft engines in the country.

1. DIAGNOSTICS OF GAS TURBINE ENGINES BASED ON MATHEMATICAL MODELS

1.1. Principle of building mathematical models

The mathematical models are based on those of the component parts, which - as a matter effect - represent their computing programs of characteristics.

The following characteristics:

inlet duct $$\pi_{in}^* = f(M) \tag{1}$$

compressor $$\pi^*_c = f\,[q(\lambda)_c\,, n_c\,], \quad \eta_c = f\,[q(\lambda)_c\,, n_c\,] \tag{2}$$

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turbine $\quad q(\lambda)_t = f(\pi^*_t, \lambda_t), \quad \eta_t = f(\pi^*_t, \lambda_t)$ (3)

nozzle $\quad q(\lambda)_n = f(\pi_n)$ (4)

can be given at the form of polynomials reflecting the results of the preliminary calculations or the measurements, or else they can be inserted into the skeleton-program as subroutines, respectively.

In expressions (1)-(4) π^*, π - stagnation and static pressure ratio, $q(\lambda)$ - dimensionless mass-flow rate, n_c - corrected rotational speed, η - efficiency, λ - dimensionless velocity, M - flight Mach number. Subscripts c,t,n- are compressor, turbine, nozzle respectively.

The component-parts are connected by the skeleton-program which, at the same time, ensures the fulfilment of the laws of conservation.

The set of skeleton equations forming the mathematical model is constituted of the conservation equations of the possible mass- and energy stores found in the power-plant, as well as of the rules of controlling ($n=const$, $T=const.$, etc.).

There are three kinds of energy stores to be distinguished: the mass store, the mechanical energy store and the heat store. The mass stores represent the interspace between the components (compressors, turbines, nozzle) taking part in the conversion process of heat-to-mechanical energy. Mechanical energy is stored by the rotors, while heat energy is stored by the metallic parts and gases.

The general equation of mass stores accordingly will be

$$\dot{m}(j) - \dot{m}(j+1) = \frac{dm}{dt} \qquad (5)$$

$$m = \frac{pV}{RT} \qquad (6)$$

where $m(j)$, $m(j+1)$ - mass-flow-rates entering and leaving the store, m,V,T,p - mass, volume, mean temperature and pressure of gas in store, t- time.

The storage of the mechanical energy in the rotors will take place with the help of the following equation:

$$P_t(i)\eta_m - P_c - \Delta P(i) = 4\pi^2 n(i)\theta(i)\frac{dn(i)}{dt} \qquad (7)$$

where P_t ,P_c - actual turbine and compressor power, ΔP- power extracted from i-th rotor, η_m - mechanical efficiency, n- number of revolutions, Θ - moment of inertia.

The thermal energy to be stored in the metallic parts , and gases can be taken into consideration by the following relationship:

$$m_m c_m \frac{dT_m}{dt} + Vc_p \frac{d(\rho_g T_g)}{dt} = \dot{m}(j)c_p T(j) - \dot{m}(j+1)c_p T(j+1) \tag{8}$$

where m_m ,c_m ,T_m - mass, specific heat, temperature of metal parts, ρ_g ,T_g - density, and temperature of gas. The time delay occurring in the processes can be calculated for the individual cases on the basis of the following equation:

$$\tau = \frac{pV}{RT\dot{m}} \tag{9}$$

In the case of the twin-spool jet-engine shown in *Fig. 1* the set of equations will have the following form:

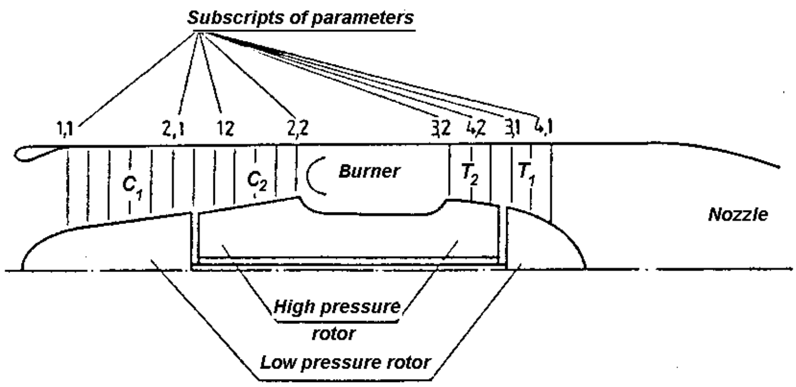

Fig. 1

Arrangement of twin spool turbojet

$$\dot{m}_c(2) - \dot{m}_c(1) - \Delta\dot{m}_c(1) = \frac{dm_{c1,2}}{dt} \tag{10}$$

$$\dot{m}_c(2) - \Delta\dot{m}(2) + \dot{m}_f - \dot{m}_t(2) = \frac{dm_b}{dt} \tag{11}$$

$$\dot{m}_t(2) + \Delta\dot{m}_r(2) + \Delta\dot{m}_s(1) - \dot{m}_t(1) = \frac{dm_{t1,2}}{dt} \tag{12}$$

$$\dot{m}_t(1) - \Delta\dot{m}_r(2) - \dot{m}_n = \frac{dm_{t,n}}{dt} \tag{13}$$

$$P_t(i)\eta_m - P_c(i) - \Delta P(i) = 4\pi^2\theta(i)n(i)\frac{dn(i)}{dt} \tag{14}$$

(i=1,2)

rule of controlling (15)

Similarly for a low bypass ratio turbofan with mixed exhaust streams (Fig. 2.) the set of equations have the following form:

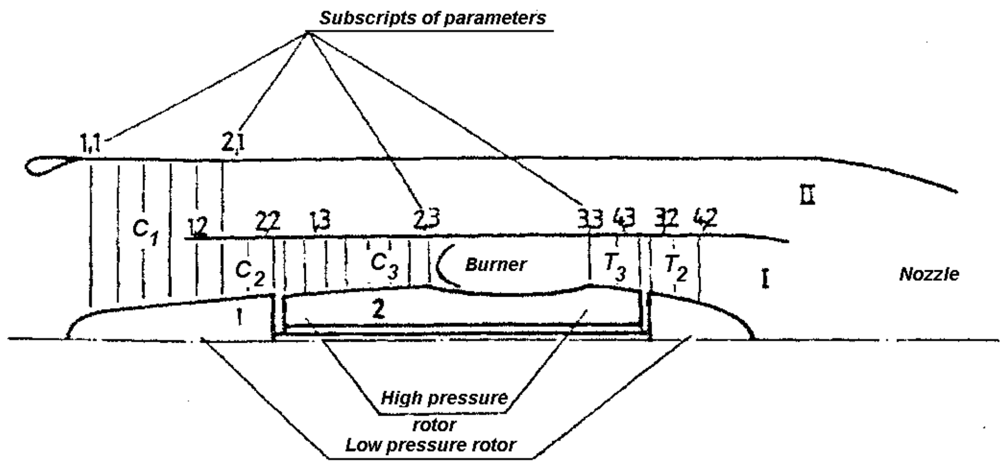

Fig. 2.

Arrangement of a twin spool low bypass ratio turbofan with mixed exhaust streams

$$\frac{\dot{m}_{c1}}{\alpha+1} - \dot{m}_{c2} = \frac{dm_{c1,2}}{dt} \tag{16}$$

$$\dot{m}_c(3) - \dot{m}_c(2) - \Delta\dot{m}_c(2) = \frac{dm_{c2,3}}{dt} \tag{17}$$

$$\dot{m}_c(3) - \Delta\dot{m}_c(3) + \Delta\dot{m}_s(3) - \dot{m}_T(3) = \frac{dm_b}{dt} \tag{18}$$

$$\dot{m}_T(3) - \Delta\dot{m}_r(3) + \Delta\dot{m}_s(2) - \dot{m}_T(2) = \frac{dm_{T2,3}}{dt} \tag{19}$$

$$\frac{\alpha}{\alpha+1}\dot{m}_c(1) - \dot{m}_{II} = \frac{dm_{m2}}{dt} \tag{20}$$

$$\dot{m}_T(2) - \Delta\dot{m}_r(2) - \dot{m}_I = \frac{dm_{m1}}{dt} \tag{21}$$

$$\dot{m}_I + \dot{m}_{II} - \dot{m}_n = \frac{dm_{mix}}{dt} \tag{22}$$

$$P_T(2)\eta_{m2} - \left[P_c(1) + P_c(2)\right] - \Delta P(2) = 4\pi^2\theta(2)n(2)\frac{dn(2)}{dt} \tag{23}$$

$$P_T(3)\eta_{m3} - P_c(3) - \Delta P(3) = 4\pi^2\theta(3)n(3)\frac{dn(3)}{dt} \tag{24}$$

rule of controlling (25)

Here, accordingly, m_c , m_t ,m_n - inlet, compressor, turbine and nozzle mass-flow-rates, Δm - air flow extracted from the compressor, Δm_r ,Δm_s - masses of air recirculated into the gas flow for cooling the rotor and stator blades, m_f- fuel flow rate, $m_{c1,2}$, $m_{t1,,2}$ - masses of gases in volumes between compressors and turbines, m_b - mass of gases in burner, $m_{t,n}$ - mass of gases in volumes between low-pressure turbine and nozzle, m_{mix} - mass of gases in mixing chamber, α - bypass ratio.

Within the set of non-linear, skeleton differential equation formed in this way, each of the equations constitutes a closed unit in itself. According to the flow path direction of gases, the equations can be completed with the use of the relationships provided by gas turbine theory.

In the models, the gas characteristics and the specific heat were determined as temperature- and composition-dependent in each case. In the equations, it was suitable to calculate with the normed, relative values - i.e. values related to the nominal duty - of the unknowns. In this way, the roots are yielded as of nearly identical order of magnitude, which - in turn - will take an influence on the convergence of the method, too, through the accuracy. The required partial derivatives were replaced with difference quotients.

The examinations carried out with the help of the full multi-store set of equations clearly show that in the case of the modelled gas turbine engines, it is sufficient to take into consideration the energy-storing effect of the rotors, only. The influence of the other stores, as well as that of the non-steady-state heat transfer between the metallic parts and gases is practically negligible, or else it can be compensated in the course of model adaptation. The same holds for the deviations due to the assumption of the originally quasi-steady-state process, and for the modifications in the characteristics occurring with transient processes.

Possible rules controlling the transient process will be

$$\dot{m}_f = f(t)$$

or with coastdown, it may have the form:

$$\dot{m}_f = 0$$

The set of non-linear differential equations obtained in this way was solved with the help of the fourth-order Runge- Kutta method. The modelling equation of the steady-state operational modes can be obtained with the zero values of the derivatives with respect to time. According to the rule of controlling in case of turbojet engine n(1)=const. To solve the set of non-linear equations, the Newton-Raphson method proved to be the most effective [1],[2]. The block diagram of mathematical model is shown in *Fig. 3.*

1.2. Adaptation of mathematical models to the objects modelled

To describe the modelled gas turbine equipment in the best possible way with the models built up in the way introduced above, it is required to carry out the adaptation of the model on the basis of the measurement results. This means that the different internal constants of the mathematical model are changed in a way that its set of equations should remain unchanged, however the solutions provided by it should draw nearest to the measurement results for this purpose the least square method is applied [1],[4].

The adaptation of transient processes, the different operational duties will represent the power-plant states at the different points of time. For performing the adaptations, processed (filtered and smoothed) results should be used. The aim of adaptation can be the correction of the compressor- and turbine characteristics. The convergence process of adaptation of three operational parameters (efficiencies) in case of two measured parameters for two operational duty demonstrated in *Fig. 3*. Block diagram of the computer program is shown in *Fig.* 4.

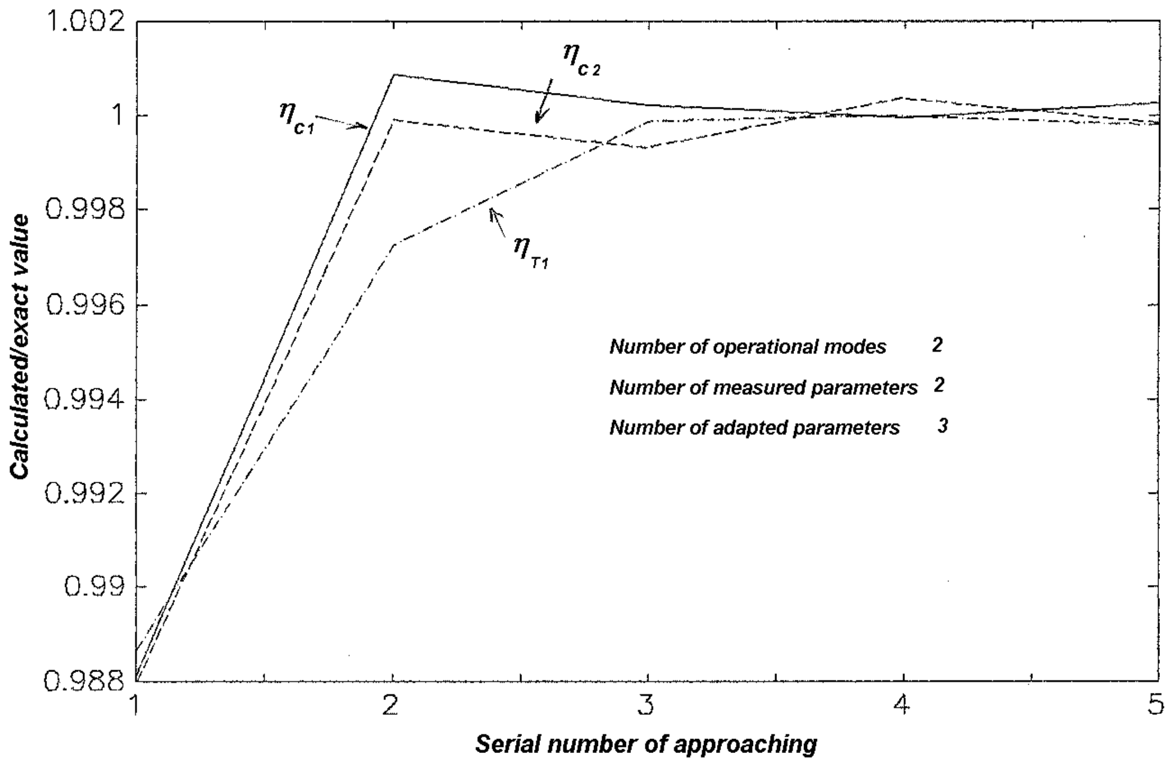

Fig. 3.

Process of convergence of operational parameters in adaptation.

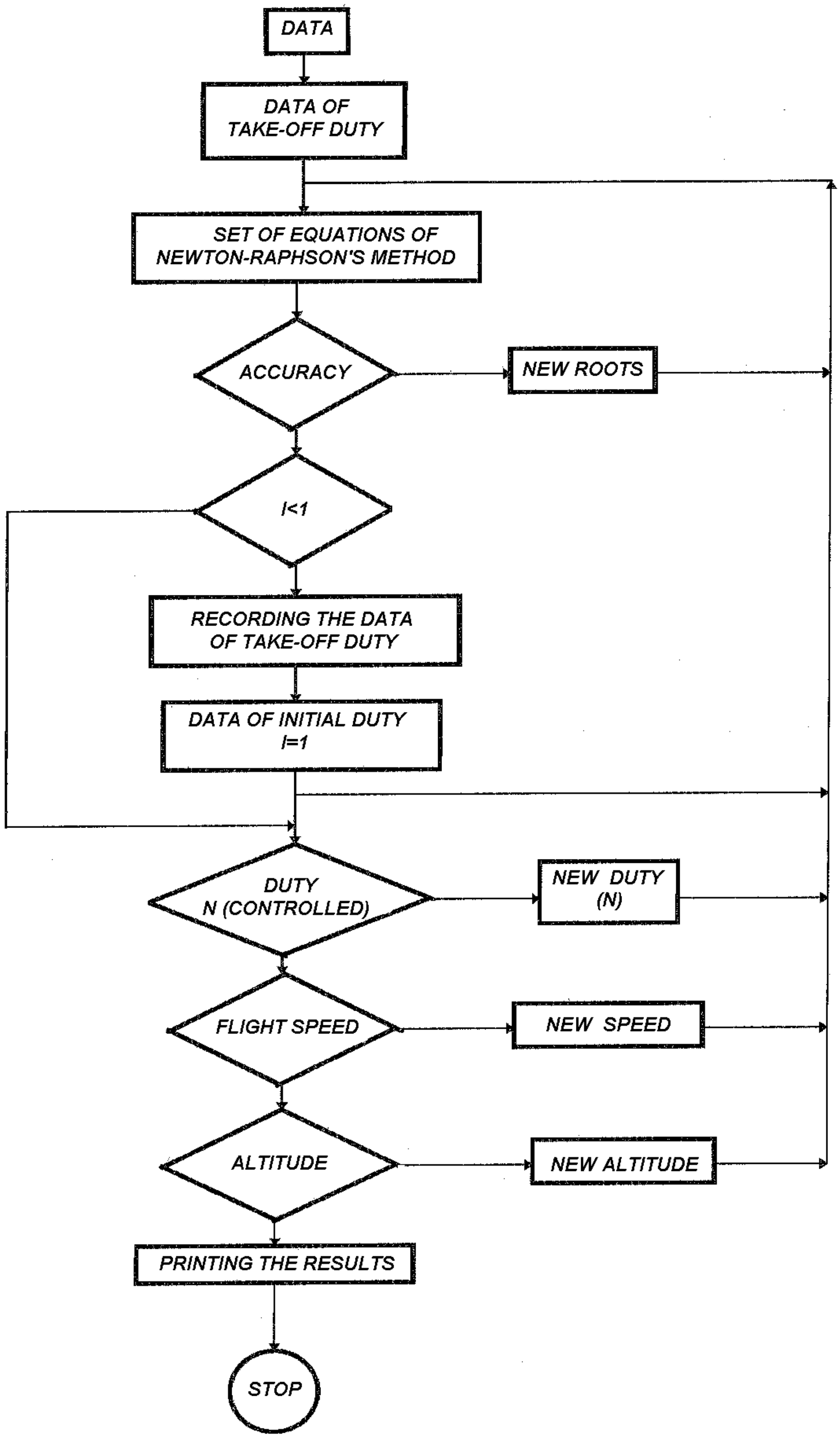

Fig. 4.

Flow diagram for mathematical model

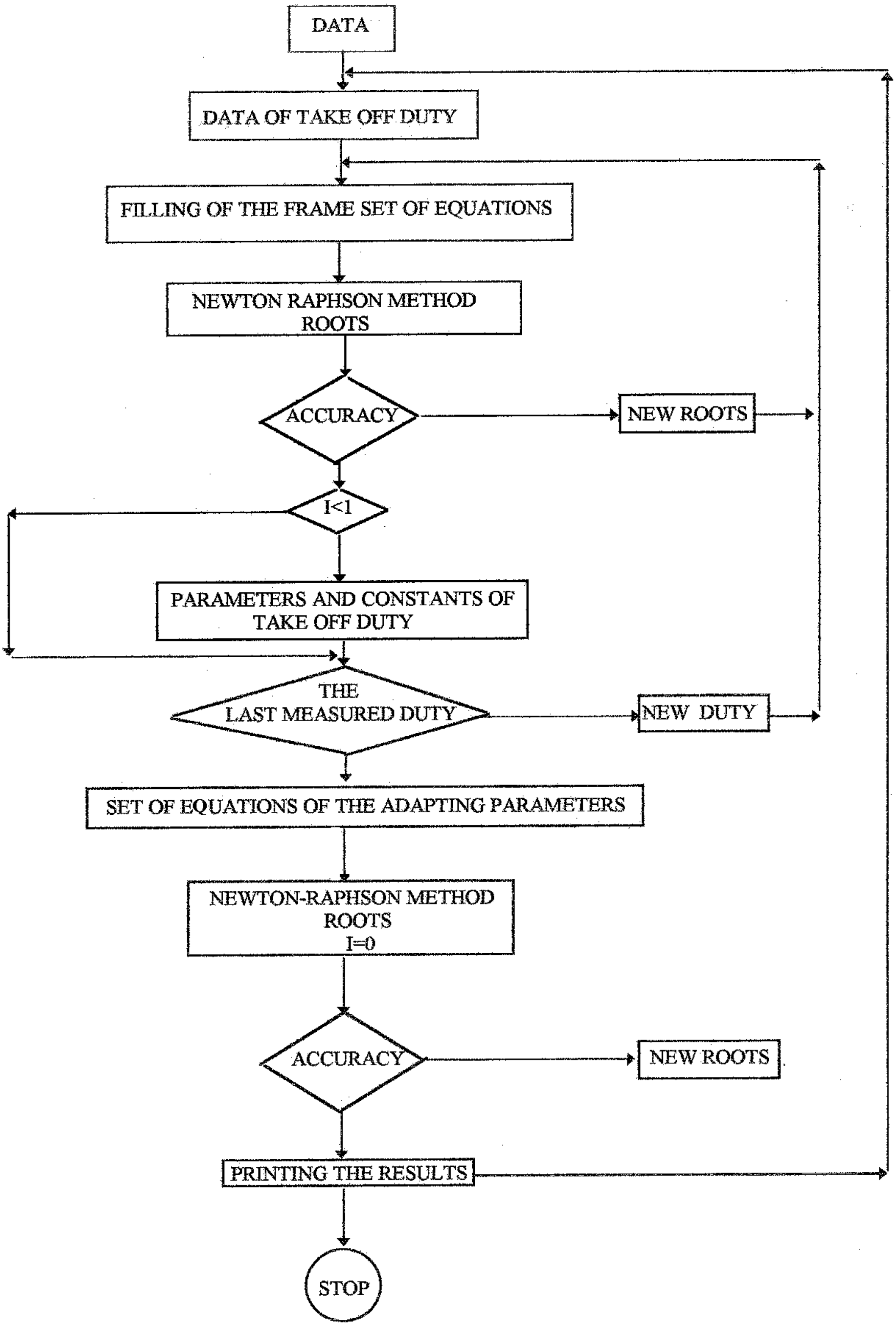

Fig. 5.

Block diagram of computer program of adaptation.

1.3. Application of models to analysing the behaviour of gas turbine power-plants

The adapted model is suitable for the examination of the processes taking place in the gas-turbine power-plants, for the determination of such parameter values which will not undergo measurement. The analysis of the behaviour of power-plants can be performed in the case of different simulated conditions, damage.

In *Fig. 6* the modification of the operating line of a twin-spool turbojet as occurred due to the reduction of compressor efficiencies can be seen on the maps of the low- and high-pressure compressors, respectively. In the Figures it is visualised clearly that the decreasing of the high-pressure compressor efficiency shifts the operating-line towards the surge line on the maps of the low- and high-pressure compressors.

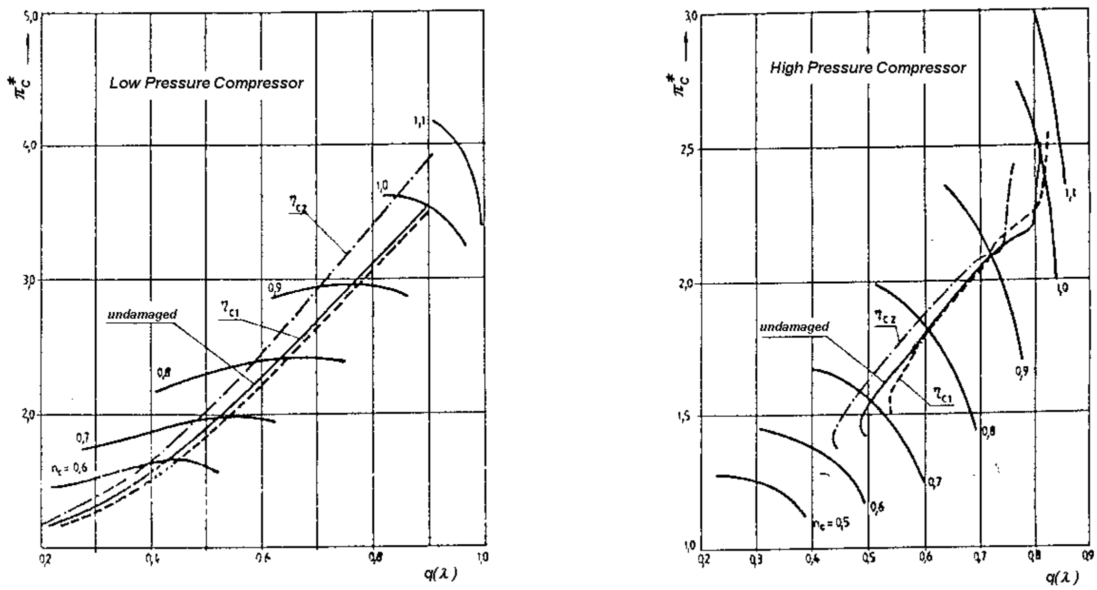

Fig. 6.
Effect of reduction of compressor efficiencies on operating line of a turbojet.

1. 4. Application of models in diagnostics

With the help of the elaborated models, examinations of parameter-sensitivity can be performed both in steady- state and transient operational duties. These examinations show that with the individual state characteristics (efficiencies, cross-sectional areas, losses etc.) the sensitivity of the power-plant parameters measured or measurable potentially will be different (*Fig. 7., Fig. 8.)*

On the basis of such examinations, the diagnostic matrix of the given engine can be generated [3].

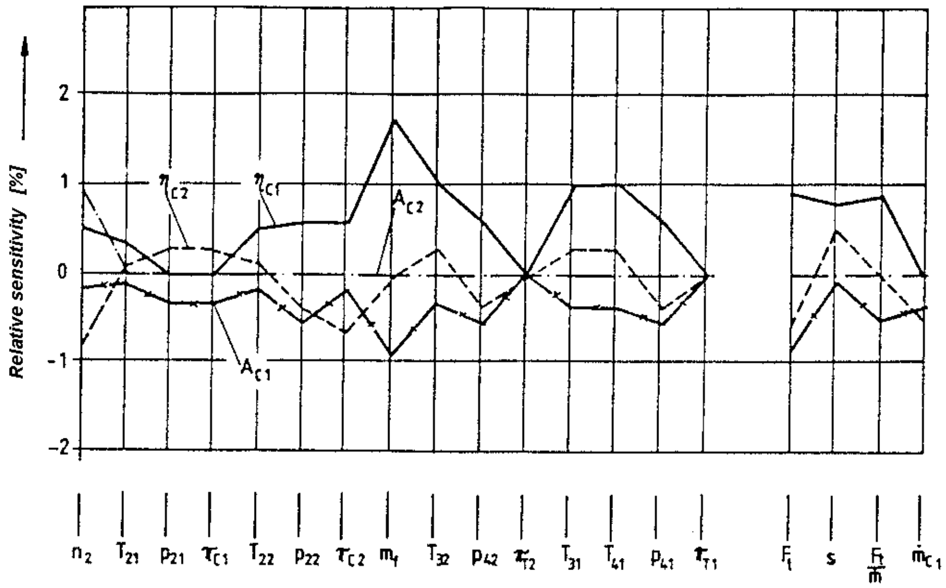

Fig. 7.

Relative sensitivity of turbojet parameters to compressor efficiencies and cross-sectional areas

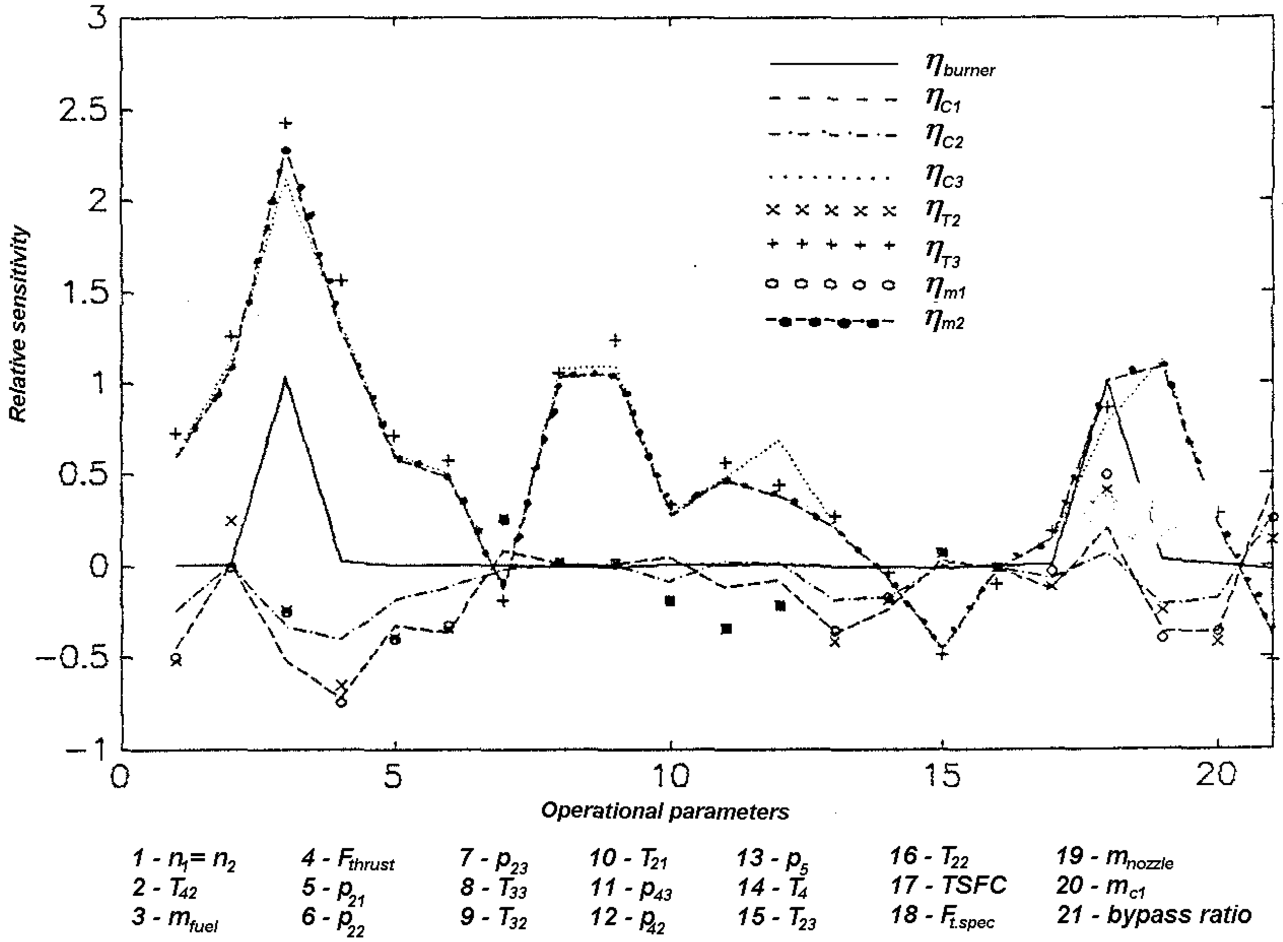

Fig. 8.

Effect of reduction of efficiencies on turbofan engine operational parameters.

Investigations shows, that the sensitivity of the individual parameters varies in a different measure on the different operational duties as well as the ratio of those variations relative to each other [1],[2],[3],[4]. Sensitivity parameters will change in case of transient operational modes according to steady ones.

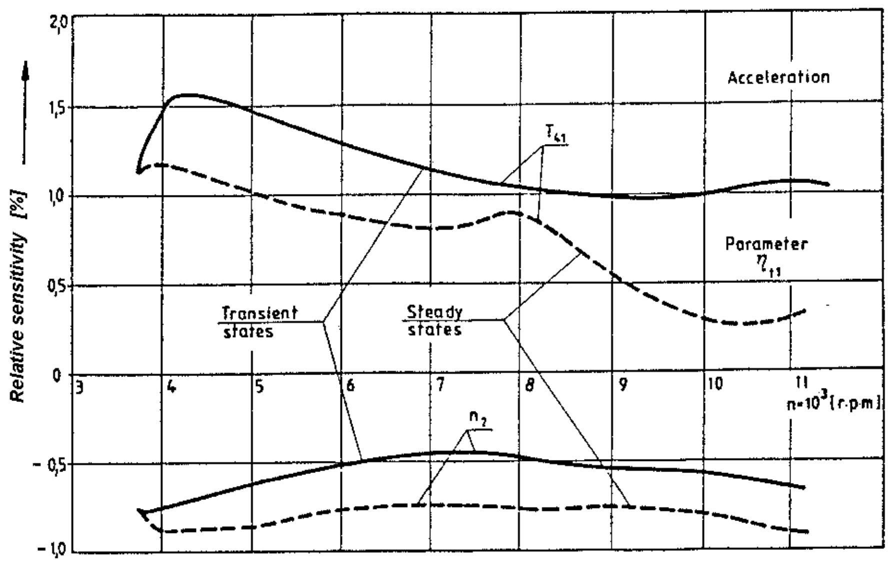

Fig. 9.

Comparison of relative sensitivities of T_{41} and n_2 to low-pressure turbine efficiency on steady and transient operational modes.

The change in the relative parameter sensitivities (T_{41} and n_2) is shown in *Fig. 9* occurring with the transient (accelerating) and steady duties of a turbojet in case of low-pressure turbine efficiency reduction illustrated versus high-pressure rotor rotational speed. Consecvently, a controlled transient process (acceleration or deceleration) can provide additional information about the state of the power-plant and has a diagnostic content [3],[5].

Inasmuch, the already adapted, undamaged (basic) model is adapted to the processed measurement results of the damaged power-plant with the help of the corresponding adaptation program [1],[2], then the cause of damage can be traced back more accurately. Naturally, every power-plant should have its own adapted model.

The adaptation is carried out by the program in a way that the characteristic efficiencies, losses etc. of the engine will be determined under identical conditions (nominal duty, standard environmental circumstances) on the basis of measurements performed on a wide range of operational duties. In this way, the variation of the given characteristic can be evaluated, the trend of variation can be followed after appropriate (e.g. exponential) smoothing.

$$\overline{\Delta x} = \varsigma\, \Delta x_i + (1-\varsigma)\, \overline{\Delta x}_{i-1}$$

where Δx - deviation of investigated parameters; $\overline{\Delta x}$ - the smoothed value of Δx; $\varsigma = 0,1 \div 0,3$; i - serial number of measurement.

The mathematical model gives an opportunity to determine internal engine parameters (temperatures, pressures, numbers of revolutions) independently of measured operational mode, or converts them to a chosen duty, so it will be possible to put down their degradation in time.

One can collect these data parallel and evaluate them with parameters of other diagnostic methods (e.g. vibration diagnostics, lubrication oil analysis) and the decision about the failure will has greater probability.

2. VIBRATION MONITORING OF JET-ENGINES

The on-board vibration monitoring systems give an approximate information on rotors. These units cannot separate the effects of rotors because of the measured statistical parameter of vibration (RMS or Peak) within a limited frequency range.

As a part of integrated engine monitoring concept a complete earth-based vibration monitoring system has been developed for detailed vibration control of rotating parts and housing.
The measured vibration data with other parameters (for example oil debris) give addition information on real condition of engines.

Users (service or maintenance departments.) demands against the control system:

- full vibration control during usual engine check,
- instant „go - no go” information,
- error recognition,
- trend analysis,
- simple handling and sensor installations.

The vibration monitoring system is based on a Bruel&Kjaer 3550 type multichannel analyser. Depending on engine type it makes possible real-time data acquisition or spectral analysis with 5-7 sensors. The measured data are stored in an IBM compatible portable computer. In the computer a special programme makes the spectral component identification, error recognition, etc.

Main functions of the key elements of the system:

Analyser:
- real time spectrum averaging
- time enhancement (calculation of rotor phase position)
- water-fall type dynamic spectrum

Computer: - data storage and engine identification
- error recognition
- trend analysis

A function of computer programme checks the correlations between measured data of different condition monitoring techniques (i.e. oil analysis and vibration). The error recognition is a typical self teaching function. A new error pattern might be integrated into the data bank with simple commands. The trend analysis serves to find connections between measured data and total flight time.

Typical engine loads during vibration monitoring: Base
app. 70% (under critical speed)
90% or 100%
100% to 90% or 90% to 85% cont. dropped
Base+10% to Base cont. dropped

Typical sensor positioning: Plane of rotor bearings (with axial and radial measuring direction)
Transmission system (2-3 sensors)

The vibration monitoring system has been checked and tested with R-25 engines.
Recently test measurements are carried out on RD-33 engines.

During the test measurements the vibration monitoring system gave useful informations on different engine problems that occurred for example in transmission and hydraulic system. The *Fig. 10.* and *Fig. 11.* show typical short term trend analysis of R-25 type engine.

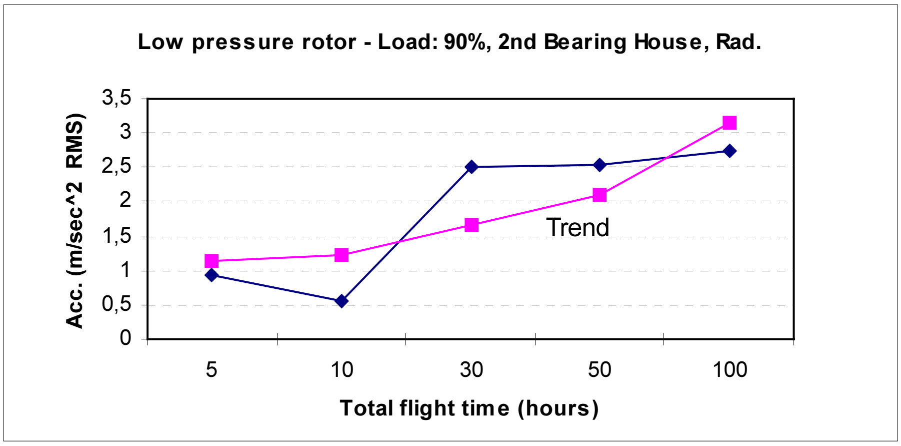

Fig. 10.

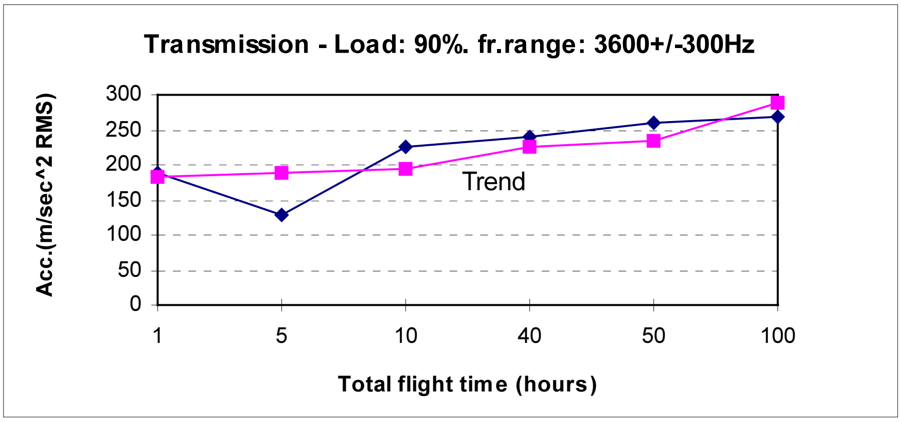

Fig. 11

3. OIL MONITORING

Oil monitoring is an effective method for assessing the condition of the oil itself and the components with which the oil comes into contact. In general, the oil properties indicate the quality of the oil and the failure of the components.

The basic equipment in the applied system is a BAIRD MOA type multielement oil analyser, which gives opportunity to identify concentrations of particular elements within the oil. This information is trended and used to identify components undergoing excessive wear conditions.

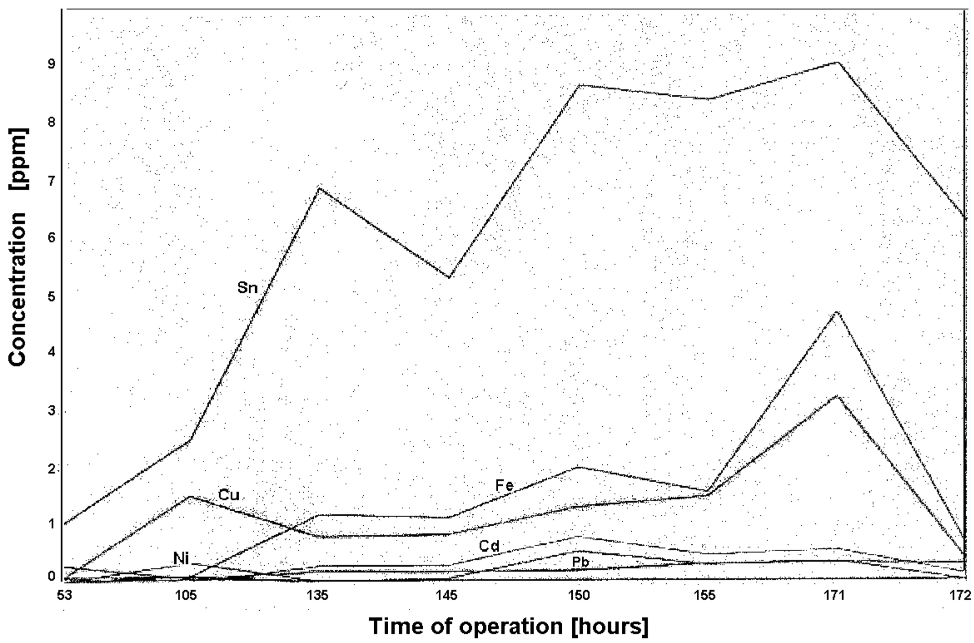

Fig. 12

Change in concentration of elements in oil for engine RD 33.

The increase in concentration of elements versus total time of operation for the gas turbine engine type RD-33 is shown in *Fig. 12*. (Data of measurements of HDF Aeronautical Eng. Dir.)

Samples are taken from engines periodically, following the diagnostic procedure. Results of analyses are collected and saved with other methods of monitoring in computer of diagnostic centre.

The warning limits of concentrations will be determined during operation time of the engines. We should take into account, that synthesis of results of individual diagnostic methods and state the diagnosis by them requires appropriate data base and diagnostic- and operational experience. The failure recognition is a self teaching process.

5. SUMMARY

This paper discusses an integrated monitoring system, which based on thermodynamic parameter-, vibration- and oil monitoring. It describes the modelling of transient and steady operational modes of gas turbine engines, provides some results of their analysis. The results of investigations shows valuable diagnostic content of change in parameters of steady and transient processes. Adaptation of adapted mathematical models of undamaged gas turbine engines to measured operational data gives possibility to determine the cause of damage.

The individual adapted mathematical model gives an opportunity to determine internal engine parameters (temperatures, pressures, numbers of revolutions) independently of measured operational mode, or converts them to a chosen duty, so it will be possible to put down their degradation in time.

As a part of integrated engine monitoring concept a complete land-based vibration monitoring system has been developed for detailed vibration control of rotating parts and housing during a given test sequence.

The lubrication system monitoring based on multielement oil analysis of samples which are taken from engines periodically, following the diagnostic procedure.

Collecting and evaluate themo-gasdynamic data parallel with parameters of vibration diagnostics and lubrication oil analysis, decision about the failure will has greater probability.

6. REFERENCES

1. SÁNTA I.: Mathematical Modelling of Gas Turbine Engines and Their Application in Diagnostics. Járművek,Mezőgazdasági Gépek 1985.11, pp.426-431. (In Hungarian)
2. SANTA, I.: Non-Linear Mathematical, Thermal Models of Gas Turbine Engines and Their Application in Operation, ICAS Proceedings 1990. pp. 2264-2270.
3. SANTA, I: Transient analysis of Gas Turbine Power Plants in Different Cases of Deterioration. Proceedings of Third Mini Conference On Vehicle System Dynamics, Identification and Anomalies, Budapest, 1992.
4. SÁNTA, I: Thermodynamic Modelling of Gas Turbine Engines and Application of models. Ph.D. Thesis, Budapest.1994. (In Hungarian)
5. CYRUS B MEHER HOMJI, RAKESH BHARGAVA.: Condition Monitoring and Diagnostic Aspects of Gas Turbine Transient Response. ASME Paper 92-GT-100.

CONDITION MONITORING SYSTEM BASED ON NON-INTERFERENCE DISCRETE-PHASE COMPRESSOR BLADE VIBRATION MEASURING METHOD

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INTRODUCTION

The paper present non-interference technique of turbo-machine blade vibration phase method, one of the interesting such complex jet engine diagnostic method as well as the for dynamic phenomena investigation of a running engine. The method is based on discrete blade vibration amplitude measurement and its numerical response analysis referred to the jet engine technical condition analysis. The described method is used in some units of Polish Air Force as SNDŁ-1b/SPŁ-2b SO-3 jet engine diagnostic system. This engine powers polish TS-11 "Iskra" training aircraft.

1. MEASUREMENT METHOD

The blades ring forced vibration and transient value of speed is measured using the discrete, precise measurement of time intervals between following pulses generated in a sensor by rotating blades (Fig 1).

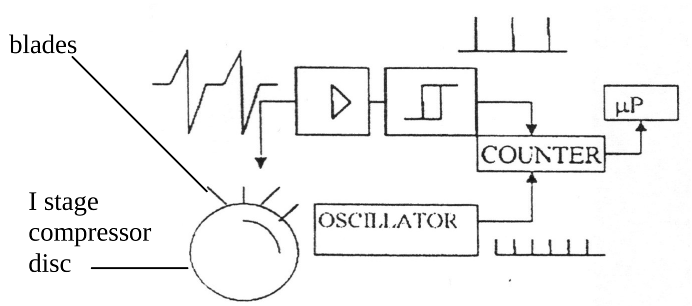


Fig. 1. Measurement method

2. MEASURING OF BLADES FORCED VIBRATION

The measurement of blades forced vibrations during engine run is performed for definition of:
- blades energetic (projected by flow disturbances) and technical conditions,
- mutual blades ring vibration correlation (revealing synchronous vibration phenomena).

Additionally exact information about real rotational speed of the rotor is got during measurement which allows for objective engine fuel system condition evaluation. The blades ring forced vibration and transient value of rotational speed was measured using the discrete, precise measurement of time intervals between following pulses generated in a sensor by rotating blades.

Characteristic feature of the described method of measurement, are:
- measuring sensor is placed in stator of the engine;
- there is a recording of the all blade vibration spectra of monitored compressor stage (nearly simultaneously);
- consciously advantage of using aliasing phenomena in amplitude blade vibration spectra analysis is is taken – i.e. theorem Nyquista-Shanona-Kotielnikowa (frequency of measured pulses is two to ten time lower than first flexural blade vibration mode);
- measured value (time) proofs high accuracy of measurement.

3. NUMERAL ANALYSIS OF BLADES VIBRATION

The effect of blade upon its generalised displacement is numerically analysed by considering blades as appropriately shaped, beam fixed to the rotor disc)self – likeness condition) in centrifugal force field. An actual flow disturbances level of running engine is evaluated by employing the linear dependence of disturbances with vibration amplitude (in elastic deformation range). Numerical procedures employ sectional statistical analysis to get average values of blade vibration amplitudes and standard deviation. The analysis results are compared with the engine type model see fig. 2 and 3. Fig 2. introduces projection of first compressor stage amplitude vibration spectrum model of SO-3 engine, fig 3 – amplitude vibration spectrum during foreign object dwelling on the first stage compressor stator blades. (Y-axis-amplitude of vibration, x-axis rotational speed of the engine). Effect of blade phase resonance characteristic project by low fluctuation of vibration amplitude (in dynamic pitch) was used in numerical procedure. Narrow band filtering of measured signal was implemented with use aliasing effect in numerical procedure. The core of blades technical condition analysis is based on comparing of the engine dynamic pitch projection to the model projection of an engine.

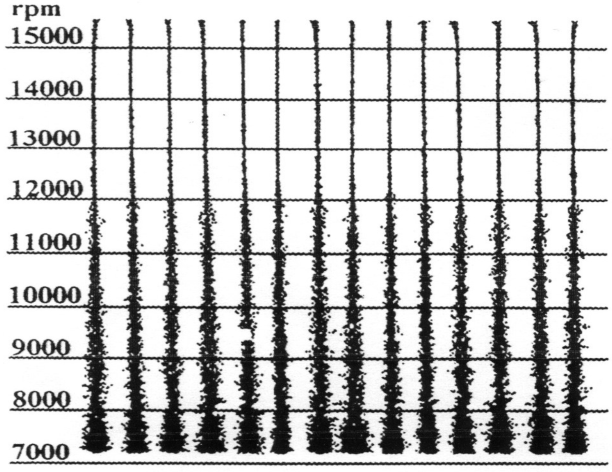


Fig. 2. First stage amplitude – phase compressor blade vibration model spectra of SO-3 engine

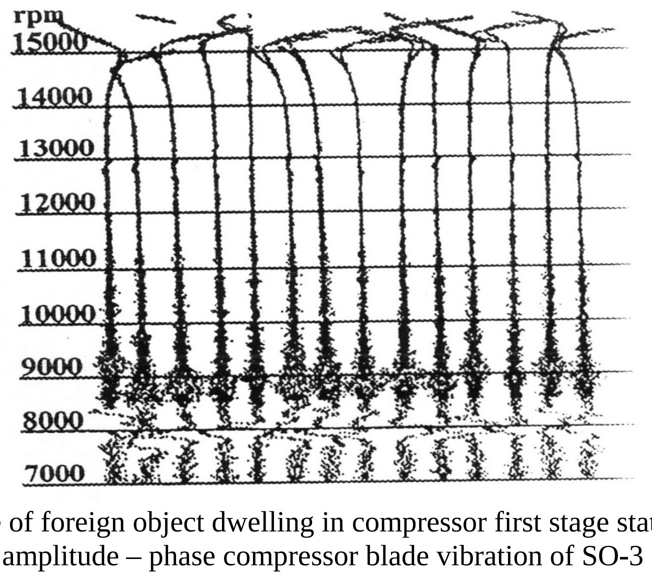


Fig. 3. Influence of foreign object dwelling in compressor first stage stator blades on first stage amplitude – phase compressor blade vibration of SO-3 engine

To get explicit identification of crack suspect or suspect blade an numerical procedure was used to estimate blade free vibration frequencies. Narrow band filtering of measured signal was implemented by low square (LS) method.

After blades free vibration frequencies estimation an identification process with factory or overhall basic blades frequencies is following. During test bench investigations the lowering of blade vibration amplitude was observed particularly in blade resonance excitations range. This phenomena is lowering of Q – factor the mechanical narrow band filter which the blade creatures. During cracks propagation process the crossection of the blade is decreasing which causes decrease of free vibration frequency "f_s" and blade dynamic frequency factor "B".

During engine run these changes project in decreasing of blades forced vibration frequencies "f_d" – fig. 4 (y-axis-dynamic pitch phase, x-axis rotational speed). For first compressor stage blades of SO-3 engine the dynamic changes were the most distinct near maximum of rotational speed of the engine in which the second synchronous excitation of resonance vibration occurs – fig. 5 (y-axis rotational speed, x-axis synchronous component of vibration amplitude phase change).

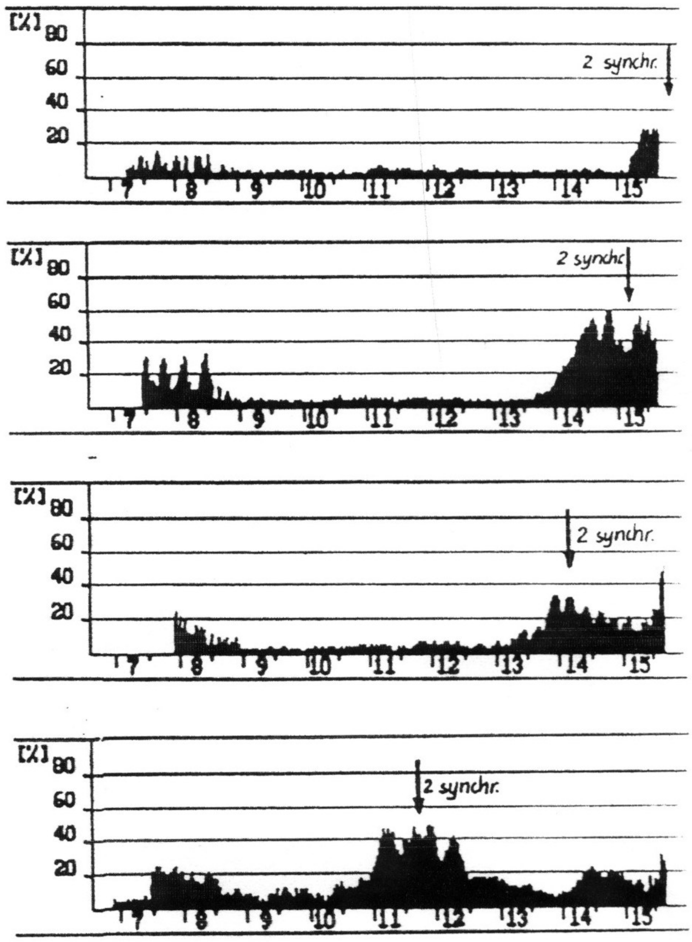


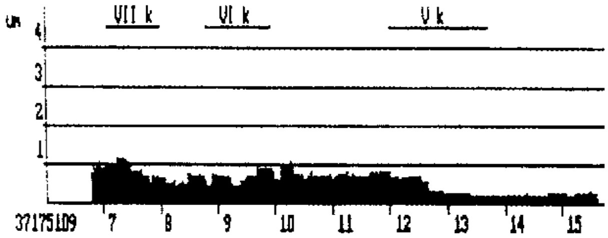
Fig. 4. Projection of blade crack propagation process in blade phase vibration spectra

4. NUMERAL ANALYSIS OF DYNAMIC LOAD OF ENGINE ROTOR BEARINGS

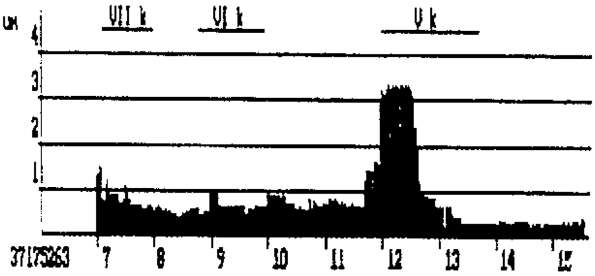
Blades amplitude and vibration frequency data having got during numerical analysis of all rotor blades were used for identification of some disadvantageous dynamic phenomenas time to time occurred during engine exploitation process. For this numerical procedure gives relation between:

- bearing system technical state and blade load effect,
- blade characteristic vibration and bearings dynamic load level.

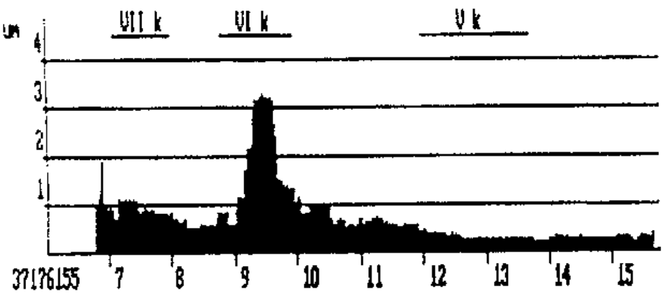
Procedure has sufficient sensitivity for revealing such phenomenas like synchronous and asynchronous rotor blades resonance vibration – fig. 5. The results of the analysis are projected conventionally in the shape of normalised radius and dynamic rotor unbalance phase (y-axis – engine rotational speed, x-axis – conventional, normalised dynamic unbalance radius).



a) normal level of blades vibration



b) excessive blades vibration level during engine acceleration
(close to compressor surge)



c) excessive blades vibration level during
engine deceleration

Fig. 5. Influence of blade vibration on the rotor dynamic
conventional disbalance level

5. TECHNICAL ESTIMATION OF FUEL SYSTEM CONDITION

Big frequency of sampling of the rotational speed of the engine (once for each rotor turn allows in very simple way to possess information concering transient value of rotational speed. Having in view relations between engine fuel consumption and rotational speed (related to real atmospheric weather conditions) an numerical procedure for fuel system technical conditions estimation was elaborated. Procedure compares real dynamic characteristic taken during non stabilised range of engine operation (acceleration and deceleration) with model dynamic characteristics – fig.6 (Y-axis – engine rotational increase of rotational speed). Having in view structure and principles of the engine fuel system work, procedure allows:

- all fuel system aggregates and the whole fuel system adjustment and control – fig. 7;
- early revealing of engine fuel system and fuel aggregates failures – fig. 8.

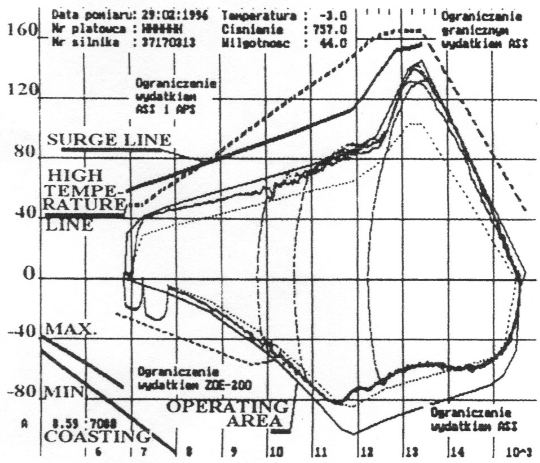


Fig. 6. Model characteristic of engine fuel system during transient condition of SO-3

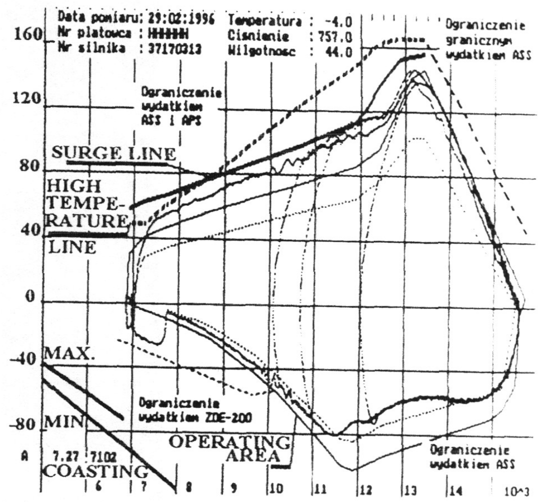


Fig. 7. The presentation of improper engine acceleration time adjustment – reduction of the margins of compressor stability (surge threat)

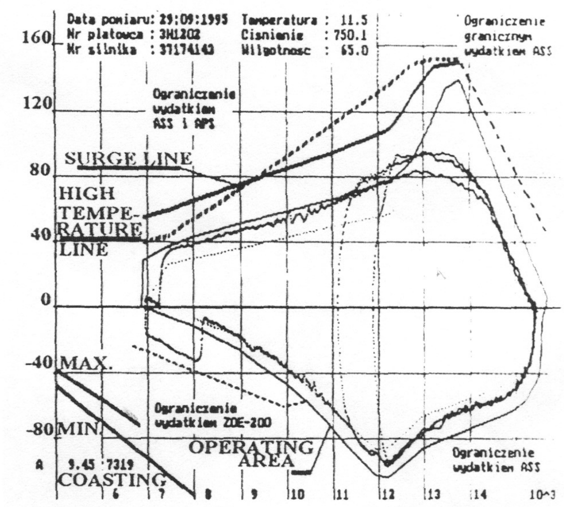


Fig. 8. Typical presentation of engine acceleration aggregate fault

6. CONCLUSIONS

1. Described method is of great importance for flight safety and may be recommended as a completion of the existing systems of diagnostic of turbine engines.
2. By using the discrete, non-contact method od measuring blades vibration and numerical analysis of blade vibration spectrum it is possible to conduct complex, monitoring of technical state of major engine components like:
   - compressor blades,
   - fuel system,
   - bearing system.

REFERENCES

1. Kudelski R., Szczepanik R. *Urządzenie do bezdotykowego wyznaczania amplitudy drgań w łopatkach sprężarki silników lotniczych*, V KNT "Metody pomiarowe w technice lotniczej", ITWL Warszawa 1987.
2. Szczepanik R. *Analiza i badanie przyczyn urywania się łopatek I stopnia sprężarki silników SO-3/SO-3W.* Sprawozdanie ITWL Warszawa 1987 (nie publikowana).
3. Szczepanik R. Badanie stanu drgań i naprężeń w łopatkach I stopnia sprężarki silników typu SO-3 w eksploatacji. Sprawozdanie ITWL Warszawa 1988 (nie publikowana).
4. Szczepanik R., Witoś M. *Quality Assessment of Diagnostic Non-Interference Discrete-Phase Method of Measuring Compressor Blades Vibration*, XIX Konferencja ICAS Anheim 1994.
5. Szczepanik R., Witoś M. *Wpływ procesu eksploatacji turbinowego silnika odrzutowego na żywotność łopatek wirnikowych*, VII KNT Przepływowe Maszyny Wirnikowe, Rzeszów 1993.
6. Szczepanik R., Witoś M. *Computer Aided Engine Condition System based on non-interference discrete phase blade vibration measuring method, SAE/SAS*, Symposium/Workshop Stockholm 1995.
7. Szczepankowski A., Witoś M. *Obiektywna kontrola stanu technicznego układu paliwowego turbinowego silnika odrzutowego typu SO-3 (SO-3, SO-3W, SO-3W22)*, Konferencja DIAG 95 Szczyrk 1995.
8. Szczepankowski A., Witoś. *Wpływ stanu technicznego układu paliwowego turbinowego silnika odrzutowego na drgania łopatek wirnikowych sprężarki*, Konferencja DIAG 95 Szczyrk 1995.
9. Witoś M. *Bezdotykowa metoda kontroli stanu technicznego łopatek wirnikowych sprężarki osiowej*, Konferencja DIAG 95 Szczyrk 1995.

Readout and Analysis

AIMS 2000
AIRCRAFT INTEGRATED MONITORING SYSTEM
FOQA PROGRAM WITH AIRCREW INVOLVEMENT IN INITIAL DATA ANALYSIS

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ABSTRACT

Flight Operational Quality Assurance (FOQA) programs record and analyse flight data from routine flights and attempt to pick out potential problems before they lead to accidents. Traditionally, maintainers handle the recordings and so these programs are operated without aircrew involvement. The BAE SYSTEMS approach to FOQA is based on our helicopter Integrated Health and Usage Monitoring System (IHUMS) operational philosophy. This differs from the traditional approach by involving the aircrew in the download process, including initial post flight analysis of data.

Initial processing displays detected FOQA events (flight exceedances and/or incidents). The aircrew is then able to comment on each displayed event. Additionally the crew is able to enter confidential incident reports. The advantage is that aircrew feedback is provided while the event is still fresh in their mind. In effect, this results in a combined FOQA and Incident Reporting Program.

CONTENTS

ABBREVIATIONS

ARINC	Aeronautical Radio Inc.
ASRS	Aviation Safety Reporting System
CHIRP	Confidential Human Factors Incident Reporting Programme
COTS	Commercial Off The Shelf
CQAR	Card Quick Access Recorder
DFDAU	Digital Flight Data Acquisition Unit
DFDR	Digital Flight Data Recorder
FAA	Federal Aviation Authority
FOQA	Flight Operational Quality Assurance
GSS	Ground Support System
HOMP	Helicopter Operational Monitoring Project
IHUMS	Integrated Health and Usage Monitoring System
NTSB	National Transportation Safety Board
OFDM	Operational Flight Data Monitoring
PC	Personal Computer
PCMCIA	Personal Computer Memory Card International Association
QAR	Quick Access Recorder
US	United States

1. OVERVIEW OF FOQA SYSTEMS

Flight Operational Quality Assurance (FOQA) is a tool for continuously monitoring and evaluating operating practices and procedures. FOQA has been in existence in a number of forms since the mid 1970's. A number of programs with a variety of different names are currently in existence, for example Flight Operational Quality Assurance or Flight Operations Quality Assurance (FOQA), Operational Flight Data Monitoring (OFDM), Global Aviation Information Network (GAIN), and the Helicopter Operational Monitoring Programme (HOMP). A common philosophy driving the development of all these flight operations monitoring programmes is an attempt to enhance aviation safety by identifying potential safety events and correcting them before they result in accidents or incidents. Implementation of FOQA is currently voluntary, but Aviation Authorities are recommending that operators implement FOQA.

Implementation of FOQA can provide an operator with quantitative information regarding adherence to their safe flight operational procedures and regulations.

FOQA programmes require flight data to be gathered from routine flights and downloaded onto a ground based replay and analysis system. The downloaded data is analysed for FOQA events, i.e. occurrences when normal aircraft operational limits are exceeded. Examples of typical FOQA events are; Flight Manual Speed Exceedances, High / Low Approach Speeds, Pitch exceedances etc.

FOQA programs will flag individual events and from a collation of these events from which trends can be identified. These trends may show potential problems with the way an individual performs certain tasks, or may indicate fleet wide areas of concern, such as potential problems with a specific approach at a particular airfield. Once such trends have been identified, relevant corrective actions can be instigated, and their effectiveness monitored.

Corrective actions may take the form of additional training for an individual or a change in operational procedures. Continual operation of the FOQA programme will allow the effectiveness of the applied corrective action to be measured.

1.1. Typical Implementation

A FOQA programme comprises four main activities;

- data acquisition,
- data replay and analysis,
- event assessment,
- corrective action and monitoring

The relationship between these elements is shown in Figure 1 - Typical FOQA Implementation.

1.1.1. Data Acquisition

Aircraft data for FOQA is obtained from the aircraft Flight Data Recording System. Typically this comprises a number of sensors, a Digital Flight Data Acquisition Unit (DFDAU) and a crash protected Digital Flight Data Recorder (DFDR). The DFDAU acquires signals from these sensors and from the aircraft digital data buses. This data is converted into a format suitable for recording on the associated DFDR.

The DFDAU and DFDR, together with the aircraft sensors, form the Flight Data Recorder System for Accident and Incident investigation purposes. The addition of a high capacity Quick Access Recorder (QAR), so called because the data is recorded on a removable media which can be easily removed from the aircraft, allows data to be gathered for FOQA and other programmes. The QAR records data from the DFDAU auxiliary data output for the duration of the flight. This data may be identical to the FDR data or may include data acquired specifically for FOQA or other applications. Most modern DFDAUs allow a different data frame with additional or higher rate parameters to be output on the QAR interface. The QAR recording media is removed from the aircraft on a daily or weekly basis as a maintenance task and sent to a central ground replay department where the data is extracted and analysed.

1.1.2. Data Replay and Analysis

The data from the QAR is downloaded into the ground replay and analysis system. This comprises a network of COTS Personal Computers (PCs) running the FOQA analysis software

The software transforms the raw flight data into engineering units for display and analyses the flight information for exceptional flight events on that flight, i.e. FOQA events. Typically, the data is analysed for between 50 and 80 events. When an event is detected, a report is generated. This contains details of the event together with relevant flight data prior to, during and after the occurrence of the event. The amount of pre and post event data is often user definable. All events are written to the operators FOQA database, and raw data is archived such that additional analysis can be performed if required.

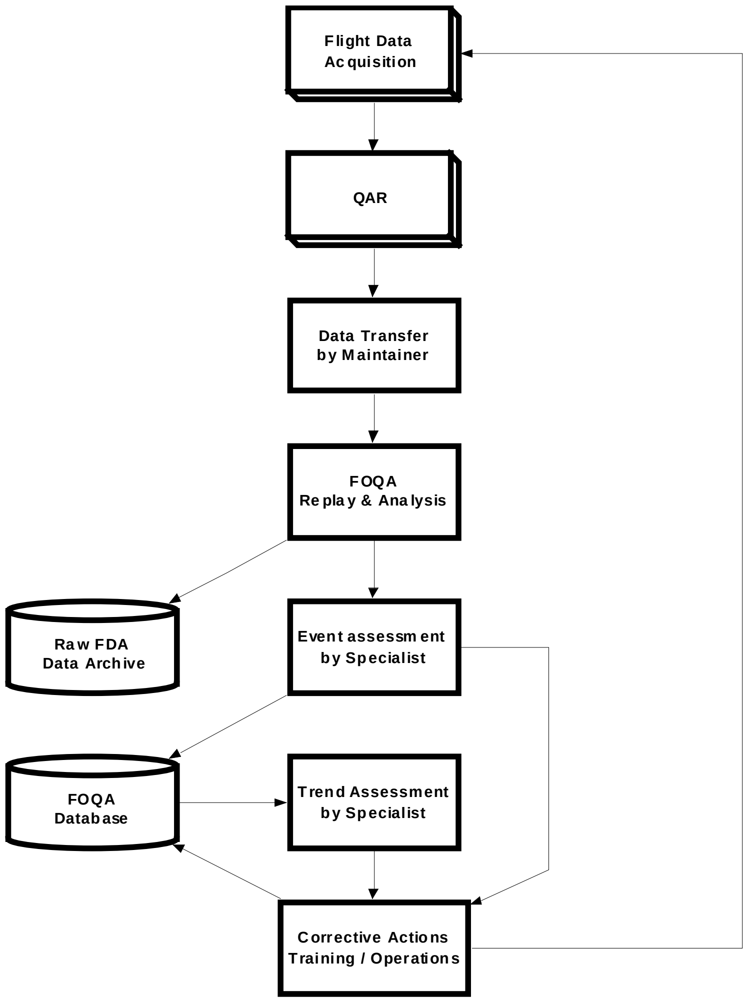

Figure 1 - Typical FOQA Implementation

1.1.3. Event Assessment

Each flight event report is then analysed by the fleet FOQA representatives in order to determine if any follow up corrective action is required. During this analysis, supplementary information surrounding the event may be obtained in order to help in the understanding of the problem. This may include weather reports, aircraft maintenance history and aircrew reports. If aircrew reports are required, the first contact with the crew may be anytime from two to five weeks after the occurrence of the event. Individual FOQA events are then stored in the organisation's FOQA database.

1.1.3.1. Database Assessment

The FOQA database is a collation of individual events, or discrete maxima values, which can be analysed statistically in order to extract and identify event trends. This analysis may group events by phase of flight, aircrew, aircraft, aircraft type, airfield, or any parameter available in the database. Such trends, for example the number of occurrences of a high rate of descent at a specific airport, may require the operational procedures to be reviewed for that specific location. This type of expert trend analysis is increasingly recognised as being fundamental to the successful implementation of safety improvement programmes such as FOQA. The process of extracting information from the data is a specialist task, requiring understanding of the FOQA system, the aircraft and their operation.

1.1.4. Corrective Action and Monitoring

Once all data has been analysed and assessed, the FOQA analysis team may decide that additional corrective action is required in order to reduce the chances of the specific event from recurring.

The ultimate aim of any FOQA type programme is to improve flight safety. An improvement in safety can be inferred from a reduction in the number and rate of measured operational incidents of events. By introducing corrective actions as a result of FOQA, a closed loop system is formed, which should result in an overall reduction in operational events.

The continual operation of the FOQA programme will inherently monitor the effectiveness of any corrective actions. Occurrence of detected events should then be seen to decline, providing all other elements remain constant.

2. OVERVIEW OF CONFIDENTIAL INCIDENT REPORTING PROGRAMS

A number of confidential aviation incident reporting programmes such as the UK Confidential Human Factors Incident Reporting Programme (CHIRP), NASA’s Aviation Safety Reporting Systems (ASRS) and others exist today. These are independent safety programmes which invite aircrew, maintainers, air traffic controllers, etc. to voluntarily report any events which the individual feels impacts aviation safety. The organisations running the programme would then de-identify and assess each report submitted, and enter the information into the database.

This incident reporting data is used to identify aviation safety hazards and deficiencies and to help develop improvements to aviation policy and regulations.
Periodically, the organisation issues confidential accounts of the incidents reported, thus spreading the experiences and lessons learned to the aviation community at large.

In the US, the FAA provides limited immunity from regulatory enforcement actions to an individual submitting a confidential report to the ASRS. This limited immunity is dependent on an event report being submitted to the ASRS within ten days of the occurrence.

3. BAE SYSTEMS' APPROACH TO FOQA

BAE SYSTEMS recognise the contribution made to flight safety by the world's major airlines through the use of FOQA. We seek to encourage the application of these techniques by regional and smaller airlines but believe that a new implementation is needed to reflect the scale of operations and minimise the cost of the operation.

3.1. Involving Aircrew in the Data Analysis

As has already been stated, with typical FOQA programmes, if a crew report is required to gain additional information about an event, the delay between the event and initial contact with the crew can be as much as five weeks. Individual crewmembers will have flown many more hours with a number of different crews and different routes by this time, so the circumstances surrounding the event may be a little hazy.

BAE SYSTEM's implementation of FOQA involves the crew during the initial data analysis phase. This takes place as part of the crew's post flight activity immediately after the flight. Crews are required to add a narrative comment to each detected event. The delay between the flight and data analysis is therefore minimal and this ensures that the crew will remember the circumstances surrounding an event. This method of reporting ensures that aircrew reports are gained in a timely manner and therefore incidents requiring immediate action are identified as early as possible.

3.2. Combining FOQA and Incident Reporting

Additionally the aircrew can use the system to enter confidential incident reports, which are unrelated to the events generated from the analysis of the FDA data. Again this incident reporting takes place immediately after the flight, while the issue is still fresh in the mind of the individual.

The FOQA analysis details "What happened" while the crew reporting gives an insight into the "Why the event occurred". Incident reporting programmes have been shown to be indispensable when identifying and assessing safety related incidents. They are particularly useful when analysing incidents involving human factors such as crew fatigue.

3.3. Data Transfer

Our approach to data transfer requires the installation of a BAE SYSTEMS Card Quick Access Recorder (CQAR). This is a Quick Access Recorder, which has been designed to be cockpit mountable. The unit is secured in place by two quarter turn camlock fasteners arranged in standard ARINC 601 rail format. The unit is shown in Figure 2 - BAE SYSTEMS Card Quick Access Recorder.

The CQAR interfaces to the Auxiliary Output of the DFDAU. It is compatible with both ARINC 573 and ARINC 717 Flight Data Acquisition Units. The unit can accommodate data rates of up to 512 words per second, selectable via programming pins in the unit's rear aircraft connector. Data is stored on a commercial Type II PCMCIA AT Flash memory card. Therefore, the data capacity of the unit is limited only by the size of the card utilised. Current cards are available in a range of sizes from 20 Mbytes to 800 Mbytes.
(800 Mbytes equates to approximately 1800 hours of flight data at a rate of 64 12-bit words per second!)

Data is stored on the card in a DOS / Windows NT compatible file system, so can be read using any standard windows PC which has a Type II PCMCIA card slot.

Additionally, the CQAR has an RS-232 / 422 interface, which can be used to store specific flight or event reports generated by the DFDAU. This serial interface operates simultaneously with the flight data interface.

Figure 2 - BAE SYSTEMS Card Quick Access Recorder

3.4. Crew Involvement

The CQAR has been designed to be cockpit mountable and is therefore ideally suited to FOQA programs which involve the aircrew in the operation of the programme.

The revised BAE SYSTEMS process is shown Figure 3 - Post Flight FOQA Analysis, and described below.

3.4.1. Pre-Flight

As part of the pre-flight procedure, either the Captain or First Officer will obtain a blank FOQA PC memory card, which they will carry to the aircraft and insert into the CQAR as part of the pre-flight process. Data will be automatically recorded onto the card for the duration of the flight.

3.4.2. Post-Flight

At the end of the day's flying, the card will be removed from the CQAR as part of the shut down procedure. The designated crewmember will then return to the operations room to perform the initial post-flight data analysis.

The post-flight data analysis procedure is an interactive process used by aircrew to transfer and validate information from the FOQA data card to the Ground Support System (GSS). Event detection is performed on validated data.

The Captain or Co-Pilot will log onto the GSS and insert the FOQA data card. They will then be required to enter elements of documentary data to characterise the individual download. Documentary data for the complete flight includes:

- Captain's Name
- Co-Pilot's Name
- Take-off Date
- Flight Reference
- Aircraft Registration

Documentary data for each sector includes:

- Number of Passengers
- Number of Crew
- Fuel Load
- Baggage Weight

This information, together with the aircraft zero fuel weight and fuel burn rate, allows the aircraft weight to be determined at any phase of the flight. Aircraft weight is required for assessment of take off and landing events and aircraft performance determination.

If no documentary data is included, the system provides default documentary data, alternatively it may be tailored to interface to an airline's computer system in order to extract the passenger, crew, baggage and fuel weights.

The flight data is written to a raw data archive prior to processing, such that it can be subsequently accessed by FOQA analysts if required.

The recorded flight data is converted into engineering units. This data conversion caters for all analogue and discrete signal types examples of which are listed below.

Analogue:

- Linear
- Polynomial
- Synchro
- High Accuracy parameters (Pressure altitude, Longitude and Latitude)

Discrete:

- Normal
- Inverted Logic
- Multi-State Logic

During this conversion, the flight data is checked for validity on a frame by frame basis. Typical validity checks will involve verifying that the ARINC 573 or 717 synchronisation words are present in the correct locations in each data frame. Any data frames with suspect data are rejected from further processing. The system generates a summary report identifying the number of valid and invalid frames, which is retained for inspection for diagnostic purposes.

The system then identifies the different phases of the flight and calculates derived data to complement the recorded flight data and to provide a Flight Identification Record. This additional data is introduced using;

- the user entered documentary data
- additional data stored within the database such as aircraft type, zero fuel weight
- derived parameters such as all up weight and aircraft weight during each flight phase, and take-off and landing locations

The Flight Identification Record comprises secure and de-identified data. Secure data is only made available to defined personnel within the FOQA programme in extenuating circumstances, for example, when further crew contact is required.

The system then examines recorded flight data to identify exceptional flight events which occurred on that flight.

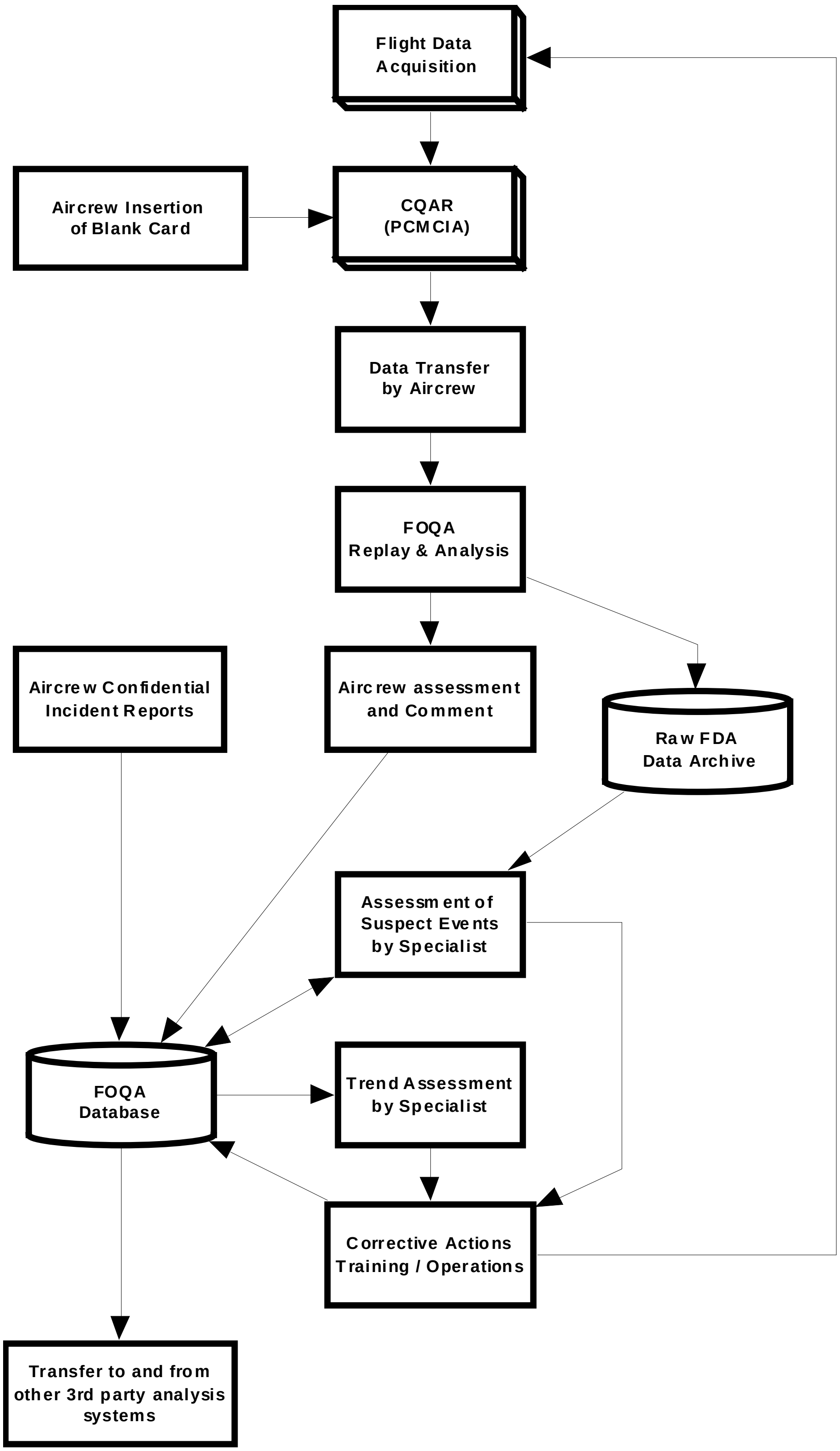

Figure 3 - Post Flight FOQA Analysis

The secure data comprises:

- Captain’s Name
- Co-pilot’s Name
- Take-off Date
- Flight Reference
- Aircraft Registration

The non-secure data comprises:

- Date of Download
- Operator
- Aircraft Type
- Take-off Location
- Take-off Time
- Take-off Date
- Landing Location
- Landing Time
- Landing Date
- Aircraft Weight

Take-off and landing locations are derived from the latitude and longitude information contained within the downloaded flight data. If the system calculates that the aircraft is greater than a certain distance from a known location, it prompts the user to enter the location name and airfield identifier.

The flight data is then examined for flight conditions and flight regimes. A flight condition is identified when the flight data parameter associated with the condition is within the maximum and minimum defined limits. A flight regime is a Boolean combination of flight conditions.

For example, a simple High Speed Approach Flight regime may be defined by two flight conditions Altitude < 800ft AND speed > 170 kts.

Significant flight regimes are then mapped onto FOQA Events. The flight data within a flight regime is examined and the key flight parameters are extracted and stored with FOQA Event details.

For each FOQA Event detected the system displays a summary, which includes

- The name of the FOQA Event
- A textural over view of the event
- The number of occurrences on that flight
- The highest value of the event parameter
- The total time of the event

The system allows the user to request the display of FDA data associated with each FOQA Event if he wishes to examine this further. An example of FDA data presentation showing a regime is given at Figure 4 - Presentation of FDA Data.

The downloaded data can be displayed in both tabular and graphical formats. Tabular data is displayed in engineering units or raw hexadecimal as selected by the user. Graphical data is displayed in engineering units plotted on a time axis. FOQA Events are shown across all graphed parameters, irrespective of whether the displayed parameter formed part of the regime definition or not. Visualisation software may be used to animate the relevant section of the flight.

The user is required to identify each FOQA Event as either a “Confirmed” Event, or a “Suspect” Event. For both Confirmed and Suspect Events, the user is must enter a narrative description regarding the Event. This narrative is a free text comment which the crew member performing the download can use to describe the factors leading up to and surrounding the Event, and in the case of Suspect Events, describe why they feel it is suspect. This description may include factors such as ATC instructions, poor radio communications, human factors within the cockpit, etc., which are not available from the recorded FDA data alone.

“Confirmed” events may also be sub-marked as “Maintenance” Events. These may relate to instrumentation failures observed by the crew, or some other incident that the crew feels needs immediate attention by maintenance. Maintenance events will be automatically made available to the Line Engineer for assessment and action.

Additionally, the aircrew can elect to enter stand-alone reports onto the system. These reports may be new maintenance reports or confidential incident reports. This method of reporting may be used to record non-FDA data related incidents about which the aircrew was concerned. Again, the maintenance reports will be available to the Line Engineers.

Confirmed FOQA Events, Maintenance Reports and Confidential Incident Reports are automatically written to the FOQA database, while suspect events are saved for further analysis by the programme’s FOQA analyst.

Notice that FOQA events cannot be deleted by the aircrew; their options are Confirm (written to database) or Suspect (for specialist review).

Once every Event detected on that flight has been reviewed, the initial post-flight FOQA analysis is complete. The CQAR data card is erased in readiness for the next flight. The aircrew’s involvement ceases at this point, unless the FOQA analysts decide that further information is required. This is better than current practice, because aircrew input is elicited during the download phase.

Figure 4 - Presentation of FDA Data

3.5. Away From Base Downloads

A common problem, which may be perceived with this approach, is the need to cater for away from base downloads, i.e. operation of an aircraft away from its normal operating base either as a result of route planning, diversions or aircraft snags. A number of solutions are possible.

The simplest requires the use of a FOQA data card with sufficient capacity to cater for the duration of the away from base operation. As has been shown, card capacity is not an issue, since commercially available cards are capable of recording in excess of 1800 hours worth of data.

The second approach involves the installation of a networked remote GSS, the data from which can be routinely transferred to the operators central database once the initial post flight process has been completed. With the low cost and ready availability of Internet services, this now provides a suitable means of secure network communications.

3.6. Further FOQA Analysis

On completion of the initial analysis, the FOQA data is available for expert analysis.

This expert analysis may be performed by the individual operator. However in the case of small operators, it will be most likely performed by a third party such as the aircraft manufacturer, a local authority, an equipment manufacturer or a larger partner airline. The use of a third party for expert analysis has numerous benefits. Firstly, it draws on a pool of specialised skills, which a small operator is unlikely to have. Secondly, it ensures that database analysis is performed on a statistically significant amount of data, which allows meaning results to be obtained. Thirdly, consistent recommendations are made to all operators using the system, allowing operators to share their experiences and learn from the experiences of others and finally, spreading the cost over a wider group of operators significantly reduces the cost of FOQA analysis.

In order to utilise an independent analysis house, the operator must transfer their FOQA data. Technology allows this to be easily achieved in a number of ways. The data can be transferred via a dedicated modem link, or a secure Internet link could be established. The data is presented to the analyst in the same format as it was initially presented to the aircrew during the download process. The analyst has full access to all associated flight data for review purposes and may again use flight data visualisation techniques to see an animated view of the events surrounding the FOQA incident.

However, in addition to the FOQA Event summary data and associated FDA, the analyst has the aircrew's description of events surrounding the incident. In most cases, this will negate the need for further follow up contact with the crew.

Combined with the quantitative analysis of the FOQA Event provided by the flight data, this narrative provides the analyst with an insight into the other contributory factors surrounding the incident, there by giving an enhanced understanding.

4. WILL THIS APPROACH BE ACCEPTABLE?

Initial inspection of this proposed method of involving the aircrew in the direct implementation of a FOQA program may suggest that the increased workload and time taken to perform the download process would be unacceptable. However, BAE SYSTEMS experience with the helicopter Integrated Health and Usage Monitoring System (IHUMS) shows this not to be the case. The IHUMS operates in a virtually identical way to the proposed FOQA programme. Aircrews perform the additional pre and post flight activities on a routine basis. Increased aircrew workload is minimal, and the benefits of their involvement far outweigh any additional time requirements.

Experience has shown that aircrews will accept such a responsibility. For example, aircrews like to know that any aircraft problems (heavy landing, excessive use of brakes) from the preceding flight were trapped before they take the aircraft. In the case of IHUMS, the aircraft is also given an in-depth health check between each flight, thereby improving flight safety. IHUMS has utilised this approach for the last 10 years and has been used for over 600, 000 flying hours worldwide.

5. BENEFITS FOR SMALLER OPERATORS

Although FOQA type programmes have been shown to provide improvements in safety, are recommended by many regulatory authorities and in some circumstances have been demonstrated to provide improvements in an operator's efficiency, they are not currently mandatory and can require a significant investment to implement.

This investment takes the form of initial cost of installing the system, and continued costs involved in running the system.

5.1. System Installation Costs

The cost of equipment installation is minimised by utilising the BAE SYSTEMS CQAR which, is a fraction of the price of conventional QARs. It is ARINC 573 / 717 compatible, so will function with any ARING 573 / 717 DFDAU. The data frame size and rate is selected via pins in the aircraft connector, making a single unit suitable for a variety of installations, thus reducing an operators required spares holding. The recording media is COTS PCMCIA PC Memory cards, which are commercially available in a variety of capacities to suit individual operational requirements. The analysis software runs on standard commercially available PCs.

5.2. System Operating Costs

In a typical FOQA programme, the retrieval and replay of the QAR data can require a significant amount of maintenance effort, which smaller operators may feel they cannot support within their current infrastructure.

By utilising the aircrew, who are walking to and from the aircraft anyway, to retrieve and replay the FOQA data at the end of every flight, an additional maintenance task is avoided. The inclusion of an incident reporting system not only ensures that better information is received on FDA events in a timely manner, but also removes the effort required to trace aircrew at a later stage. By expanding this reporting scheme to include maintenance and confidential incident reports, further overheads typically associated with these schemes are reduced.

Small operators cannot support the cost of dedicated staff for FOQA, and must be part of a larger group for database analysis to be meaningful. Therefore analysis by a partner / parent airline or third party specialist is a recommended option.

6. CONCLUSION

For small operators, pilot transfer of data and post-flight assessment combined with third party expert assessment provides a cost-effective FOQA operation that should make FOQA accessible for regional airlines and small operators.

DIAGNOSTICS AS A BASE FOR MAKING PREVENTIVE ACTIVITY MORE EFFECTIVE

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1. Introduction

Effective prevention in aviation needs some methodical base and information. The sources of information are the results of diagnostic investigation. Such the investigations can concern single aircraft or some aeronautical incidents and accidents. Information can be received from the analysis of phenomena worked out from the database of disorders and of flight safety.

In the paper there are presented methods of mathematical analysis of information received from database as well as of working out of presentation activity. As a base for preparation of decision making data the method of Markov-process has been used.

2. The methods of safety rates

With the examination of the rates that describe the level of safety as a tool, it is possible to study the trend of their changes against an independent variable, e.g. time, a number of objects, a performance parameter (the distance covered, the number of passengers or the cargo carried). Periodic changes become known. The study is aimed to either finding the causes of favourable changes, or avoiding unfavourable changes by means of effective prevention [1].

A kind of a function closely related with safety has been defined and labelled ‘the accident function (FW)’. The FW function together with the estimated rates of average losses p enable the safety level (as measured with some selected rate, e.g. the number of aircraft/air crew losses) to be estimated - forecasted. The FW function affords possibilities for following the changes in mean values, the long- and the short-term changes, the susceptibility of a selected parameter to changes within some definite time intervals. Correlation methods, interval functions, and trends of parameters' relative changes in the logarithmic scale have been applied to the above-mentioned analyses.

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The method can be applied, for instance to evaluate preventive actions carried out to improve safety of the C-T-O (Man-Engineering Products -Environment) system [2].

3. The accident function and its use for prediction of losses

The accident function, being a kind analogue of a reliability function, has been defined as follows:

The accident function (**FW**) is a real function **W(t)** which determines, for every real value t, the probability **P** of aeronautical system (defined by the number of aircraft) of some capacity, measurable with the use of selected property (number of airworthy aircraft) at the $\mathbf{t_0}$ time, and which will decrease the capacity after the Δ**t** time proportionally to the change of accident function Δ**W**.

Function **W(t)** depends on the capacity properties of the analysed system characterised by {**k**}, on the condition of capacity and on the requirements (s) to be met in order to consider the system as airworthy. If the incidents and accidents, considered generally, will be determined with the use of accident rate, for instant the so called 'heavy accident' WWc, when the aircraft are lost and when the rate will be multiplied by mean losses coefficient **p** and by expected total time of flights **Np** (as the standard of 10^5 hours of flights) the function **W(t)** can be expressed as follows:

$$\mathbf{W(t) = P\{p, Np, WWc(\tau; k, s); 0 < \tau \le t, WWc(o; k, s)\}} \qquad (1)$$

where: τ is the time range of prediction.

When assuming the τ value equal to one year the expect total time of flight **Np** will be considered also in relation to τ , but **Np** will be the total expected time of flights during the year multiplied by 10^5.

The real calculation can be done for losses:

a) of aircraft in accidents (estimated p_{SP} = 0,8÷1,1)
b) of the aircraft crew (for pilots estimated p_{ZA} = 0,3÷1,5)

So, the losses of system related to the number of aircraft (L_{SP}) in one year (τ0 can be calculated with the use of formula

$$\mathbf{L_{SP}(\tau) = LSP(0) - W_{SP}(\tau)} \qquad (2)$$

where: $$\mathbf{W_{SP}(\tau) = WWc(\tau) \bullet p_{SP} \bullet Np} \qquad (3)$$

And, when considering the losses in relation to the number of losses crews (LPL) in one year this can be calculated with the use of formula:

$$\mathbf{LPL = LPL(0) - W_{ZA}(\tau)} \qquad (4)$$

where: $$\mathbf{W_{ZA}(\tau) = WWc(\tau)_{SP} \bullet p_{ZA} \bullet Np} \qquad (5)$$

For practical purposes there are interesting the losses expressed in numbers of aircraft W_{SP} and crew members W_{ZA} as well. Results of such the calculations can indicate the suitableness of preventive activity.

4. Preventive activity

The preventive activity should be considered as different actions concerned with wide area of problems. Selected actions on prevention problems (for instance for selected aircraft, or its assembly, for out-of-order structural element or mistake of organisation etc) can be inspired by observation and recording of some negative events concerned with flight (accidents, incidents or technical defects) as well as by the trends analysis of disadvantageous or profitable changes.

The area of **prevention** consists of preparation, of introduction to practice and of checking the activity as well. One should realise the much wider area concerned with prevention. This may include the methodology of working out some preventive activity (logistic, social, organisational etc). Up to the time being such the problems are not presented in papers published.

When considering the problem of **prevention** the following areas should be taken into account: working out of preventive documents, methods of implementation of preventive decisions, methods of checking of implementation effectiveness, determination of the last term (in time) for completing the decided preventive actions etc. These can be presented in form of basic algorithm.

Within the aeronautical regulations, in part concerned with air accidents investigation there is in use formula '**air accidents prevention activity'**. This is a part of **prevention** in aeronautics as all.

When working out the documents, this is necessary to define the kind of prevention, its area etc. In many cases the results of such the work can influence on methods, procedures and tools to be successfully used. The kind of prevention can be precisely defined with the use of casual groups of accidents and incidents. This is important because each one algorithm of activity when investigating the air accident or incident will have the final part addressed to **working out the decision on preventive actions** to be undertaken.

These actions may be (or even should be) worked out basing on the results of periodical analysis. Generally, for some technical objects the signal for starting the prevention activity can be the analysis of observed trends, expectations, or results of statistical analysis of some selected defects. This may concern directly the flight safety. In [2] there is considered the statistical analysis of 'heavy accidents' when the crew was consisted of one or of two members on board. The results of work have confirmed the usefulness of such the analysis. There was shown that the ratio of 'heavy' accidents to accidents and incidents is higher when on board is the crew of two.

This is a good example for planning and undertaking the prevention activity concerned with the contents of manual for specific situations in flight.

5. The credibility of prevention

When making decision on prevention implementation three is to expect one of the three following results: successful, partially successful or unsuccessful. This can be presented as follows

$$\mathbf{P\{W(\tau_{n+1}) = d_j, \tau_{n+1} - \tau_n < t, W(\tau_n) = d_i\}, i, j = 1,2,3; i \neq j} \qquad (6)$$

where: $W(\tau)$ – is the semi-Markov process;

n – is the number of intervals;

d_i – is the value presenting success.

For formula (6) there are needed the data for initial states:

$$\mathbf{p_i = P\{W(0) = d_i\}, i = 1,2,3} \qquad (7)$$

and the form of matrix function

$$\mathbf{Q(t) = [Q_{ij}(t)] = P\{W(\tau_{n+1}) = d_j, \tau_{n+1} - \tau_n < t, W(\tau_n) = d_i\} i \neq j} \qquad (8)$$

From the theory of semi-Markov process i results that is the homogenous Markov chain with the matrix of probability transitions:

$$\mathbf{P = [p_{ij}; i, j = 1,2,3]} \qquad (9)$$

The matrix (9) makes possible to determine the credibility and level of successful prevention.

6. Conclusions

- Working out of the basic materials for implementation of prevention with: description of problem, assumption of prevention, methods, procedures and tools, criteria of prevention effectiveness, method of collection of information on the prevention results and the results evaluation.

- Method of implementation of the prevention undertaken with information on: place, time (beginning and finishing), organisation units, responsability.
- Checking system for implementation of prevention undertaken and its evaluation.
- System for having an advantage from the results of prevention undertaken in design, construction, repair of aircraft, maintenance procedures etc; in improvement of manuals and in flight operation, schooling, flight control etc.

Reference

1. Lewitowicz J., Urbański M.: *Determination of flight safety rates and examination of the variation with time in correlation to reliability rates.* Proc. 21 Conf. ICAS, Melbourne 1998, No ICAS- 98-6.7.4, T1, p. 1-8.
2. Lewitowicz J., Urbański M., Wołkowski W.: *Influence of human and technical factors on flying safety with Airplanes flown by a single pilot or crew two.* Proc. Conf. Inter. Aviation Safety, Rotterdam 1997.

ANOMALIES IN INTEGRATED AIRCRAFT SYSTEMS

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ABSTRACT

The deviation in system parameters not generating failures or brakedonwn of systems, but greater then parameter uncertainties normally taking into account in system design, are called as anomalies. The anomalies have considerable influences on the aerodynamics, flight performance stability and controllability. The investigation and monitoring of anomalies as well as the control of system parameter deviation process are actual problem of design and application of the integrated aircraft systems.

The lecture describes the basic background of theory of anomalies. It defines the problems and methods of solving. Especially it determines the new requirements to design and producing the integrated aircraft systems with anomalies. It gives mathematical methods for investigation and monitoring the system anomalies. The applicability of described new approach to system anomalies is demonstrated on the subsystem of hydraulic servomechanism used in aircraft control systems.

INTRODUCTION

The modern aircraft systems are the integrated systems. They are integrated with the avionics, measurement and data processing systems and they are connected with the other systems through the aerodynamics, flight performance, stability and controllability. Nowadays the dynamics and accuracy of systems play the one of the most important role in the synthesis and design of the new systems. However, the non-linearities and scattered parameters of the systems can be described as the deviation in the system parameters. These deviations can be greater then that could be taken into account during synthesis and design of systems. Therefore the design of systems with reduced sensitivity to the system parameter changes is an actual theoretical and practical problem.

1. SYSTEM ANOMALIES

The system parameters i.e. real constructional-technical characteristics of aircraft and its systems are scattered to a great extent in the neighbourhood of the rated values prescribed in technical documentation. During the operation the deviations mentioned above continue to increase stochastically mostly in a cumulative way [1] as a function of

- the physical-technical peculiarities of the structural material applied,
- the peculiarities of their design and manufacture,
- the technical and economical condition of operation, and
- the intensity of operation.

According to our examinations [1, 2, 3] these parameter deviations are achievement the 5 - 10 % sometimes even the 25 - 40 % relatively to the nominal values given in technical documentation.

The space of deviations in system parameters can be divided into three parts (Fig. 1.)

- parameter uncertainties, which can be obtained by some specific methods of control, i.e. robust control,
- the anomalies, which are greater deviations than uncertainties but do not generate the failures or standstill of systems,
- errors, namely deviations in characteristics beyond their prescribed tolerance zones and generate the failures and standstills of systems.

Generally, the errors are investigated by reliability theory and risk analysis very well. The small disturbances in parameters, e.g. parameter uncertainties, are investigated very well too. The robust control gives possibilities to manage with uncertainties. But the influences of system parameter anomalies on the system characteristics have not studied yet on the level needed. The investigation of the aircraft system parameter anomalies on real flight situations, on risk of flight operation and on the control quality is the one of the most actual problem of aeronautical sciences

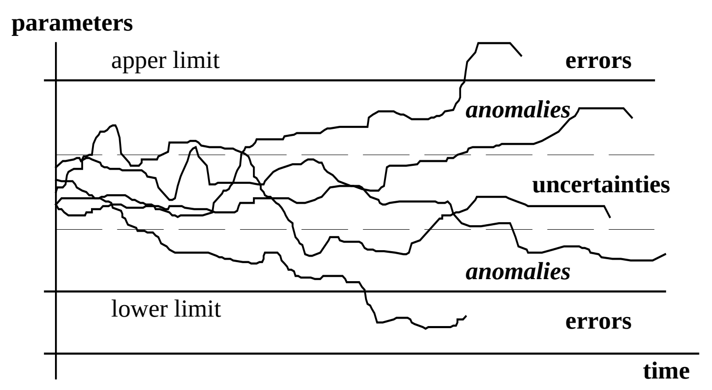

Fig.1. Parameter deviations and their classification

The changes in the system parameters naturally involve also the changes in system characteristics, in our case the deviations in aerodynamic, flight engineering and flight safety characteristics. These anomalies and their effects on the system characteristics should be investigated in details. For this aim we recommend to develop the system anomalies theory.

The theory of anomalies deals with system anomalies and their effects on the system dynamics. The problems of theory of system anomalies can be classified as the mean tasks of given theory:

- initial task - investigation of the structural and operational characteristics, warning the anomalies and the statistical description of the anomalies,
- direct task - study the effects of system parameter anomalies on the aerodynamics, flight mechanics, controllability and stability,
- inverse task (synthesis) - determining of the bounds for the system parameters from the given bounds (admissible field) of operational characteristics e.g. flight safety, quality of manoeuvres, controllability,
- basic task - create the control for the system with anomalies,
- complementary task - determining the basic and additional information for solving the different problems connected with system anomalies, e.g. model-formation, optimal control, identification, etc.

The initial task can be solved by application of theory of measurement, statistics and stochastic approximation. The direct and the inverse tasks are based on the theory of flight and on the using the sensitivity theory.

The basic task is the completely new task. The mean idea is the change of the goal of operation. The new goal is the using the aircraft to maximum time with minimum specific total life cycle cost expenditure under predefined service condition with keeping the system parameters in the prescribed tolerance zones. Therefore we should define the control of anomalies. The technical condition of aircraft can be controlled on two different levels. At first, we can use of the pilot or automatic control of aircraft in case if the motion characteristics are different from given values. In this case the control should be realised on the board of aircraft. Secondary, on the basis of state or parameter estimation results we can use some technical methods of operation (e.g. methods of maintenance or repair) for replacing the nominal system parameters. The connection between these two different levels of control is the fault or error detection.

The complementary task should be solved through the application of the different theories like state and parameter estimation, theory of diagnosis, etc.

2. GENERAL MODEL

When examining the dynamics or technical condition of complicated systems like aircraft, it seems to be describable easily for an engineer if the variation of its state vector **x** chosen appropriately is expressed as follows

$$\dot{\mathbf{x}} = \mathbf{F}(\mathbf{x}, \mathbf{u}, \mathrm{t}), \qquad (1)$$

where **u** is a control vector.

In reality the controlled dynamics of the systems can be investigated on the basis of much more complicated picture. In fact, the variation of state vector **x** is influenced by the variation in the instantaneous values of a number of factors (service conditions, methods of maintenance and repair applied, the realised management, the characteristics of the flight, the atmospheric conditions, etc.). These influences can be given in terms of stochastic processes, random variables or random space (turbulence of atmosphere). Moreover, state vector **x** can not generally be measured directly. Instead, some output signal vector **y** can be measured. Consequently, the controlled motion of the aircraft or their technical conditions, their dynamics can be described only by a much more complicated model than in (1), namely by the following general set of stochastic differential equations [4, 5]:

$$\begin{aligned}
\mathrm{d}\mathbf{x} &= f_{\mathrm{x}}\left[\mathbf{x}(\mathrm{t}), \mathbf{x}(\mathrm{t} - \tau_{\mathrm{x}}), \mathbf{p}(\mathbf{x}, \mathbf{z}, \omega, \mu, \mathrm{t}), \mathbf{z}(\mu, \mathrm{t}), \mathbf{u}(\mathrm{t}), \omega, \mu, \mathrm{t}\right]\mathrm{dt} + \boldsymbol{\sigma}_{\mathrm{x}}(\mathbf{x}, \mathbf{p}, \mathbf{z}, \omega, \mu, \mathrm{t})\,\mathrm{d}\mathbf{W}_{\mathrm{x}}, \\
\mathbf{y}(\mathrm{t}) &= f_{\mathrm{y}}\left[\mathbf{x}(\mathrm{t}), \mathbf{x}(\mathrm{t} - \tau_{\mathrm{y}}), \mathbf{p}(\mathbf{x}, \mathbf{z}, \omega, \mu, \mathrm{t}), \mathbf{z}(\mu, \mathrm{t}), \mathbf{u}(\mathrm{t}), \omega, \mu, \mathrm{t}\right] + \boldsymbol{\sigma}_{\mathrm{y}}(\mathbf{x}, \mathbf{p}, \mathbf{z}, \omega, \mu, \mathrm{t})\,\boldsymbol{\xi}, \\
\mathbf{u}(\mathrm{t}) &= f_{\mathrm{u}}\left[\mathbf{x}(\mathrm{t}), \mathbf{x}(\mathrm{t} - \tau_{\mathrm{u}}), \mathbf{p}(\mathbf{x}, \mathbf{z}, \omega, \mu, \mathrm{t}), \mathbf{z}(\mu, \mathrm{t}), \omega, \mu, \mathrm{t}\right], \qquad (2) \\
\mathbf{x}(\mathrm{t} = \mathrm{t}_0) &= \mathbf{x}_0(\mathrm{t} = \mathrm{t}_0, \omega_0, \mu_0, \mathrm{t}), \\
\mathbf{y}(\mathrm{t} = \mathrm{t}_0) &= \mathbf{y}_0(\mathrm{t} = \mathrm{t}_0, \omega_0, \mu_0, \mathrm{t}),
\end{aligned}$$

where $\mathbf{x} \in \mathrm{R}^{\mathrm{n}}$ is the state vector, $\mathbf{p} \in \mathrm{R}^{\mathrm{k}}$ is the parameter vector characterising the state of the aircraft, $\mathbf{z} \in \mathrm{R}^{\mathrm{l}}$ is the vector of environmental characteristics (vector of service conditions), $\mathbf{u} \in \mathrm{R}^{\mathrm{m}}$ is the input (control) vector, $\mathbf{y} \in \mathrm{R}^{\mathrm{r}}$ is the output (measurable) signal vector $\mathbf{W} \in \mathrm{R}^{\mathrm{s}}$ and $\xi \in \mathrm{R}^{\mathrm{q}}$ are the noise vectors (in simplified case the Wiener and Gaussian noise vectors respectively), σ_x, σ_y are the noise transfer matrices, ω and μ are the random variables assigning the position of vectors **p** and **z** within admissible space Ω_p Ω_z described by density functions $f_p(\bullet)$, $f_z(\bullet)$, t is the time, and τ_x, τ_y, τ_u, are the time-delay vectors.

The general model (2) can be rewritten in following linearised form:

$$\begin{aligned}\dot{\mathbf{x}}(t) &= \mathbf{A}(\omega,t)\mathbf{x}(t)+\mathbf{B}(\omega,t)\mathbf{u}(t)+\mathbf{H}(\mu,t)\mathbf{z}(t)+\mathbf{A}_{\mathrm{nx}}(\mathbf{x},\mathbf{u},\omega,t)+\mathbf{G}_{\mathrm{x}}(\omega,t)\boldsymbol{\eta}(t),\\ \mathbf{y}(t) &= \mathbf{C}(\omega,t)\mathbf{x}(t)+\mathbf{D}(\omega,t)\mathbf{u}(t)+\mathbf{A}_{\mathrm{ny}}(\mathbf{x},\mathbf{u},\omega,t)+\mathbf{G}_{\mathrm{y}}(\omega,t)\boldsymbol{\xi}(t),\end{aligned} \tag{3}$$

where **A, B, H**, **C** and **D** are the state, control, environmental, output and input influence matrices of n x n, n x m, n x l, r x m and r x m dimensions respectively, $\mathbf{G_x}$, $\mathbf{G}_\mathrm{y}$ are the noise transfer matrices, η and ξ are the noise vectors, ω and μ are the random values determine the deviations in the matrix elements. The stochastic time-varying vectors $\mathbf{A}_{\mathrm{nx}}$, $\mathbf{A}_{\mathrm{ny}}$ include the effects of system anomalies depend on the real flight situations initiated by realised control.

In a very simplified case, for the first approximation the linearised model with system anomalies can be given in following form:

$$\begin{aligned}\dot{\mathbf{x}}(t) &= \mathbf{A}(\omega,t)\mathbf{x}(t)+\mathbf{B}(\omega,t)\mathbf{u}(t)+\mathbf{A}_{\mathrm{nx}}(\omega,t),\\ \mathbf{y}(t) &= \mathbf{C}(\omega,t)\mathbf{x}(t).\end{aligned} \tag{4}$$

3. DESCRIPTION OF THE SYSTEM ANOMALIES EFFECTS

The system anomalies effects can be characterised and taken into account by the following ways:

- Including the additive elements in the simplified linearised models results to the form of multivariable perturbed linear system with time-delay and additive system anomalies:

$$\begin{aligned}\dot{\mathbf{x}}(t) &= (\mathbf{A}+\Delta\mathbf{A})\mathbf{x}(t)+(\mathbf{A}_\mathrm{d}+\Delta\mathbf{A}_\mathrm{d})\mathbf{x}(t-\tau_\mathrm{d})+\mathbf{A}_\mathrm{n}(\mathbf{B}+\Delta\mathbf{B})\mathbf{u}(t)\ ,\\ \mathbf{y}(t) &= (\mathbf{C}+\Delta\mathbf{C})\mathbf{x}(t)+(\mathbf{D}+\Delta\mathbf{D})\mathbf{u}(t)+\mathbf{H}_\mathrm{m}\ ,\end{aligned} \tag{5}$$

  where the $\Delta\mathbf{A}$, $\Delta\mathbf{A}_\mathrm{d}$, $\mathbf{A}_\mathrm{n}$, $\Delta\mathbf{B}$, $\Delta\mathbf{C}$, $\Delta\mathbf{D}$ and $\mathbf{H}_\mathrm{m}$ represent the uncertainties and anomalies.

  In set of equations (5) the $\Delta\mathbf{A}$, $\Delta\mathbf{A}_\mathrm{d}$, $\Delta\mathbf{B}$, $\Delta\mathbf{C}$, and $\Delta\mathbf{D}$ contain the changes in the partial derivatives of vector functions f_x and f_y of model (2) respectively to state and control vector elements. So, they take into account the changes in gradient of these functions only. They are the multiplicative anomalies.

  The $\mathbf{A}_\mathrm{n}$ and $\mathbf{H}_\mathrm{m}$ are the additive anomalies in the state and measurement characteristics.

- Characterising the system anomalies effects by probabilities [4] can be given in the following forms:

$$\begin{aligned}&\mathrm{P}_1\{\mathbf{y}(\mathrm{t})\in\Omega_\mathrm{y}\,|\,\mathrm{to}\le \mathrm{t}\le \mathrm{to}+\tau,\mathbf{x}\in\Omega_\mathrm{x},\mathbf{u}\in\Omega_\mathrm{u},\mathbf{z}\in\Omega_\mathrm{z},\mathbf{p}\in\Omega_\mathrm{p}\}\\ &\mathrm{P}_2\{\mathbf{u}(\mathrm{t})\in\Omega_\mathrm{u}\,|\,\mathrm{to}\le \mathrm{t}\le \mathrm{to}+\tau,\mathbf{x}\in\Omega_\mathrm{x},\mathbf{z}\in\Omega_\mathrm{z},\mathbf{p}\in\Omega_\mathrm{p},\mathbf{y}\in\Omega_\mathrm{y}\}\end{aligned} \tag{6}$$

  where the admissible vectorial fields of characteristics are given by Ω.

  If joint density function,

$$f_\Sigma = f[\mathbf{x}(\mathrm{t}),\mathbf{u}(\mathrm{t}),\mathbf{z}(\mathrm{t}),\mathbf{p}(\mathrm{t}),\mathbf{y}(\mathrm{t})] \tag{7}$$

is known, then the recommended characteristics (6) can be calculated as follows:

$$P_1\{\mathbf{y}(t)\in\Omega_y|\ldots\} = \frac{\int_{\Omega_i} f_\Sigma \mathrm{d}\mathbf{x}\mathrm{d}\mathbf{u}\mathrm{d}\mathbf{z}\mathrm{d}\mathbf{p}\mathrm{d}\mathbf{y}}{\int_{-\infty}^{+\infty}\mathrm{d}\mathbf{y}\int_{\Omega_j} f_\Sigma \mathrm{d}\mathbf{x}\mathrm{d}\mathbf{u}\mathrm{d}\mathbf{z}\mathrm{d}\mathbf{p}} \qquad \begin{matrix}(i\in \mathbf{x},\mathbf{u},\mathbf{z},\mathbf{p},\mathbf{y})\\ (j\in \mathbf{x},\mathbf{u},\mathbf{z},\mathbf{p})\end{matrix}$$

$$P_2\{\mathbf{u}(t)\in\Omega_u|\ldots\} = \frac{\int_{\Omega_i} f_\Sigma \mathrm{d}\mathbf{x}\mathrm{d}\mathbf{u}\mathrm{d}\mathbf{z}\mathrm{d}\mathbf{p}\mathrm{d}\mathbf{y}}{\int_{-\infty}^{+\infty}\mathrm{d}\mathbf{u}\int_{\Omega_j} f_\Sigma \mathrm{d}\mathbf{x}\mathrm{d}\mathbf{z}\mathrm{d}\mathbf{p}\mathrm{d}\mathbf{y}} \qquad \begin{matrix}(i\in \mathbf{x},\mathbf{u},\mathbf{z},\mathbf{p},\mathbf{y})\\ (j\in \mathbf{x},\mathbf{z},\mathbf{p},\mathbf{y})\end{matrix} \qquad (8)$$

In practice the application of the described method (Fig.2.) for characterising the system anomalies seems too complicated. Therefore, it may would better to describe the system could taken into account the system parameter anomalies through the internal model.

- Internal model principle [6, 7] using state space approach is may the most important method can be applied for control the system with system parameter anomalies.

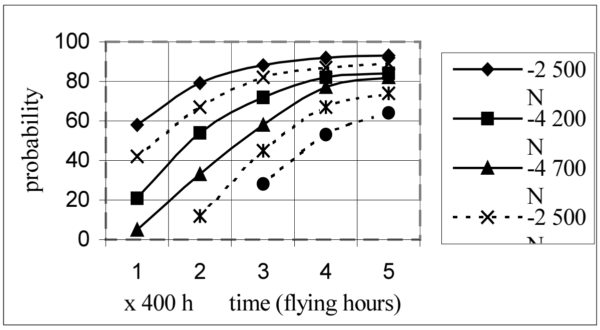

Fig. 2. Probability of lack of lift at fighters Míg-21 due the changes in wing geometry during the operation (line - sigle seat, dotline - double seats aircraft)

The general system, after decomposition and supposing that the matrix elements and environmental characteristics are changing relatively slowly, can be represented by model shown on Figure 3.

reference system: $\dot{\mathbf{x}}_r = \mathbf{A}_r\mathbf{x}_r + \mathbf{v}_r$ and $\mathbf{y}_r = \mathbf{C}_r\mathbf{x}_r$ (9a)

disturbance: $\dot{\mathbf{x}}_w = \mathbf{A}_w\mathbf{x}_w + \mathbf{v}_w$ and $\mathbf{w} = \mathbf{C}_w\mathbf{x}_w$ (9b)

plant (initial system): $\dot{\mathbf{x}}_p = \mathbf{A}_p\mathbf{x}_p + \mathbf{B}_p\mathbf{u}$ and $\mathbf{y}_p = \mathbf{C}_p\mathbf{x}_p + \mathbf{D}_p\mathbf{u}_p + \mathbf{G}_p\mathbf{w}$ (9c)

servo-controller: $\mathbf{u} = \mathbf{u}_c + \mathbf{K}_p\mathbf{x}_p + \mathbf{K}_w\mathbf{x}_w + \mathbf{K}_r\mathbf{x}_r$ (9d)

servo-conpensator: $\dot{\mathbf{x}}_c = \mathbf{A}_c\mathbf{x}_c + \mathbf{B}_c\mathbf{e}$ and $\mathbf{u}_c = \mathbf{F}_c\mathbf{x}_c$ (9e)

error vector: $\mathbf{e} = \mathbf{y}_r - \mathbf{y}$ (9f)

The last method can be useful in application of integrated monitoring and diagnostic subsystem to the system designed. According to our definition the monitoring is the signalisation of the anomalies, e.g. system parameter uncertainties greater then were bounded. The diagnostics estimates the values of anomalies and localises their "places" in the investigated system.

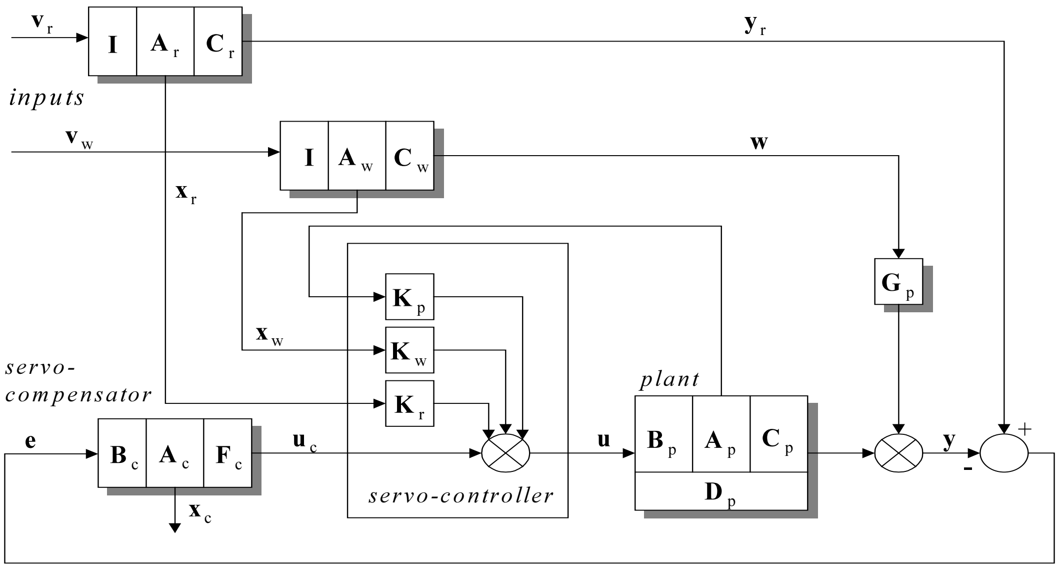

Fig. 3. Application the internal model for control the system with parameter anomalies

4. INVESTIGATION ON HYDRAULIC SERVO ACTUATOR APPLIED TO AIRCRAFT LONGITUDINAL MOTION CONTROL

4.1. Mathematical Model of Hydraulic servo-actuator

The general shame of the investigated hydraulic servo-actuator [8, 9] is shown on the Figure 4. This is the linear symmetric 4-way servo-actuator with mechanical feedback.

The specific form of the equations of the hydraulic servo-actuator must be established, however the basic mathematical model of the system is embodied in the definition of different forces acting on the body. The performance and the dynamic behaviour can be calculated for as wide range of flight conditions as that for which load can be defined. The

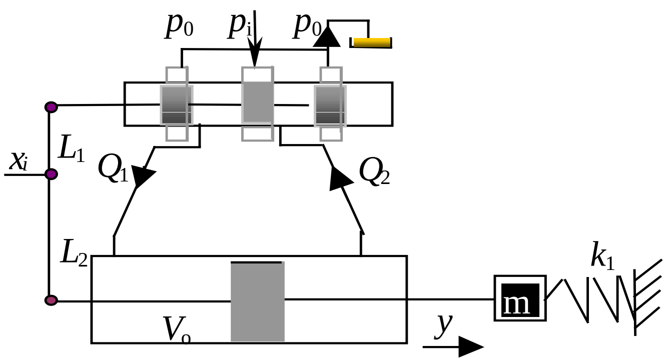

Fig. 4. The investigated model of hydraulic servo-actuator with mechanical feedback (x_i - input, control rod displacement, L_1, L_2 - length of rods, Q_1, Q_2 -input and output flow rate, p_1, p_0 - hydraulic liquid inlet, system pressure, outlet pressure, V_0 - initial volume in cylinder, m - reduced mass of piston k_1 - load coefficient)

different equations of the system can be obtained by the application of the dynamics of hydraulic and mechanical part [9, 10, 11].

The dynamics of mechanical part can be described by using the Newton's law.

$$\sum_i F = m\frac{d^2y}{dt^2} = F_p - F_d - F_f \tag{10}$$

where: - $Fp = A_p\,\Delta p = A_p(p_1\text{-}p_2)$ - is the force causing by the pressure rising in the different sides of the jack piston (in practice $p_1 = p = p_s$, and $p_2 = 0$),

- F_d represent the external loading forces coming from outside of the hydraulic actuator and as usually depends on the displacement of the hydraulic actuator. In our case, it depends on the aerodynamic and flight mechanical characteristics too.
- F_f represent the frictional having two different parts, the coulomb and viscous forces.

In the first approach with using the friction (k_f), proportionality (k_x) and load (k_1) coefficients, the differential equation (10) can be written in the form:

$$m\frac{d^2y}{dt^2} = A_p\,p_d\,(x) - k_f\,\text{sign}(\dot{y}) - k_x(x)\,\dot{y} - k_1 y \tag{11}$$

The second part of the set of differential equation is given by the dynamics of hydraulic:

$$\frac{V_0}{2E}\,\dot{p}_d(x) = -k_p(x)\,p_d(x) - A_p\,\dot{y} - k_x(x)\frac{L_1}{L_2}\,y + k_x(x)\frac{L_1 + L_2}{k_0 L_2} \tag{12}$$

Combining the two parts of set of equations, the following non linear system of equations defined:

$$\begin{cases} \dfrac{V_0}{2E}\,p_d(x) = -k_p(x)p_d(x) - A_p\,\dot{y} - k_x(x)\dfrac{L_1}{L_2}\,y + k_x(x)\dfrac{L_1 + L_2}{k_0 L_2}\,x \\ m\,\ddot{y} = p_d A_p - k_c sign(\dot{y}) - k_x\,\dot{y} - k_1 y \end{cases} \tag{13}$$

4.2. Aircraft longitudinal motion

The equations of longitudinal motion of a symmetric aeroplane are studied very well in the references like [12, 13, 14]. These equations can be rewritten in the following linearised form depending on the different motion and state variables.

$$\begin{aligned} \frac{d\Delta\alpha}{dt} &= \left(\frac{-Z^{\alpha} - M_a g\sin\theta_0}{M_a u_0}\right)\Delta\alpha + \left(\frac{g\sin\theta_0}{u_0}\right)\Delta\upsilon + \frac{d\upsilon}{dt} - \frac{Z^{\delta_e}}{M_a u_0}\Delta\delta_e\ , \\ \frac{d^2\Delta\upsilon}{dt^2} &= \frac{M_{y0}^{\alpha}}{I_y}\Delta\alpha + \frac{M_{y0}^{\dot{\alpha}}}{I_y}\Delta\dot{\alpha} + \frac{M_{y0}^{\dot{\upsilon}}}{I_y}\frac{d\Delta\upsilon}{dt} + \frac{M_{y0}^{\delta_e}}{I_y}\Delta\delta_e\ , \end{aligned} \tag{14}$$

where: α - angle of attack, θ - trajectory angle, Z result force component due to z axis of reference co-ordinate system, M_a - aircraft mass, M_y - result moment around the y axis, I_y - moment of inertia around y axis, index 0 - indicates the initial condition, upper index means the partial derivative with respect to the characteristic given by index itself, $\upsilon = \alpha + \theta$.

4.3. Investigation of the hydraulic servo actuator

To investigate the steady state and the dynamic behaviour of the hydraulic servo-actuator, a number of experiments were performed in the last six years as a series of measurement (pressure, velocity and displacement of the piston). The investigated model was in full-scale of the applied hydraulic system of fighter Mig.21. The measurements were organised at the National Defence University of Szolnok [15, 16].

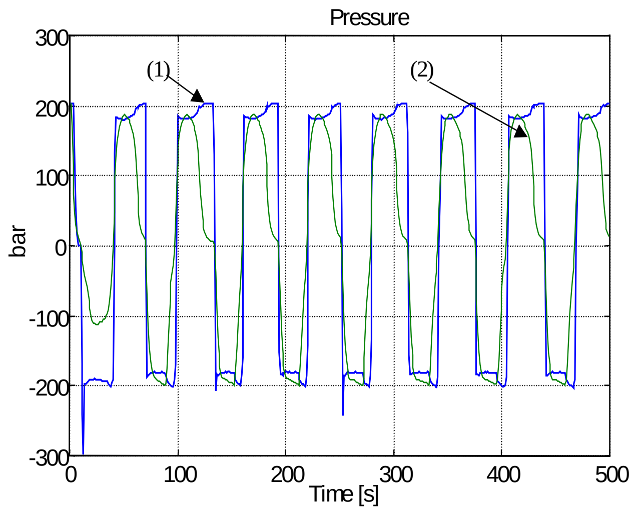

Fig. 5. The different in measured (1) and simulated (2) pressure changes

The measurements were done with different input functions acting on the control rod. The effects of failures were investigated in case when the system elements were changed. The collected data after preliminary data processing were used for identification of the model rewritten in simplified linearised form of state space representation:

$$\dot{\mathbf{x}} = \mathbf{A}\mathbf{x} + \mathbf{B}\mathbf{u}\ , \qquad\qquad (15)$$
$$\mathbf{y} = \mathbf{C}\mathbf{x} + \mathbf{D}\mathbf{u}\ ,$$

where $\mathbf{x} = \begin{bmatrix} p_d = \text{different in inlet and outlet pressure} \\ \dot{y} = \text{piston's jack velocity} \\ y = \text{displacement of the piston's jack} \end{bmatrix}$,

$$\mathbf{u} = \left[x_i = \text{displacement of the control rod}\right] .$$

The Matlab Identification Toolbox had been used for to prescribed model could not give an acceptable results. The method based on the random minimisation of the least square error criteria developed by us [17] resulted to following estimated matrices:

$$\mathbf{A} = \begin{bmatrix} -5.2*10^{-3} & -51.6050 & -949.5652 \\ 4.0956*10^{6} & -3.6521*10^{6} & -1.4057*10^{3} \\ 0 & 1 & 0 \end{bmatrix}, \qquad \mathbf{B} = \begin{bmatrix} 847.40 \\ 0 \\ 0 \end{bmatrix}.$$

The accuracy of the results are demonstrated by the following figures showing the measurement data and results of simulation with estimated model (Fig. 5. - 7.).

4.4. Effects of parameter changes on the aircraft dynamics

The set of equations (14) of the aircraft short period longitudinal disturbed motion can be rewritten in form of state space representation like model (15). The hydraulic servo actuator subsystem and the aircraft longitudinal dynamics are linked in series, that the output of hydraulic servo mechanism - multiplied by ratio coefficient resulting to elevator deflection (δ_e) - is an input for aircraft dynamics and the system outputs two linked subsystems are the flying state variables (α, θ, and θ dot). The diagram in the Figure 8. illustrates the connection between the system one and system two.

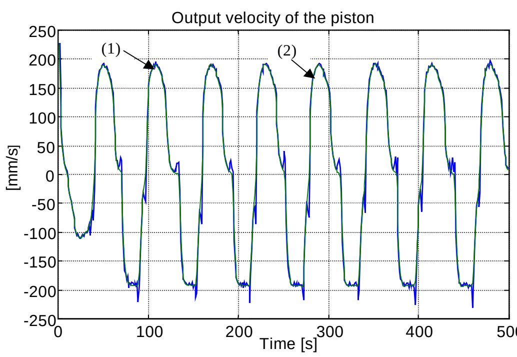

Fig. 6. The measured (1) and simulated (2) velocity of the piston

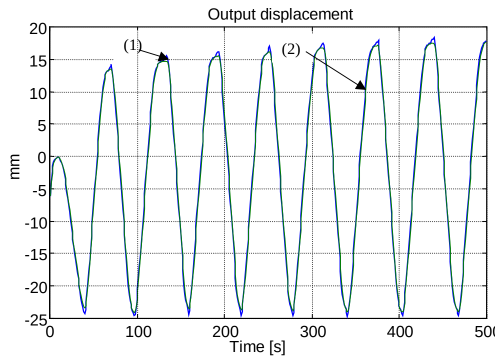

Fig. 7. The measured (1) and simulated (2) displacement of the piston

We have investigated the effects of the parameter changes on the servo-actuator, and aircraft dynamics. The Figures 9. - 11. demonstrate some results of parameter sensitivity and stability investigation [16, 17].

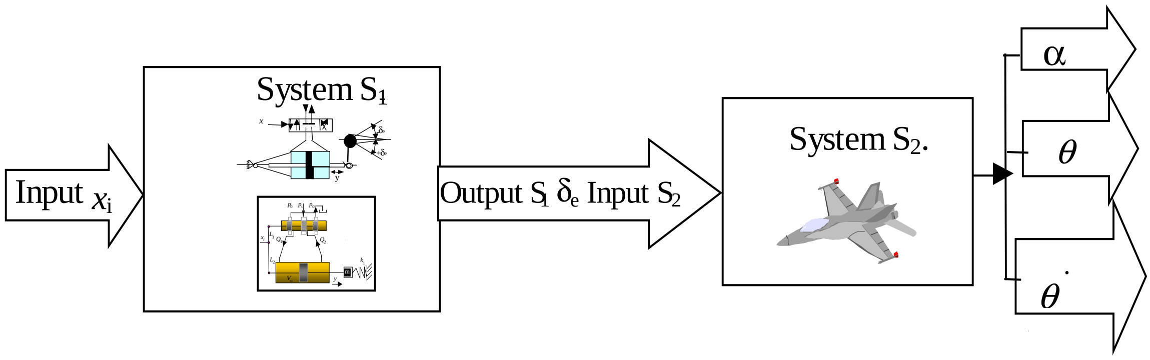

Fig. 8. Connection of hydraulic servo actuator to aircraft dynamics

Effects of Parameter Changes on the Hydraulic Servo-Actuator Dynamics

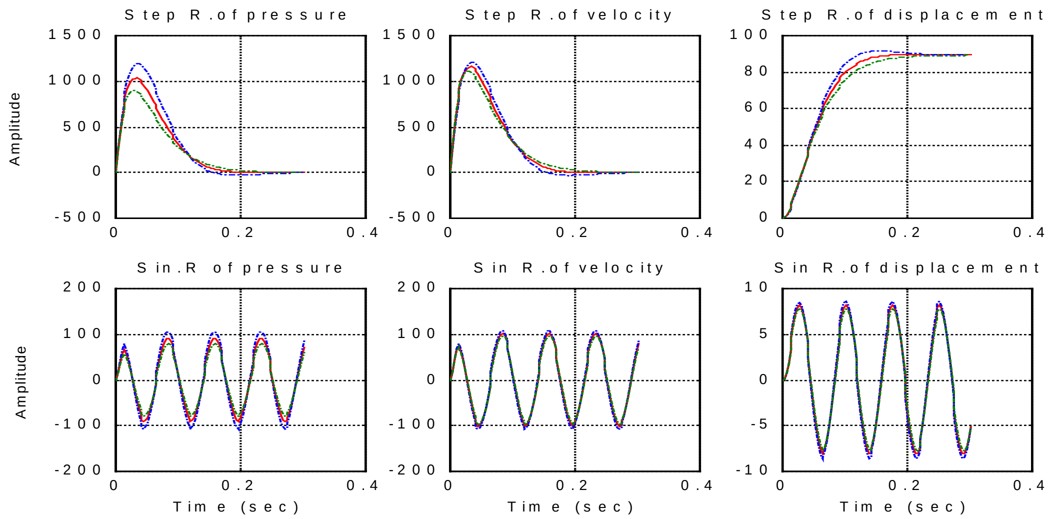

Fig. 9. Effect of changes in area of piston on the hydraulic servo actuator dynamics

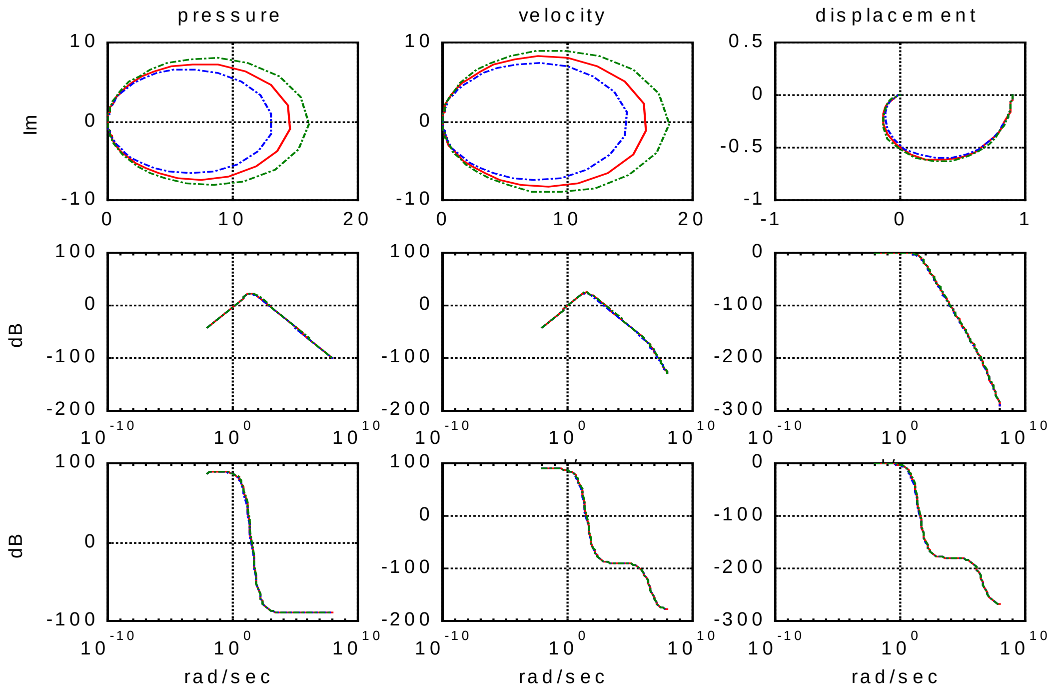

Fig. 10. Effect of changes in flow coefficient on the hydraulic servo actuator stability diagrams

Effects of Parameter Changes on the Aircraft Dynamics

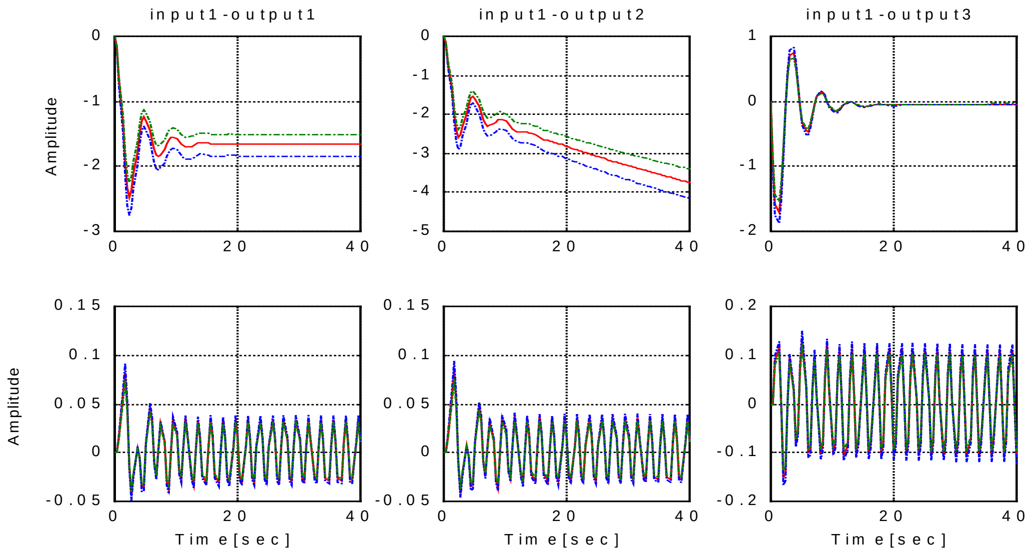

Fig. 11. Effect of coefficient of proportionality parameter and effect of feedback length

The figures show the results of simulation with identified models. The changes in the given parameters mean changes for + 10% and -10% in the chosen characteristics.

The results of sensitivity and stability investigation demonstrate the relatively small effects of anomalies, e.g. parameter deviations in hydraulic servo actuator on the dynamics of the hydraulic servo actuator and aircraft longitudinal motion. However these effects can play important role in case of high accuracy requirements to the actuator dynamics.

5. USING THE INTERNAL MODEL PRINCIPLE

The study of the control system including the hydraulic servo actuator shows that the change of some geometrical characteristics of control systems, like length of rockers or change the ratio L_1 / L_2 of mechanical feedback (see Figure 4.) at the hydraulic servo actuator can be realised relatively easily. Therefore, the use of internal model with one feedback and two feedforwards can be applied to the control systems even in case of system with the mechanical hydraulic servo actuator, too.

The applicability of the internal model approach was studied in the model investigation and simulation. At first we changed several parameters of the hydraulic servo actuator or the aerodynamic model elements applied in the aircraft dynamic model. After that, we tried to compensate the effects of these parameter deviations through the application of the $\mathbf{K}_r$ (see Figure 3.) estimated from the different between the reference and real output characteristics. The results of the model investigation demonstrate the good applicability of the internal model approach (Fig. 12.).

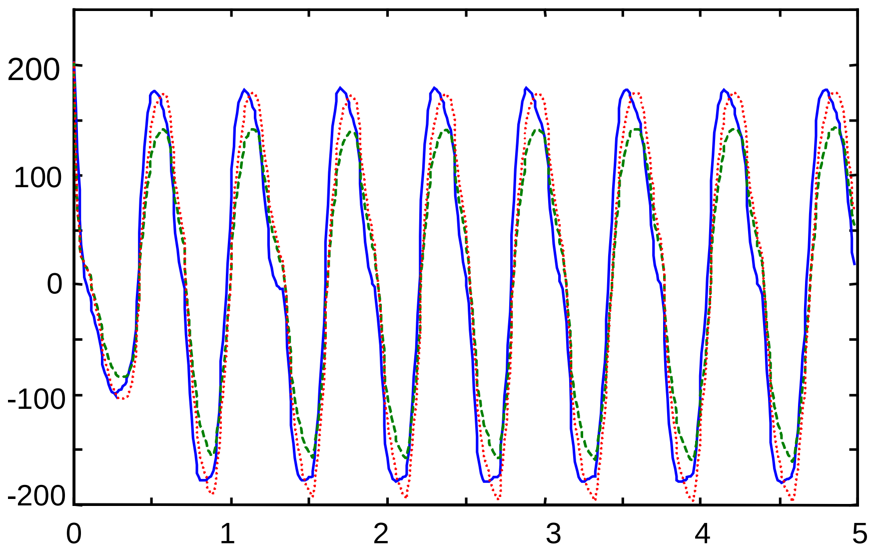

Fig. 12. Use of internal model approach for creating the integrated aircraft control system with hydraulic servo actuator (horizontal axis - time in sec., vertical axis - pressure equals to system pressure minus backflow pressure in bar, line - initial model, broken line - model with 20% increased backflow coefficient of the hydraulic servo actuator model, dotted line - modified model with use of internal model approach)

In any case, the effects of anomalies in hydraulic servo actuator have influence on the system dynamics depending on the shape of input signals. The changes in the load coefficient has not a considerable effect on the system dynamics. In case of change of the internal friction coefficient of hydraulic servo actuator the internal model approach could not give a predicted effect. The 10 - 15 % changes in the aerodynamic coefficients of the aircraft longitudinal motion model initiate the relatively small effects on the aircraft dynamics which can be damped well by use of internal model approach.

CONCLUSIONS

The goal of our long period investigation is the study and development of the theory of anomalies. The deviations in the system parameters having considerable influences on the system characteristics can be called as anomalies.

The effects of anomalies on the system dynamics can be studied and characterised by use of stochastic models.

The effects can be warned and damped with application of a new approach based on the integrated system development. In this case, the measured filtered and estimated characteristics can be compered to the thresholds of the given characteristics. If the estimated characteristics of are greater then their thresholds, the new control approach should switch on. The feedforwards and feedback have to be estimated and applied with the internal reference model.

This approach can be realised easily in case of use of electric servo actuators and servo compensators. However, there is the possible way to use of the feedforward based on the changing the geometrical characteristics of the mechanical elements like rocker length of the hydraulic servo actuator control system.

The practical and theoretical investigation of the system anomalies in control system with hydraulic servo actuator shows that the effects of anomalies on the system dynamics are relatively small. The recommended internal model approach can be useful for damping the system anomalies effects. However this damping effects are depend on the realised input signals and in some cases it can not give the expected results.

REFERENCES

1. Rohács J.: Analysis of methods for modelling real flight situations, 17th Congress of ICAS, Stockholm Sweden, 1990, ICAS Proceedings, 1990, pp. 2046-2054.
2. Rohács J.: Evaluation and Prediction of Permanent deformation in Delta Wings, Periodica Polytechnica Transportation Engineering, BME, 1986, No.1. pp. 17-32.
3. Rohács J.: Detection of technical states with aircraft "proceedings of the 16th Symposium "Aircraft Integrated Monitoring Systems" September 17 - 19, 1991 in München", DLR-Mitt. 92-01pp. 129 - 147.
4. Rohács J.: On Adaptive Control of Technical System, 9th IFAC/IFORS Symposium, Identification and System Parameter Estimation, Budapest, Hungary, 1991, pp. 748-753.

5. Rohács J.: Anomalies in the aircraft control systems "19th Congress of the International Council of Aeronautical Sciences, Anaheim, CA USA, Sept. 18 - 23 1994"ICAS Proceedings, 1994. pp. 1400-1406.
6. Weinmann, A.: Uncertain Models and Robust Control, Springer - Verlag,Wien,New York, 1991.
7. Cheok, K. C., Loh, N. K., Ho, J. B.: Continous-time optimal robust servo-controller design with internal model principle, International Journal of Control, 1988, pp. 1993-2010.
8. Technical Manual on the Aircraft MíG-21., 1982.
9. Garmunin, H. S.: Design the Hydraulic Servo Actuators for Flying Machines (In Russian), Machinostroenie, Moscow, 1981
10. Green, W. L.: Aircraft Hydraulic Systems, an Introduction to the Analysis of Systems and Components, 1990.
11. McLean D.: Automatic Flight Control Systems, Prentice Hall, 1990.
12. Gorbatenko, S. A., Makasov, Ye. M, Poluskin, Yu. F., Seftal, L. V.: Mechanika Polyota (in Russian, Flight Mechanics), Machinostroenie, Moscow, 1969.
13. Lebedjev, A. A., Bobronnikov, V. T., Krasiljsikov, M. N., Maliisev, V. V.: Statistical Dynamics of the Controlled Flight, (in Russian), Machinostroenie, Moscow, 1978.
14. McRuer D., Ashkenas I., Graham D.: Aircraft Dynamics and Automatic Control, Princenton University Press, Princenton, New Jersey, 1973.
15. Balogh, E.: Built of the Measurement System for Investigation of Dynamic Characteristics of the Aircraft System, (in Hungarian), MSc. Thesis, Technical University of Budapest, Budapest, 1990.
16. Abdesselam, A.: Effects of Hydraulic Servo-Actuator Anomalies on Aircraft Dynamics, PhD. Thesis, Technical University of Budapest, Budapest, 1999. Pp. 149. Rohács,J.: HVS ICAS
17. Rohács, J.: Effects of Anomalies in Hydraulic Actuators into Dynamics of Aircraft Control System, Proceedings of 20th Congress of International Council of Aeronautical Sciences, Sorrento, Italy, 8 – 13 September, 1996, AIAA, 1996, pp. 2320-2325.

General Safety & Benefit Perspectives

FDAP - THE PILOTS VIEW POINT

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Introduction

I guess that Paul McCarthy and myself are the "Mutt and Jeff" or "Albert and Costello" show. I will leave you to work out who is who! Paul will explain some of the most important features of these programs, namely security, confidentiality and legality, but I want to expand on the whole theme of how to achieve the stated "aims" (excuse the pun) of a FOQA program by involving the line pilots. The simple message is that without the pilots' confidence and involvement in the program, it will fail to achieve its full potential. In the modern idiom, they must have ownership of the scheme.

In the introduction to my presentation, it was explained that I chair the IFALPA Aircraft Design and Operations (ADO) Committee and that I also perform other technical functions both within the Federation and my own Association. I would first like to briefly explain what the International Federation of Airline Pilots Associations is for those that may not know about it. IFALPA is a global organisation that brings together airline pilots associations under one umbrella. It now consists of over 90 member associations and has a total membership of over 100,000 pilots. It is one of only two organisations, (the other being IATA) that have permanent observer status on the Air Navigation Commission of ICAO. There are 11 standing committees dealing with all aspects of professional and technical matters. I chair the ADO Committee and until recently Paul McCarthy chaired the Accident Analysis Committee, so we have worked together for a number of years.

However, I am principally here today in my capacity as an ordinary line pilot, who is part of a FOQA program or, as we call it, a Flight Data Analysis Program (FDAP). My presentation does not necessarily represent either my airline's or the Federation's view nor is it intended either to praise or criticise the program that I am involved in. It is my ideas, based on my observation of a number of FOQA programs, as to the necessary ingredients required to insure that these programs are successful from a line pilot's viewpoint. I hope to show that this results in the most effective way to use the data to achieve safety and operational improvements. There are basically three elements, which will provide for a successful outcome, which I will describe borrowing the acronym "AIMS".

Agreement with the Aircrew (ALPA)

An agreement with the aircrew (airline pilots association- alpa) is an essential part of the process. Is it the most important? I will answer that by saying it should not be, but, perhaps, at first it has to be. The agreement must state that the program's aim is to improve flight safety and operational efficiency and is not intended in any way to be used as a disciplinary tool. It must ensure the independence of the team and its members. It must protect the individual pilot's rights and ensure the confidentiality of specific data. It should describe the general process by which the data will be analysed and how the results will be used and how they will be published. It must be strictly adhered to and only changed by an agreed system. In the case of individual counselling, it must clearly state the methods and processes to be used and the designated persons to be involved. I do not wish to dwell on these details or the legal aspects, as these will be covered expertly by Paul in his presentation. Agreements should not be viewed as a negative, after all the word itself implies the coming together of parties in a mutual undertaking. It does, however, provide pilots with the necessary comfort blanket to allow their daily operations to be observed, albeit, in this case, by data analysis.

Involvement.

"*The (accomplished) professional pilot brings a professional knowledge base to bear on the problem, irrespective of the changing environment/technology."* (Ces. Crook-Massey University.)

It is imperative that this agreement with the local pilot's association should translate into involvement in the program by pilots and, by pilots, I mean predominantly line pilots.

This is a theme that Paul and I continually emphasise. So what is so special about the line pilot? It is not that other pilots do not have knowledge, skill or expertise, they just have different ones. A management pilot has a company perspective; the regulator has an authority's (legislative) perspective; a test pilot has a purist's perspective; and a manufacturer's pilot has a commercial perspective. The line pilot has a unique perspective, which he or she brings to any problem. It is simply that he or she regularly operates the company's routes day and night and in all kinds of weather. This expertise, coupled with proper training on the analysis program, allows these experienced pilots to look at the data and understand why something is happening not just how. It allows the comments attached to the analysed results to have credence. I will return to this point later with some illustrations.

Another great danger is the 'Holier than Thou" syndrome. 'Look at what this pilot or these pilots are doing." I know that, because I fly the line, I am part of the data and that some of those events are mine. This also helps with the acceptability of the program by other line pilots.

We have not reached a point, in the program that I am involved in, that we can carry out individual counselling, as in Quatas or BA. I will only say that both systems rely on the first contact being made by the respective Association's representative, another pilot, which is why the process is generally well accepted. Furthermore, the process is seen as remedial counselling rather than a disciplinary procedure.

Integrity of the Data.

When I was putting together this presentation, I knew that there would be other presentations on FOQAs, but not their content. It may be, therefore, that what I have to say about the integrity of the data will have already been said. If it has, I can only repeat that it is vitally important that the data been very carefully verified before it is analysed. This does not necessarily have to be a task for the pilots. Computer programs can do some of the filtering, whilst trained individuals can do the fine filtering before it is passed for analysis. I remember Colin Tate, from the Flight Data Company, labouring this point, when he came to brief and help train our team, but it is not only excellent, but also essential advice. If the data is not correct, the conclusions and trends will incorrect and, apart from undermining the program, this may result in wrong or inappropriate amendments to procedures. It is equally important to understand why the data has been rejected and to take firm action to improve the quality. It may also be, that within the rejected data, lies other important information.

It is also very easy to get mislead by rates rather than concentrating on trends. I remember when I was doing my statistics course at university, our lecturer likened statistics to a drunk holding onto a lamppost. "More for support than illumination." My other analogy is somewhat sexist, so I apologise in advance. "Statistics are like a bikini, what they show is interesting, what they hide is vital!" Both these apply to the FDAP programs. The statistics can be skewed very easily by bad or "trashed" data, which can also "hide" good data. It is absolutely vital that we do not misuse the absolute values until the program has reached an acceptable and repeatable level of accuracy.

Improvements in recorders and program.

Let me briefly put on my ADO Chairman's hat. Whilst there have undoubtedly been some significant technological improvements to both recording devices, databuses and to data gathering and analysis programs, there are still glitches in the system. I note that there will be a number of speakers that will address these issues at this Conference, so I do intend to dwell on these technicalities. Instead I wish to consider their impact from the line pilot's perspective. It is very easy for people to be enamoured by technology and lose sight of the real problem. Some new programs allow the data to be run on computers by generating images of aircraft and engine instruments. Whilst these are a useful tool, the old adage "Garbage in-garbage out" applies. It is also essential to understand the assumptions made in mathematical modelling, which especially in the case of terrain may be misleading. I remember seeing a replay of an approach flown into the old Hong Kong airport by one of local airlines' aircraft that did not take account of the side step made by most local pilots, in good weather, to get a longer line up. Now that Kai Tak is closed, I can reveal the secrets of the "barn door" approach.

What are needed are improvements in recorders and programs, a clear understanding of the program's limitations and for the review committee, an education of the program's capabilities, warts and all. This stops inappropriate notices to crew, which undermine the credibility of the whole program. In order to prevent this the FDAP team members need to understand these issues by having a close liaison with the program vendor and the engineering team responsible for recorders and data analysis

Meaningful results.

The analysis should then result in meaningful results. So far that means that the analysis of the data is sound and that any recommendations or conclusions to the LOMS (review) Committee are supportable. Now Paul McCarthy and my fellow members in ISASI, the International Society of Air Safety Investigators will tell you that one of the "Golden Rules" of accident investigation is "Deal only in facts". Now what I am about to say may seem to contravene this rule, but I do not think so. I believe it is a sensible application. Let me use a hypothetical example. Imagine a busy airport with closely spaced parallel runways that has an approach that keeps the aircraft high and then requires the aircraft to lose height rapidly in order to intercept the glideslope. Individual crews may experience exciting approaches, some resulting in overshoots. A few such approaches may generate an air safety report, but not enough to show a trend or to trigger a response from management. Now remember that the picture of what might have occurred is built from lines of data. The trend shows that a significant number of aircraft have high rates of descent at different stages of the approach, with late stabilisation, along with a few approaches that result in "go- arounds". The pilots that have also flown those approaches have a pretty good idea for the reason why. The data now supports that individual experience, the experience gained by regularly flying the line. The report can go to the monitoring committee, with an interpretation of the trend analysis, supported by the data and with a recommendation to review this approach rather than a set of figures merely showing a series of unrelated events. What are the benefits? Hopefully better prepared crews, safer approaches through revised procedures and a reduction or elimination of missed approaches. The latter is a clear gain to both the airline and the airport in question. Could this be done without the line pilot's involvement? - possibly, but more importantly, probably not.

Let me now give a specific example- the performance procedures were changed for our Airbus fleet in order to reduce the V speeds. The FDAP program parameters, however, were still based on the original performance figures, which in turn came from the Aircraft Operational Manual. (Improvements have now been made so that the information comes from the Flight Management

Computer.) This resulted in most flights showing high speeds after take-off. It was apparent that either every pilot was mishandling the aircraft or there was an error. Notice the error was not in the recorded data, it was correct, but the parameters being used were incorrect. You may think this was obvious and, perhaps it was, but I know of one airline, which does not use pilots for its analysis, that published an article to its pilots warning them that most of them were using excessive climb speeds, as a result of similar data. This sort of incorrect conclusion can undermine the credibility of these programs and negate the correct conclusions.

Having provided the best analysis possible, it is now up to the review committee to make recommendations on procedures and training. This committee consists of management and training pilots and, of course, members of the safety pilot. Once again the inclusion of a pilot representative on this committee will help with a positive outcome and an acceptance by the aircrew body.

Finally, let me briefly say something about the individual counselling programs. It is not always easy to take criticism, particularly if it is related to piloting skills. If, however, an individual is identified as regularly carrying out a bad practice, such as over-rotating an aircraft at take off, a call from a fellow pilot is much easier to accept than being called to the manager's office. The crucial part of selling this message, at this stage, is that the company is unaware of the individual's identity. If the trait is corrected, it never will, if not, a properly agreed remedial program will swing into place. Line pilots have full ownership and it therefore works.

Success
Sell the idea.

The crucial message is that we do not leave a "successful outcome" to chance. The first action by the team is to sell the idea by explaining what the program is and by alleviating any fear or suspicion that it is another "Spy in the Sky". Pilots have to be confident that the data will be treated confidentially and securely and it will not be used against them. The fact that their peers are involved with the consent and agreement of their association helps to allay those concerns. One of my roles is to act as an honest broker and to step in, if I think the agreement is in danger of being breached. In actuality, a mutual trust has developed in which everyone has become sensitive to the issues. Any potential abuses are almost always the result of an over enthusiasm, rather than any Machivellian plot to breach the agreement. In our program, one of the team is a full association board member and two of us are on our technical and safety committee. This starts with the agreement and with ALPA involvement, but in the long term it will only be successful, if the program delivers meaningful and timely results. Then and only then will suspicion be replaced by support.

Safety culture and Systemic changes.

It is a "given" that a safety program can only work if the management buy into by acting in a positive way i.e. by adopting, adapting or improving the safety culture and, where necessary, making systemic changes. It needs to be supported from a high, the higher the better. It is often necessary to explain the cost benefits, though most CEOs should understand the need. I like Dr Rob Lee's quote (Australian Transport Safety Bureau)-"Safety is a profit centre, not a cost". If it is done right, this is true. There is now, however, another incentive, namely, the new legislation on corporate manslaughter, which should concentrate every executive's mind! The main message is that if a systemic problem is identified, there must be the support to implement the necessary changes.

SOPS

How does this translate for the pilot? By improving Standard Operating Procedures, changing crew operational training manuals and by input into simulator and other training. There is no point telling pilots not to use high rates of descents, when the procedure for a particular airport is the basic cause of it. A revised Company procedure, a better briefing, simulator training and/or an approach to ATC to try to alleviate the underlying problem are all means of trying to solve the real problem, not the apparent one. In other words, it has to be seen by the line pilot that the flight operations management have understood the problem and provided a practical, workable solution Perception, as well as reality, is vitally important.

Let me give you some specific examples from our program:

a. A maximum speed has now been imposed below 2500 feet to reduce the incidents of high rates of descent near to the ground.
b. Aircraft must now been fully configured by 1500 feet (except for certain visual/circling approaches) to ensure stabalised approaches.
c. Procedures and briefings have been changed for certain airports.

The data from the FDAP indicates that these revised procedures are reducing the incidents that were previously highlighted.

Ideally, the FDAP (FOQA) should be part of an integrated safety system, not a "stand alone program". The data should be viewed along with other reports and the safety culture should encourage pilots to report any occurrence in order to identify potential causes. Initiatives, such as LOSA, the BASI "Indicate" program and confidential human factor reporting systems, should be used to highlight errors in normal operations and to identify and assess the existing defences. These inputs can then be collated and analysed to produce better overall system improvements to protect against latent and active failures. FDAP programs can then be used, in part, to measure if and how successful these changes have been.

Safety and Operational Improvements

How do we measure whether it has been successful from line pilot's point of view? The obvious answer is that we get a safer and more efficient operation. It is possible to measure that, to some extent, by seeing if there is a reduction of recorded/highlighted events. This may be indicative that the problem has been fixed or it may be that pilots are reacting to the "flavour of the month". It, therefore, needs to be monitored over a long period and verified by other means, such as positive feedback from the pilots, which will probably mean more reports being submitted rather than less. The greatest measure of success is an acceptance of these initiatives by the pilots and an active support of the programs.

Summary –The Bottom Line

In summary, FDAP (FOQA) programs must have the following ingredients:

Agreement with the Aircrew (local ALPA)
Involvement by pilots and **I**ntegrity of Data.
Meaningful and timely results
In order to achieve a
Successful Outcome

So that is my message. The "Bottom Line" is very simple and it applies not only to FOQA programs, but to most programs and safety initiatives within the airline industry:

Aircrew (ALPA)
Involvement
=
More
Safety and Operational Efficiency.

The Global Aviation Information Network (GAIN)

Using Information Proactively to Improve Aviation Safety

U. S. Federal Aviation Administration
Office of System Safety
May 2000

Abstract

The United States Federal Aviation Administration (FAA) proposed the Global Aviation Information Network (GAIN) as a voluntary, privately owned and operated network of systems that collect and use aviation safety information about flight operations, air traffic control operations and maintenance to improve international aviation safety. Improved technology has enhanced the aviation community's capability to obtain information about adverse trends, and experience has demonstrated that the systematic collection and use of this information can facilitate the correction of those trends *before* they cause accidents or incidents.

The concept of using information proactively to improve aviation safety recently received support from in the U.S. from the highest levels – the President and Congress. The concept is also being implemented in other transportation modes in the U.S. – maritime, rail, highway, and pipelines – where programs are in various states of development. Outside of transportation, the Critical Infrastructure Assurance Office (CIAO) was recently created in response to the President's concern about the vulnerability of information infrastructures in the U.S., and CIAO is now developing ways to collect information about near breaches of infrastructure security in an effort to prevent actual breaches. Also, the Institute of Medicine recently issued a reporting noting that nearly 100,000 people die each year from medical errors, and it proposed the establishment of processes to collect and use information proactively help prevent such errors.

Much more extensive information about GAIN can be found on the Internet at http://www.gainweb.org.

A. The Accident Rate Plateau

After declining significantly for about 30 years to a commendably low rate, the worldwide commercial aviation fatality accident rate has been stubbornly constant since about 1985. Given the projected increase in volume in international aviation traffic, studies by Boeing forecast that unless the aviation community gets *off* the accident rate "plateau," there will be *a major hull loss every week or ten days, somewhere in the world, by the year 2015*. The FAA proposed GAIN because that is an unacceptable result.

The question is how to get *off* the accident rate plateau. Ongoing activities around the world are directed at improving aviation safety, including the introduction of new technologies, such as enhanced ground proximity warning systems, more sophisticated collision avoidance systems, satellite navigation to allow more precise navigation anywhere in the world and eliminate non-precision approaches, and many more. Aviation regulatory agencies have all played a major role in their regulatory, inspection, and enforcement responsibilities. Significantly improved training for pilots, mechanics, flight attendants, air traffic controllers, dispatchers, manufacturer personnel, and other aviation professionals has also contributed to improved safety.

All of these activities, and more, are crucial, and they have all contributed to the dramatic reduction of the worldwide aviation accident rate since 1950. Moreover, increasing international collaboration will help to insure that these activities will continue and expand, as they must. The leveling of the accident rate curve, however, suggests that the marginal safety returns from these previous ways of improving safety are diminishing, and that it is necessary to find *new* ways of preventing accidents and incidents.

One of the "new" ways that many in the world aviation community are now exploring is the *voluntary international collection and sharing of information about aviation safety problems before those problems result in accidents or incidents*. All too often, the testimony at accident hearings from the "hands-on" people on the "front lines" is that, "We all knew about that problem." The challenge is to get the information that "we all knew about" – not only from pilots, but also from flight attendants, air traffic controllers, mechanics, dispatchers, manufacturers, designers, airport operators, the people on the ramp who close the cargo door, and others – and *do* something about it *before* it hurts someone.

B. <u>The Heinrich Pyramid</u>

Not hearing about problems that "We all knew about" is a common characteristic of potentially hazardous endeavors of all kinds, as depicted by the Heinrich Pyramid (Figure 1). The Heinrich Pyramid shows that for every major accident in a given endeavor, there will be 3-5 less significant accidents, and 7-10 incidents, but there will be at least *several hundred unreported occurrences* (the exact numbers obviously vary with the nature of the endeavor).

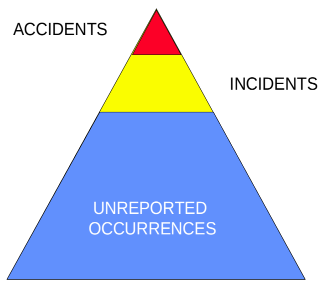

Figure 1-- Heinrich Pyramid

Usually these occurrences are not reported because, by themselves, they are innocuous, <u>i.e</u>., nobody was hurt and nothing was damaged. This is the case in the vast majority of commercial aviation operations – no harm is done because systems are generally so well designed and robust with backups, and backups upon backups, that things rarely go wrong, and when they do, harm or damage is even more rare. Today's unreported occurrences, however, are the "building blocks" of tomorrow's accidents and incidents; and when they happen in conjunction with other building blocks from the "unreported occurrences" part of the pyramid, they may someday become an accident or incident.

There are many aviation examples of the Heinrich Pyramid concept, but two accidents, one relatively old and the other more recent, vividly make the point and show that the problem is just as real today as it ever was. The first accident occurred in 1974, west of Dulles International Airport, near Washington, D.C. The pilots were apparently confused by the approach chart for a non-precision approach, in conjunction with what the air traffic controllers said, about how soon they could descend. They descended too soon and hit a ridge.

At the accident hearing it was revealed that other pilots had previously experienced the same confusion on that approach into Dulles, but the ridge was not in the clouds during those previous approaches. When the accident occurred, on the other hand, the ridge was obscured by the clouds. One of the most tragic aspects of this accident is that pilots from one airline had apparently reported the problem to their management – which in itself was unusual in those days – but the crash involved a different airline.

The other accident occurred in Strasbourg, France, in 1992. One possible cause of the accident was that the pilots thought they had directed their autopilot to make a 3.2 degree descent in a non-precision night approach, but they erroneously directed it to make a descent of 3200 feet per minute. Although the mode distinction between *angle* of descent and *rate* of descent was apparent elsewhere in the cockpit, the window into which they dialed the number would have said "3.2" for a 3.2 degree angle of descent, but instead it said "32" – without the period – which meant a 3200 fpm descent. Once again, the pilots “all knew about” the potential problem.

Several significant lessons are apparent from these two accidents. First, they are textbook examples of problems that the hands-on personnel knew about, but that nonetheless resulted in an accident before the problems were corrected.

Second, each of the links in the chains that led to both accidents were in the "unreported occurrences" part of the pyramid, i.e., the approach chart confusion and the autopilot mode error, by themselves, were innocuous, and thus unlikely to be reported in most reporting systems in place today, whether voluntary or mandatory – until they resulted in an accident or incident.

Third, the approach chart confusion and the mode error, by themselves, are not only not accidents or incidents, they are normally not even potential regulatory violations. Although most commercial aviation systems usually learn about nearly all accidents and most incidents, and some reporting programs have also attempted to mandate the reporting of potential regulatory violations (and some programs, such as ASRS, provide incentive for the *voluntary* reporting of potential regulatory violations), there is no conceivable reasonable way to *mandate* the reporting of occurrences that do not rise to the level of accidents, incidents, or potential regulatory violations. Instead, short of an accident or incident, *the system will generally have to rely upon voluntary reporting to learn about these types of problems*.

Last, but not least, these accidents illustrate the importance of *international* information sharing, because (a) operators from all over the world fly into Dulles; and (b) the autopilot in the Strasbourg airplane was in airliners all over the world, and the accident could have occurred anywhere in the world.

Why do aviation professionals who are highly trained, very competent, and proud of doing what they do well, *still* make inadvertent and potentially life-threatening mistakes? Blaming the problems on "human error" is accurate, but does nothing to prevent recurrences of the problem. If people are tripping over a step "x" times out of a thousand, how big must "x" be before we stop blaming the *person* for tripping, and start focusing more attention on the *step*, e.g., should it be painted, or lit, or ramped? The flattening of the accident rate curve tells us that focusing on the individual, while important, is no longer sufficient. Instead of simply increasing regulation, punishment, or training, which focus primarily upon the *operator*, it is time to start sharing information that can help *improve the system* in which the operator is operating.

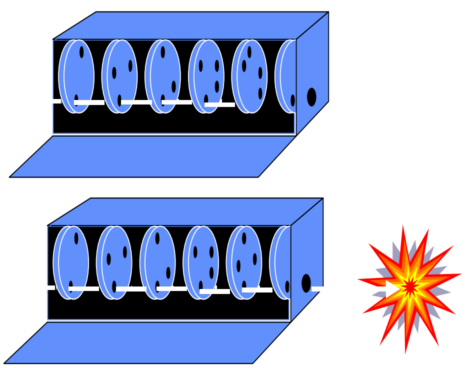

Figure 2

The next question, then, is *how* to improve the aviation system. Commercial aviation accidents are such rare and random events that they are analogous to light coming out of a box without any discernible pattern – causing us to wonder why the light emerges so randomly. Upon opening the box, we discover that it contains a series of spinning disks with holes, lined up along a common axis. The light emerges from the box – an accident occurs – if and only if the holes in the disks happen to line up in front of the light (Figure 2). This analogy is borrowed from the Swiss cheese analogy used by Prof. James Reason from Manchester University in England – when the holes in a stack of cheese slices line up, that represents an accident.

Each spinning disk (or slice of cheese) could be compared to a link in the chain of events that led to an accident. One disk might be the confusing approach chart, another might be the autopilot mode confusion, another might be a confusing page in a maintenance manual, and still another might represent management's attitude toward safety. A study by Boeing reveals accident chains with as many as *twenty* links, each one of which represents an event that, with a different outcome, would have broken the chain and avoided an accident. Every one of these links, separately, is usually innocuous and in the "unreported occurrences" part of the pyramid, but when they happen to combine in just the wrong way – when the holes in the spinning wheels happen to line up – that is an accident.

Viewed in that manner, the challenge in collecting and sharing information to prevent accidents is to obtain information about each spinning wheel, each link in the chain, separately, to try to determine how to reduce the number of holes in each wheel. This effectively dissects a potential accident or incident into its component parts in order to facilitate a separate remedy for each component part of the problem.

C. **<u>The GAIN Concept</u>**

In order to accomplish this information sharing to learn about the potential individual links in an accident chain, the FAA proposed the Global Aviation Information Network, or GAIN. With a privately owned and operated and voluntary global network of data exchanges – thus the name Global Aviation Information *Network* (Figure 3) – government, industry, and labor can cooperate with each other, to their mutual benefit, to make the system safer.

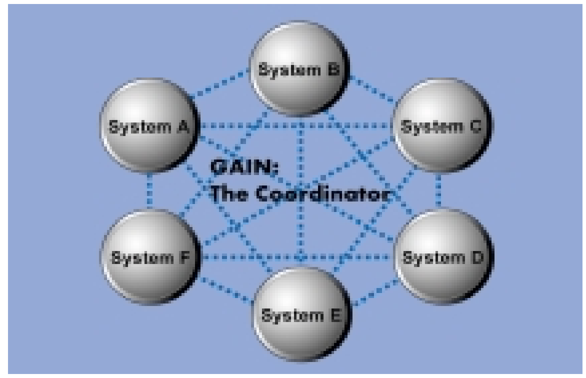

Figure 3 – The GAIN Concept

1. <u>The Importance Of Private Ownership</u>. Experience has shown that proactive use of information not only has the potential to improve safety, but can also result in significant savings in operations and maintenance. If this proactive information concept generates savings for the aviation community and helps to prevent accidents, then the aviation community will *want* to own it, and the *savings will create a strong incentive to operate more safely*. Thus, private ownership would make GAIN work far more efficiently and effectively than government agency operation because – without criticizing any government agency – it is a basic fact of human nature that we run faster when there is another runner to create an incentive to win. Even with the best of government regulator intentions, good intentions will rarely be as powerful an incentive for regulators as the savings can be for the aviation community.

On the other hand, if the concept does *not* save money and prevent accidents, then it is not worth doing.

2. <u>Potential Information Sources</u>. Information would be collected from a variety of aviation community professionals – pilots, mechanics, flight attendants, air traffic controllers, ramp personnel, dispatchers, airport operators, the military, manufacturers, government regulators, and others. Then, instead of throwing away the vast majority of the information, as we do today, we would analyze it, determine norms, determine problems, and otherwise "mine" the information for all of the valuable gold it contains.

Many of these potential information sources are obvious, but others are not. The business aviation community, for example, could be a valuable source of information. Before an airline buys several hundred of a new high technology autopilot, for example, several Fortune 500 companies were probably already using it in their fleets. Thus, the Fortune 500 fleet use provides an actual-use experience base for many advanced technologies that is otherwise unobtainable any other way; and this provides valuable information about many operational human factors problems that are discernible only by actual use.

3. <u>The Need For More Powerful Analytical Tools</u>. The technologies are improving every day to facilitate the collection of the data and the dissemination of the valuable information it contains, but what is needed most is powerful new analytical tools to take the enormous quantity of input

and convert it to value-added output that will help the aviation community become safer. These tools must be able to *transform information into life-saving knowledge*.

The analytical step is by far the most critical step because GAIN will be *completely voluntary*. Consequently, unless the *output* of GAIN is widely viewed by the aviation community as providing value-added information to help improve the system, the community will simply stop wasting its time providing *input*. Here again, the savings will provide a powerful incentive to encourage the development of analytical tools that will efficiently and effectively accomplish this result of turning large quantities of input into value-added output.

D. <u>Existing Information Collection and Analysis Activities</u>

The FAA did not originate the concept of using aviation safety information proactively. To the contrary, there were already many activities of this type all over the world before the FAA proposed GAIN. Instead, the FAA proposed GAIN in an effort to be a *facilitator* to help bring the numerous information collecting, analyzing, and sharing activities around the world into a more unified international network.

Among the world leaders in this endeavor are the CAA and some of the airlines in the United Kingdom, where flight data recorders have been routinely accessed as a source of valuable information for decades. Thus, it is significant that the CAA joined the FAA to ask the Royal Aeronautical Society to host the second GAIN conference in London in May 1997. Similarly, British Airways developed the British Airways Safety Information System, or BASIS, and hundreds of airlines and other aviation entities all over the world are now using it.

In addition, in 1996, the French Academie Nationale de L'Air et de L'Espace published a document entitled "Feedback From Experience in Civil Transport Aviation" that recommended a proposal to collect, analyze, and disseminate aviation safety information, which GAIN closely resembles. Some of the Scandinavian countries have been reading flight data recorders routinely for many years; Japan Air Lines has had a proactive information program for several years; and proactive aviation safety information activities have been pursued in the former Soviet Union. And who better to talk about eliminating accidents than Australia, where their commercial accident rate is *zero*.

E. <u>A Major Obstacle: Fear of Information Misuse</u>

One of the major problems with systematically collecting and analyzing large quantities of information is that information is a very powerful tool; and like any powerful tool, it can be used properly with great benefit, or it can be used improperly and cause considerable harm. There are several possible ways in which this information can be misused, and many of them are largely responsible for the concerns that are being expressed about information sharing programs.

1. <u>Punishment/Enforcement</u>. First, company management and/or regulatory authorities might use the information for punitive or enforcement purposes. Thus, a pilot might be reluctant to report about an approach that did not go well because of a confusing approach chart, or a mechanic might be reluctant to report about an improper installation that resulted from a confusing page in

the maintenance manual, because of fear that management or the government might disagree that the approach chart or maintenance manual was confusing and then punish the pilot or mechanic.

Such punishment causes two problems. First, the confusing approach chart or maintenance manual page will still be in use in the system, potentially confusing other pilots or mechanics.

Second, and far worse, is that such punishment, in effect, "shoots the messenger." By shooting *one* messenger, management or the government *effectively guarantees that they will never again hear from any other messengers*. This, in turn, guarantees that those problems in the "unreported occurrences" part of the pyramid will remain unreported – until, of course, they cause an accident or incident, and we hear once again at the accident hearing that, "We all knew about that."

Some government regulatory agencies, notably the UK CAA, announced long ago that, absent egregious behavior (*e.g.*, wrongdoing that is intentional, criminal, or otherwise an outlier), they would not shoot the messenger, and encouraged their airlines also to take the same approach. That is clearly one of the major reasons why the UK has some of the world's leading aviation safety information sharing programs, both government-operated and privately owned. In the U.S., the FAA made a similar commitment in 1998 in its Flight Operations Quality Assurance (FOQA) program to access flight data recorder information routinely, and Congress recently enacted legislation that requires the FAA to create a rule to that effect. The type of facilitating environment created by the CAA is clearly a must in order for effective aviation safety information collection and sharing programs to be created and maintained.

Similarly, in order to get information from pilots, mechanics, and others for BASIS, British Airways gave assurances that they would also not shoot the messenger. Many other airlines around the world are beginning to conclude that they must do the same in order to be able to obtain the information they need to be proactive about safety.

2. Public Access. Another problem in some countries is public access, including media access, to information that is held by government agencies in certain countries. This problem does not affect the ability of the aviation community to create GAIN, but it could affect the ability of government agencies in some countries to receive information from GAIN. Thus, in 1996 the FAA obtained legislation that requires the agency to protect voluntarily supplied aviation safety information from public disclosure. This will not deprive the public of any information to which it would otherwise have access, because the agency would not otherwise receive the information; but on the other hand, there is a significant public benefit for the FAA to have the information because it helps the FAA prevent accidents and incidents. The FAA is now developing regulations to implement that legislation.

As with the enforcement problem, countries that have public information access problems will encounter difficulty getting the information they need to be proactive unless measures are taken, such as the legislation that the FAA obtained, to create the type of environment that can facilitate and encourage information sharing programs to be created and maintained.

3. Criminal Sanctions. A problem in some countries is the fear of criminal prosecution for regulatory infractions. Such a fear would be an obvious obstacle to the flow of aviation safety

information. This has not historically been a major problem in the U.S., but the trend from some recent accidents is troubling.

4. Civil Litigation. Probably the most significant problem, certainly in the U.S., is the concern that the information will be used against the reporter in accident litigation. Some have suggested that, as was done in relation to the public disclosure issue, the FAA should seek legislation to protect aviation safety information from disclosure in litigation. In comparison with the public disclosure issue, however, the chances of obtaining such legislation are probably very remote; and a failed attempt to obtain such legislation could exacerbate the situation further because these disclosure issues are now determined case by case, and a judge who is considering this issue might conclude that a court should not give protection that Congress refused to give.

The concern is that any information collected by an entity would be discoverable in litigation and could be used against the entity. Notwithstanding that concern, the programs underway to date have resulted in so many success stories that many entities are now becoming concerned that *failure* to collect information systematically could be a basis for liability.

F. GAIN Progress

In 1996, the FAA published a concept paper to obtain public comment about GAIN. Since then, there have been three GAIN conferences, each moving the concept further toward implementation. Although the FAA proposed GAIN, industry has moved into the leadership role. Industry has taken over sponsorship of the conferences, as reflected by the fact that the third conference, in Long Beach, California, in November 1998, was sponsored by United Airlines, and the next conference, in Paris, France, June 14-15, 2000, will be co-sponsored by Air France and Airbus Industrie. The GAIN Steering Committee is led by industry and includes airlines, manufacturers, the military, unions representing pilots, mechanics, and air traffic controllers, general aviation, and the Flight Safety Foundation.

Although the FAA is not a member of the Steering Committee, governments around the world will play a significant role in GAIN, both as providers of information and as users. Accordingly, the International Civil Aviation Organization (ICAO), the aviation arm of the United Nations with more than 180 governments as members, has become involved and has taken several actions that will help facilitate the establishment of information collection, analysis, and sharing programs around the world. Also being created is a GAIN government advisory committee, through which governments can determine how they can best assist with, and benefit from, the development of GAIN-type programs.

Most significantly, GAIN's work of helping members of the worldwide aviation community develop their individual GAIN-type programs is being done almost entirely by four working groups – Aviation Operator Safety Practices, Analytical Methods and Tools, Global Information Sharing Prototypes, and Reducing Impediments. All of the GAIN working groups are populated by volunteers from the worldwide aviation community, mostly from industry but also from various governments. They are making great strides in their respective areas, and will demonstrate their products and other results in the upcoming Paris conference.

G. Conclusion

There are many programs around the world that are already using aviation safety information proactively to prevent accidents. Recognizing that no single element of the aviation community can improve safety by itself, all facets of the aviation community are working together in this endeavor – airlines, manufacturers, pilots, mechanics, flight attendants, dispatchers, regulatory authorities, the military, academia, suppliers, the insurance industry, and others. The opportunity exists as never before to bring these programs together, to their mutual benefit, into an international network to collect and use information to improve worldwide aviation safety, and GAIN is helping that concept to become a reality. As FAA Administrator Jane Garvey noted in her luncheon remarks at the third GAIN conference in Long Beach, California, in November, 1998, "GAIN is one of our best hopes for enhancing aviation safety in the next century."

LEGAL ISSUES – THE PILOT'S PERSPECTIVE

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Unless and until a particular piece of information is rendered anonymous as to the operating crew, it presents a threat. To encourage complete and permanent de-identification, I will try and define the nature of the threat and the measures available to minimize it.

At the outset it must be clear that I am only discussing the nature of the information initially collected, the "raw data". The protections afforded information between the time of initial collection and distribution are governed by national law, corporate policy and collectively bargained agreements.

Pilots are exposed to three levels of threat by prosecution:

A. DISCIPLINE – for any infraction ranging from "counseling" to discharge which can have drastic consequences on the future career of the pilot.

B. CERTIFICATE ACTION BY THE REGULATORY AUTHORITY – variable from recertification to revocation of an individual airman certificate. While discipline can be levied without certificate action, certificate act invariably will be accompanied by discipline.

C. CRIMINAL LIABILITY – this can range from misdemeanor (noise rule violation, collision with object while taxiing) to major felon of a passenger due to negligent operation) to capital felony (intentional misconduct resulting in death).

To understand the extent of the threat, a review of certain evidentiary principles is in order. Direct evidence of an event may be obvious (crash site) or concealed (descent below minimum safe altitude.) In either case, all that is revealed by the direct evidence is that an event occurred. To determine culpability, circumstantial evidence or recorded data will have to be utilized. This category of evidence will require subjective interpretation by the finder of fact.

In an accident scenario or an observed violation of the regulations, all facts not otherwise protected are available for prosecution at the various levels. The most obvious example of information which is or should be protected from such use is the Cockpit Voice Recorder. Annex 13 to ICAO incorporates a standard requiring confidentiality of the cockpit voice recordings (para. 5.12). The rationale behind confidentiality is the recognition of the importance of the information to accident investigation and hence to aviation safety and an acknowledgement of the invasion of privacy which it represents to the crewmembers. As the crewmembers have the ability to disable the recorder if they feel that their privacy interests are not adequately protected, ICAO encourages national legislation which prevents abuse of the data, either by the prosecutorial function or by the media. This seems to be no more than a logical extension of legal principles which oppose forced self-incrimination. Products of forced self-incrimination should not, at the least, be admissible in furtherance of a prosecution. For cockpit voice recordings, this logic is clear. For other forms of information collection, the argument becomes somewhat more difficult, but the required balancing of interests remains.

Written reports fall into two categories: Voluntary reports, if not protected, will not be made. Voluntary reports if not encouraged, will be few and far between. Mandatory reports, if not protected will be so perfunctory as to be useless. The same rules apply to post flight interviews.

By far, the most prolific source of information is recorded data. As with voice recordings, data recordings must balance the enhancement of aviation safety against the privacy interests of the operating crew, and the requirement that evidence gained by forced self-incrimination not be used as a basis for prosecution. It is not clear that the recording of a parameter rises to the same level of privacy invasion or self-incrimination as does a voice recording. Further, of the wide variety of sources of recorded data, flight crews typically do not have the option of disabling the entire data stream.

In all cases, crew cooperation and endorsement of the information collection scheme would seem to be a practical prerequisite, easily accomplished. In a sense, the crews are the ultimate users of the collected information as a means to enhance the safety and efficiency of the operation. Coercive measures would seem antithetical to the ultimate program goals.

As stated above, a combination of legal protection and industrial agreement should result in, at a minimum, confidentiality of the collected information. Some exclusionary protection should be afforded to prevent prosecutorial abuse. These will serve to answer the protection question within a particular carrier and jurisdiction. As information moves into the Analysis Phase, national law, corporate policy and industrial agreement no longer provide a protective shield against abuse. The only answer to the protection question in the global arena is complete and permanent de-identification. If the identity key is retained by any party after a time interval suitable to local analysis, legal process may be employed to compel the production of the identity information.

The pilot community insists that the balance will always result in the requirement for anonymity of all information collected. We further encourage all operators and jurisdictions to evaluate the balance inherent in such coercive data acquisition methods and to resolve it in favor of a data policy calculated to maximize the cooperation of the pilot community in reaching the air safety goals.