Lessons Learned in Aircraft Design

The Devil is in the Details

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Introduction

The purpose of this book is to present examples of lessons learned in airplane design since 1945. The lessons are largely drawn from the aircraft design and accident/incident literature. The author hopes that this book will contribute to the safety of flight.

In Chapter 1 a brief summary is presented of safety statistics, certification and operational standards, safety standards and their relationship to design in general.

In Chapters 2-9 accident/incident discussions are presented in the following areas:

Chapter 2. Operational Experience
Chapter 3. Structural Design
Chapter 4. Flight Control System Design
Chapter 5. Powerplant Installation Design
Chapter 6. Systems Design
Chapter 7. Manufacturing and Maintenance
Chapter 8. Aerodynamic Design
Chapter 9. Configuration Design and Aircraft Sizing

In each case the discussion starts with the recounting of a problem which arose. Then the probable cause of the problem is identified. Next, one or more solutions are indicated. Finally a lesson learned is formulated.

The decision to place a given accident or incident discussion in anyone of chapters 2-8 is, in many cases, an arbitrary one. For that reason, a listing and cross-listing of all incidents and accidents is presented in Appendix A.

Finally, since many designers will eventually become program managers, it is instructive to recount some trials and tribulations associated with marketing, pricing and program decision making. That is done in Chapter 9.

An important paradigm in airplane design is and has been that single point failures in flight crucial areas of the structure or systems should not cause catastrophic results. Many aircraft certification requirements in the FAR's and/or JAR's demand this to be the case. As will be seen, in several instances this paradigm was not adhered to and/or the corresponding certification requirement was violated. Most airplanes should be viewed as an ensemble of systems the failure of any one of which can easily have "difficult to foresee" consequences. Design teams and individual designers should do their utmost to actually foresee those consequences and from that draw inferences for a different design approach.

It is noted here that design engineers, designated engineering representatives (DER's), maintenance personnel, and pilots (in other words humans) often directly affect critical airworthiness areas by their design decision making, by their maintenance actions, and by their in-flight actions.

It is the contention of the author that the single point failure paradigm in design should include the possibility of human error. Human error in the design sense, in the maintenance sense and in the cockpit action sense. Doing this should prevent a single error by humans from causing serious accidents.

Operational experience can and does often act as a precursor of a future catastrophic series of events. These precursors should be taken seriously by all involved in the design, maintenance and operation of airplanes. Several examples where precursor events were ignored are given in this book. Precursor events, when communicated in a timely and systematic manner to those involved in design, maintenance and operation, can often prevent catastrophes from occurring.

As will be shown by many examples in this book, safety of airplanes often starts in the design phase. However, sometimes the certification process itself, for whatever reason, fails. Examples of that type of failure are also given.

The author hopes that this book will:

- be useful to practicing design engineers, test pilots and program managers
- find its way into the classroom to help in the education of future aircraft designers and engineering/maintenance personnel.

Some basic knowledge of aircraft flight characteristics and applied aerodynamics is desirable to follow many parts of the text.

Lawrence, KS 2016

Chapter 1

Safety, Incidents, Accidents and their Relationship to Aircraft Design in Perspective

"Safety is no accident"

William H. Tench, 1985

1.1 Introduction

Since World War II air transportation in the USA has become widely accepted as both safe and economically viable to business and individuals. The objective of this book is to further enhance safety by encouraging good design practices. Because safety is a relative concept it is necessary to define the scope of what will be considered.

The scope of this book covers commercial and general aviation, fixed wing aircraft. Rotary aircraft are not considered in this text. A few military aircraft examples are included in cases where the author decided that a useful civil application exists.

In the United States, commercial and general aviation aircraft are certified by a set of minimum airworthiness standards laid down in the Code of Federal Regulations (CFR) as FAR 25, and FAR 23 respectively. These codes are accessible on the following website: www.gpoaccess.gov

The Federal Aviation Administration (FAA) is responsible for assuring that aircraft operating in the U.S.A. are certified to these standards.

In addition, operators of these aircraft must adhere to a set of operational requirements. These operational requirements are laid down in:

CFR Part 121 for major airlines and cargo carriers that fly large transport-category aircraft and CFR Part 135 for commercial air carriers, often referred to as commuter airlines, (i.e. scheduled Part 135) and air taxis (i.e. unscheduled Part 135).

In addition FAA is responsible for assuring that the operations conducted with these aircraft meet these operational requirements. Examples of operational requirements are those dealing with fuel reserves, weight and balance procedures, crew certification, and training standards and procedures.

The FAA is also responsible for overseeing manufacturing, maintenance and modification facilities that deal with aircraft or aircraft components.

In the U.S.A., all aircraft accidents and incidents must (by law) be investigated by the National Transportation and Safety Board (NTSB). This board is independent from the Federal Aviation Administration (FAA). The Safety Board must publicly publish the results of its investigations and recommend to the FAA any changes that should be made to enhance safety.

The following definitions are used to describe what is meant by accidents and incidents:

An accident is defined in 49 CFR 830.2 as, "an occurrence associated with the operation of an aircraft which takes place between the time any person boards the aircraft with the intention of flight and the time all such persons have disembarked, and in which any person suffers death or serious injury, or in which the aircraft receives substantial damage."

An incident is defined in 49 CFR 830.2 as, "an occurrence other than an accident which is associated with the operation of an aircraft and that affects or could affect the safety of operations."

The safety aspects of commercial and general aviation airplanes can be put in perspective by citing some statistics for 1999 (commercial aircraft) and 1998 (general aviation aircraft) taken from Refs. 1.1 and 1.2 respectively.

1.2 Commercial Airplanes

In 1999 there were 8,228 commercial aircraft operating under Part 121 or Part 135 air carrier regulations. The distribution in terms of aircraft types was as follows:

Turbojet or turbofan	5,630
Turbo-propeller	1,788
Piston-propeller	688
Rotary	122

Together these aircraft produced 488,357,000,000 passenger miles for 674,100,000 enplanements (average distance covered per flight 724 miles).

In 1999 there were 51 accidents with Part 121 aircraft, 13 with scheduled Part 135 and 73 with unscheduled Part 135 aircraft. In total there were 137 accidents. Figure 1.1 shows a comparison of Part 121 and scheduled Part 135 accidents during the period 1990-1999.

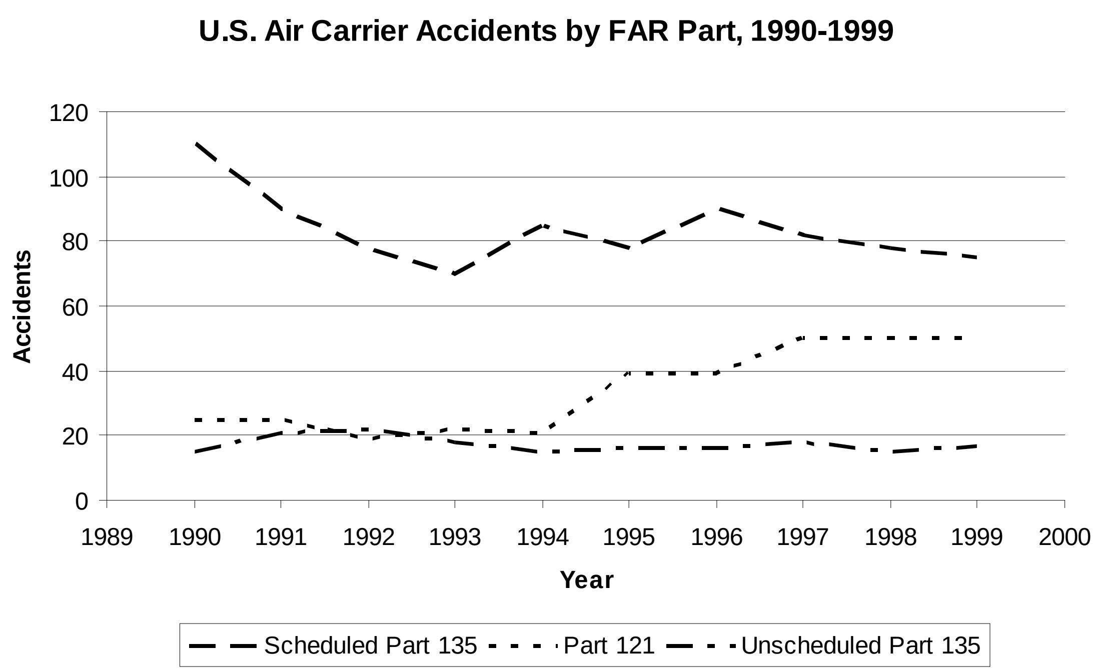


Figure 1.1 Part 121 and Scheduled Part 135 Accident Rates over a Ten Year Period (Ref. 1.1)

For 1999 the records in Ref. 1.1 indicate the following distribution of fatalities and injuries:

Table 1.1 Distribution of Fatalities and Injuries in 1999

Injury Type	Part 121	Scheduled Part 135	Unscheduled Part 135
Fatal	11	12	38
Serious	65	2	14
Minor	181	1	31

Taking the total of 137 accidents leads to the following accident rates:

Total flight hours	20.1×10^6	accident rate	6.8 per million flight hours
Total departures	11.8×10^6	accident rate	11.6 per million departures
Total miles flown	7.1×10^9	accident rate	0.02 per million miles

These statistics do not provide an insight into the relative safety of specific aircraft types. Table 1.1 gives that insight for the most prevailing commercial aircraft types.

Even though the data for the Boeing 777 (excellent as these are) are not as yet statistically significant, it is of interest to observe from Table 1.2 that the fly-by-wire airplanes (those in bold type face) have the lowest fatal accident rate.

In Ref.1.1 the causes of commercial aircraft accidents can be put into three broad categories. For 1999 the results were:

73% were determined to have personnel related causes (mostly pilot related)
23% were determined to have aircraft related causes
39% were determined to have environmental related causes (mostly weather related)

These numbers do not add to 100% because some accidents were determined to fall in more than one category. From a designer's viewpoint the good news is that aircraft related causes occur less frequently than the other causes. However, as will be seen in several examples in Chapters 2 through 8, personnel related causes are in many cases caused by a design problem. The author refers to such cases as "design induced crew errors" which definitely have design implications.

Table 1.2 Fatal Event Rate per Million Flights (From AirSafe.com as of 11-10-2006) [1]

Model	Rate	Flights (millions)	FLE*	Events
Jet Transports				
Airbus A300	0.62	9.72	5.99	9
Airbus A310	1.39	3.75	5.23	6
Airbus A320/319/321	**0.17**	21.43	3.61	6
Boeing 727	0.50	74.50	37.20	48
Boeing737-100/200	0.57	54.96	31.54	44
Boeing 737-300/400/500	0.22	50.00	10.99	14
Boeing 737 (all models)	0.37	118.86	43.53	59
Boeing 747	0.84	16.26	13.73	28
Boeing 757	0.37	14.71	5.40	7
Boeing 767	0.47	11.76	5.50	6
Boeing 777	**0.00**	2.00	0.00	0
Boeing DC-9	0.59	59.56	35.37	43
Boeing DC-10	0.70	8.49	5.91	15
Boeing MD-11	0.70	1.45	1.02	3
Boeing MD-80/MD-90	0.22	33.33	7.37	12
BAe 146/RJ-100	0.58	7.69	4.49	6
Fokker F-28	2.56	6.03	15.45	21
Fokker 70/Fokker 100	0.28	6.67	1.87	4
Lockheed L-1011	0.49	5.19	2.54	5
Turboprop Transports				
ATR 42 and ATR 72	0.26	13.20	3.40	4
Embraer Bandeirante	3.07	7.50	23.00	28
Embraer Brasilia	0.71	7.40	5.27	6
SAAB 340	0.19	11.2	2.10	3

[1] *FLE-Full Loss Equivalent: this is the sum of the proportions of passengers killed for each fatal event. The fatal event rate for a set of fatal events is found by dividing the total FLE by the number of flights in millions.

1.3 General Aviation Airplanes

In 1998 there were approximately 196,700 general aviation airplanes operating. The rough distribution in terms of aircraft types was as follows:

Homebuilt	13,000
Single-engine piston-propeller	144,000
Multi-engine piston-propeller	17,000
Turbojet or turbofan	5,600
Turbo-propeller	5,200
Rotary	6,300
Gliders	2,100
Balloon	3,500

For 1998 the number of general aviation flight hours was estimated to be 25,518,000.

There were a total of 1,904 accidents, 364 of which resulted in a total of 624 fatalities. The general aviation accident rates for 1998 are:

Total flight hours	$25.518\text{x}10^6$	accident rate	74.6 per million flight hours
Total departures	$11.8\text{x}10^6$	accident rate	161.4 per million departures
Total miles flown	$7.1\text{x}10^9$	accident rate	0.27 per million miles

It is clear that commercial aircraft enjoy a considerable safety advantage over general aviation aircraft (by more than a factor of 10). However, as Figure 1.2 shows, the corporate sector (mostly flown with jets operated by professional pilots) shows a safety level very similar to that of commercial, Part 121 operations.

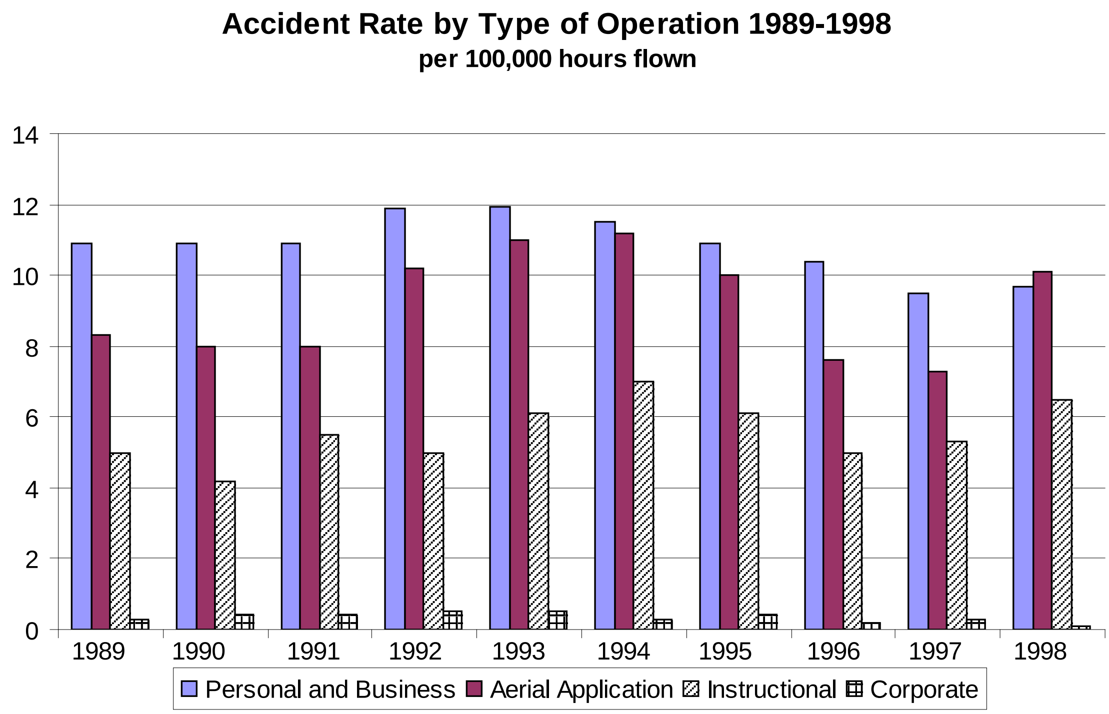


Figure 1.2 General Aviation Accident Rate by Type of Operation (Ref. 1.2)

The causes of these accidents can be put into three broad categories. For 1998 the results are:

90% were determined to have personnel related causes (mostly pilot related)
36% were determined to have aircraft related causes
37% were determined to have environmental related causes (mostly weather related)

These numbers do not add to 100% because some accidents were determined to fall in more than one category.

Because of the widely differing types of general aviation aircraft, the widely differing pilot qualifications and the widely differing types of operations conducted, the reader is urged to consult Ref. 1.2 for a more balanced discussion of general aviation accidents.

Taking personnel and environment related causes together it seems reasonable to conclude that the human being is a weak link in the safety chain. More automation in the cockpit is often pointed to as one way to lower the personnel related causal factor for accidents. However, if and when that is done, the major causal factor for accidents will be aircraft related and therefore have design implications.

Chapters 2-8 of this book focus mostly on design lessons learned from accidents and incidents where better design approaches might have made a difference.

The reader will observe the frequent occurrence of Murphy's Law. This law, stated in terms of aircraft design, manufacturing, maintenance, inspection and operations can be restated as:

- If an error in design (configuration, aerodynamic, structural, system) can be made, it will be made.
- If an error in a manufacturing procedure or process can be made, it will be made.
- If an error in a maintenance procedure or process can be made, it will be made.
- If an error in an inspection procedure or process can be made, it will be made.
- If an error in an operational procedure or process can be made, it will be made.

The paradigm of "no single cause or failure shall cause a catastrophic accident" should be modified to include all these "Murphy's."

When finished reading all cases discussed in this book, the reader should also consider what the implication of all these "lessons learned" is to the certification process of airplanes. An excellent document in this regard is Ref. 1.3. This document was recommended to the author as a starting point for this book by Mr. Marvin Nuss of the Kansas City FAA office. It is well worth reading.

Chapter 2

Design Lessons Learned from Operational Experience

“If it can happen, it will happen”

Version of Murphy’s Law

2.1 Introduction

In this chapter a number of operational experiences, which have had design implications and/or caused airworthiness rules to be amended, are reviewed. Where applicable, causes and solutions are described and lessons learned are stated. Connections with other areas of design are identified in Appendix A.

2.2 Gust Lock on During Take-off

2.2.1 Problem

In May of 1947 a United Air Lines Douglas C-54 (Figure 2.1) crashed while attempting a take-off from La Guardia Field, New York.

Figure 2.1 Douglas C-54 (Not accident aircraft or airline, Courtesy G. Helmer)

Of the 48 occupants, 43 were killed, four sustained serious injuries and one, the pilot, received only minor injuries. The aircraft was destroyed by fire.

2.2.2 Cause

Ref. 2.1 (pages 11-17) gives the probable cause of this accident as the inability of the pilot to actuate the controls due to the gust lock being on, resulting in the pilot’s decision to discontinue the take-off at a point too far down the runway to permit stopping within airport boundaries.

The following is quoted from Ref. 2.1: “An examination of the gust lock control in several of United’s C-54’s disclosed that the mechanism had been modified to allow the locking handle located to the immediate right of the pilot’s seat to remain in the “up” or “on” position without being held by either the gust lock warning tape or by a locking pin attached to the tape. Very slight pressure on the handle would release the lock; however, if no tape was strung from the reel at the top of the cockpit to the locking handle, no visual warning would be given to the pilot before take-off that the control surfaces of the aircraft were actually locked.”

Because the take-off, under the pressure of a weather front, was hurried, it is believed that the airplane began the take-off roll with the controls locked.

2.2.3 Solution

At that time a special Presidential Board was charged with studying ways to make commercial aviation safer. This Board requested that the manufacturer of this airplane redesign the gust lock system to avoid its use during take-off.

2.2.4 Lesson and Design Suggestion

Murphy's Law would state that if a gust lock can be left on during take-off, it will be. Food for thought for aircraft designers. Gust lock systems should be designed such that take-off is impossible with the lock(s) in the "on" position. According to Ref. 2.1 United Airlines had modified its fleet according to the description in Section 2.2.2. Ref. 2.1 does not indicate that this was a re-certification issue. The author's view is that it should have been.

2.3 Center of Gravity Too Far Aft I

2.3.1 Problem

In 1950 an AVRO Tudor 2 (Figure 2.2) crashed on an approach to landing.

Figure 2.2 Model of AVRO Tudor 2 (Courtesy www.collectorsaircraft.com)

The pilot, upon recognizing that the airplane was too low to make the runway, applied power. The airplane was observed to pitch up sharply, stall and crash. Even though there was no fire, no one survived (Ref. 2.2, pp 11-17).

2.3.2 Cause

The investigation showed that the airplane had taken off with a full load of passengers (78) but was still about 1,000 lbs below its maximum certified take-off weight. In the flight manual of the airplane were instructions to flight crews that, with a full load of passengers at least 2,000 lbs of luggage, cargo or ballast had to be loaded in the forward cargo hold to keep the center of gravity at or forward of the aft allowable limit. The pilots had not performed the required check of the c.g. location prior to take-off. Post crash analysis of the luggage indicated that the amount actually loaded in the forward bay was 1,031 lbs. This would have put the c.g. at take-off well behind the aft limit. Therefore, when the airplane took off, it was longitudinally unstable and required a large amount of trailing edge down elevator to trim. As it turned out, the pilots had no difficulty flying and trimming the airplane in that condition.

However, on final approach, with the airplane too low to make the runway without a change in power, the pilot applied a significant amount of power. The resulting propeller normal forces, forward of the c.g., pitched the airplane up sharply and there was no longer enough trailing edge down elevator to control the pitch-up.

2.3.3 Solution

Flight crews should never forget to verify the actual c.g. location of an airplane prior to take-off. Pre-take-off check lists should always be completed.

2.3.4 Lessons

Murphy's Law would state that if an airplane can be loaded the wrong way, it will be. Food for thought for operators!

From a design viewpoint it appears that this airplane was unduly sensitive to aft center of gravity loading when carrying a full load of passengers. The designers might have foreseen this and made appropriate design adjustments to reduce this sensitivity. For example: more nose-down elevator control power would have helped.

From an operational viewpoint, it appears that a system using force transducers installed in the landing gear legs could have been useful in alerting the pilot about the true center of gravity location and take-off weight while taxiing toward take-off. Such systems are now common in several modern transports.

Caveat: Because airplanes parked on the ramp are sensitive to winds and gusts (low wing loading and high aspect ratio wings are particularly sensitive) the system outputs have to be corrected for wind and gust effects. Even in Boeing 747 airplanes these systems are used only as a check on more conventional weight and balance calculations.

2.4 Minimum Un-Stick Speed I

2.4.1 Problem

In October of 1952 a DeHavilland Comet (Figure 2.3), taking off from Rome, Italy, settled back beyond the runway shortly after lift-off.

Figure 2.3 DeHavilland Comet (Not accident aircraft or airline, Courtesy J. de Groot)

Luckily there were no injuries although the airplane was seriously damaged. Five months later, in March of 1953 another Comet, taking off from Karachi, Pakistan, did not even un-stick and crashed. This time there were no survivors. In both cases the airplanes were observed to have an unusually high pitch angle relative to the ground during the later part of the ground-run (Ref. 2.3, pp 12-13).

The reader might observe that the second accident had as a pre-cursor the incident mentioned before. A valid question would be why no conclusions were drawn by the designers or by the airworthiness authorities.

2.4.2 Cause

The cause was found to be massive flow separation (wing stall) over the wing when the airplane is over-rotated early in the take-off run. This flow separation increases the drag so much that there is no longer enough thrust (with all engines operating) to accelerate the airplane. Figure 2.4 shows the cause of the problem.

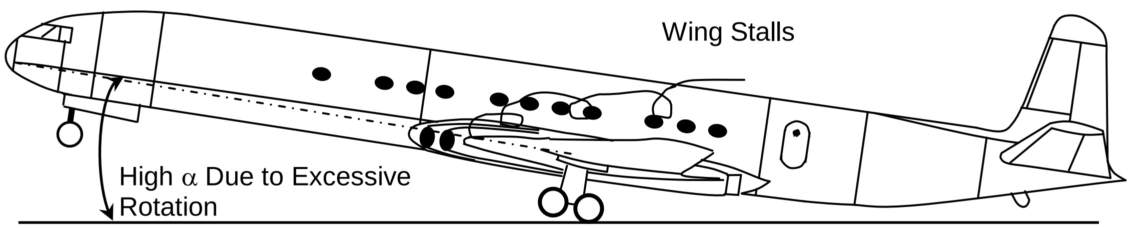


Figure 2.4 Wing Stalls due to High Take-off Rotation Angle

It can be argued that pilots should not rotate the airplane to such an excessive attitude. The problem with this argument is that it is very easy to make this mistake in many airplanes, particularly if the attitude indicator does not allow for an accurate reading of the take-off attitude. Also, in some airplanes the wing-fuselage lift grows very rapidly upon rotation and imparts a tendency toward auto-rotation on the airplane, particularly if the distance from the main landing gear to the wing-fuselage aerodynamic center is large.

2.4.3 Solution

In the case of the Comet, DeHavilland cambered the wing leading edge of the Comets down a bit to prevent wing stall during excessive take-off rotation. Added camber extends the stall angle of attack to a higher value as shown in Figure 2.5.

Modern jet transports are designed to meet the so-called un-stick requirement. Any transport must have the ability to safely un-stick even if a pilot pulls the control column completely back early in the take-off run. What this implies for wing design is that (in the take-off configuration) it may not stall even with the rear fuselage in contact with the runway. Also, the airplane, with all engines operating (AEO), must be able to continue to accelerate and become airborne.

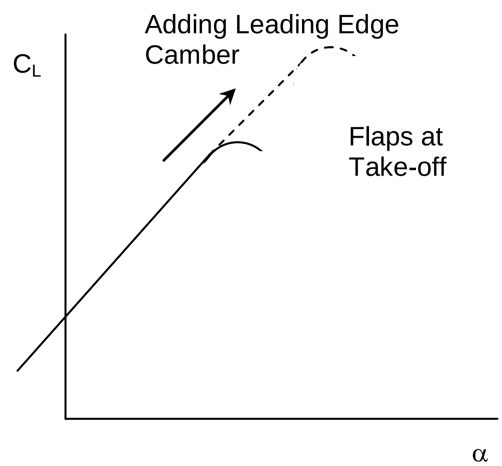


Figure 2.5 Effect of Leading Edge Camber on the Stall Angle of Attack

2.4.4 Lesson

Murphy's Law would state that if an airplane can be over-rotated at take-off, it will be. The consequence of such an event must be benign. Designers should not need a regulation to design accordingly.

2.5 Accidental Retraction of Landing Gear during Landing Roll

2.5.1 Problem

In September of 1954 a Trans World Airlines Martin 404 (Figure 2.6) experienced a partial retraction of the landing gear during a landing roll at Baer Field, Ft. Wayne in Indiana.

Figure 2.6 Martin 404 (Not accident aircraft, Courtesy www.prop-liners.com and Trans World Airlines)

During the evacuation, a few of the 30 passengers received bruises. The crew of 3 was not injured. There was no fire and the aircraft damage was relatively light.

2.5.2 Cause

Ref. 2.4 gives as the probable cause the accidental retraction of the landing gear during the early stages of the landing before the safety system became effective.

When the landing was made, the captain was in the left seat and the first officer in the right seat. The first officer was flying the airplane while the captain performed the duties of the first officer. The captain stated that when the airplane was firmly on the runway the first officer called for flaps up, cowl flaps open and propellers full increase r.p.m. The captain started the flaps up and, according to his testimony, while his right arm was brought back from the flap control (which is on the right side of the pedestal and has a spherical knob) his right hand struck the landing gear lever. He at once noted that the landing gear control was above its neutral position. He quickly actuated the control back to the down position but the propeller tips began striking the runway almost immediately.

The gear actuating control handle is located on the left side of the pedestal and has a cube-shaped knob at the end of the handle. The neutral position of this control is in the center of a vertically placed quadrant. The release of control is accomplished by lifting slightly and pulling the handle out of its detent against the pressure of a light spring. If the weight of the aircraft is compressing the gear's shock struts enough to actuate any one of the safety switches, the control handle cannot be moved to the "up" position on the quadrant. This safety factor, which is designed to minimize inadvertent retraction of the gear while on the ground, functioned normally. A movement of the gear's control handle to the "up" position is possible even though the aircraft does not skip or bounce during a landing, if none of the shock struts are compressed by the weight of the aircraft to the point where a safety switch is actuated. It is necessary that the struts compress about two inches before they actuate the switches. Any one, or any combination of these struts, if compressed to this point, will prevent inadvertent retraction.

2.5.3 Solutions

Following this accident, all company pilots were advised of the circumstances. In addition TWA initiated a program designed to preclude further occurrences.
Also, a guard has been installed over the landing gear control handle and deliberate action on the part of the pilot is now required before the handle can be actuated. TWA was also in the process of installing deceleration switches on all its Martin 404 aircraft which will lock the gear in "safety" immediately upon touchdown. Furthermore, the spring tension on the retract handle was checked to verify that this was within tolerance on all other aircraft.

2.5.4 Lesson

Murphy's Law would state that if the landing gear can be raised while in contact with the ground, it will be. In the design of cockpits the scenario which unfolded here must be considered and design conclusions drawn to prevent this.

2.6 Cowl Flaps Left Open Upon Flap Retraction

2.6.1 Problem

In April of 1956 A Northwest Airlines Boeing B-377 Stratocruiser (Figure 2.7) ditched in the Puget Sound near the Seattle-Tacoma Airport.

All occupants of the aircraft successfully evacuated the airplane but 4 of the 32 passengers and one of the crew of 6 drowned. The airplane was a total loss.

Figure 2.7 Boeing B-377 Stratocruiser (Not accident aircraft, Courtesy Mel Lawrence)

2.6.2 Cause

Ref. 2.5 states that the probable cause of this accident was: “the incorrect analysis of control difficulty which occurred on retraction of the wing flaps as a result of the flight engineer’s failure to close the engine cowl flaps, the analysis having been made under conditions of great urgency and within an extremely short period of time.”

After the ditching, the wing flaps were found fully retracted and the engine cowl flaps were found fully open. In this aircraft it is the flight engineer’s responsibility to close the full-open engine cowl flaps prior to take-off. The flight engineer stated that he was not certain the cowl flaps had been closed at that time.

The flight engineer was qualified on three different types of aircraft: DC-6, L-1049 and B-377. His B-377 time in the 90 days preceding the accident was 1 hour and 40 minutes. He testified that most of his flight time during the preceding year had been on L-1049 and DC-6 type aircraft.

This is relevant because of the fact that the cockpit cowl flap controls on the B-377's and L-1049's move in the opposite directions for the closing of these cowl flaps. The flight engineer testified that it was possible that he moved these controls in the wrong direction.

Take-off with fully open cowl flaps in the B-377 yields no distinguishable effect on control of the aircraft until the wing flaps are retracted. As the wing flaps are retracted, buffeting and lateral control difficulties begin almost immediately. The remedy is to extend the flaps. This has the effect to re-direct the wakes from the open cowl flaps downward relative to the horizontal tail.

2.6.3 Solution

Ref. 2.5 does not indicate that any recommendations were made to the CAA as a result of this accident. It would seem that in-flight training to cope with this type of mistake should have been a part of pilot type certification.

2.6.4 Lessons

Murphy's Law would state that if this type of mistake can be made, it will be made. Design engineers should draw conclusions from this in locating control levers in any new airplane.

In-flight training toward a type-certificate should involve training to cope with this type of mistake. Cross certifying of flight crew members on different aircraft can lead to trouble if critical controls move differently.

From an aerodynamic design viewpoint, this was a predictable scenario but, evidently it was not considered. It does not seem acceptable to have significant lateral control effects ensue as a result of mismanagement of cowl flaps. This should have been a certification issue.

2.7 Flight Characteristics with One Engine Inoperative

2.7.1 Problem

In August of 1959 an American Airlines Boeing 707-123 (Figure 2.8) crashed and burned in an open field near Calverton, Long Island, N.Y. during a training flight.

Figure 2.8 Boeing 707-123 (Not accident aircraft, Courtesy geminijets.com)

During this training flight several maneuvers were conducted to provide jet transition training to two captain-trainees. During its last approach to the Peconic River Airport (operated by Grumman personnel and used often by airlines for training) the airplane had its flaps extended 30 degrees and the gear down. The landing was to be made with 50% power reduction on the No. 3 and 4 engines to simulate one-engine-inoperative (OEI) conditions. The airplane yawed rapidly to the right to an estimated 17 degrees of (negative) sideslip. This is well beyond the 11-14 degrees which can be successfully controlled using full lateral control in this configuration. The crew apparently failed to recognize this and the airplane rolled to the right. When the aircraft passed the 90-degree bank position, it was yawed right about 20 degrees resulting in a roughly 30 degrees nose-down attitude. As the airplane passed the inverted position the yaw angle was reduced by corrective action consisting of advancing the throttles on the No. 3 and 4 engines and applying full left rudder and aileron. As the aircraft passed the 270 degree roll position it was in a zero yaw condition. Acceleration was held at approximately 2g which is in the buffet range and is the tightest pullout the airplane could make. At that point the power was nearly symmetrical. The airplane struck the ground in a nearly wings-level attitude.

2.7.2 Cause

Ref. 2.6 states the probable cause of this accident to be: “the crew’s failure to recognize and correct the development of excessive yaw which caused an unintentional rolling maneuver at an altitude too low to permit complete recovery.”

Swept wing transport airplanes are known to develop a large negative rolling moment due to sideslip. When in a sideslip maneuver close to the ground (such as in a simulated engine out maneuver) the resulting rolling moment is very large and timely application of lateral controls is needed to prevent large bank angles from occurring. Recovery from large bank angles close to the ground can be very problematic.

2.7.3 Solution

Do not conduct engine-out training maneuvers close to the ground. Subsequent to this accident the FAA discontinued the requirement that Boeing 707 aircraft make actual landings with simulated failure of 50% of the power units on one side during training flights, type ratings, and proficiency checks. These maneuvers may be simulated at high altitude.
Nowadays these maneuvers are also conducted in flight simulators.

2.7.4 Lesson

These maneuvers, which are relatively safe to carry out in straight wing transport airplanes, should have had no place in swept wing jets in the first place. Due to the large rolling moment due to sideslip (which is characteristic of aft swept wings) it was asking for trouble to do this close to the ground.

2.8 Tail Stall in Icing Conditions

2.8.1 Problem

In April of 1958 a Capital Airlines Vickers Viscount (Figure 2.9) crashed and burned near the Tri-City Airport, Freeland, Michigan. All 44 passengers and 3 crew members were killed.

Figure 2.9 Vickers Viscount (Not accident aircraft, Courtesy www.prop-liners.com and Capital Airlines)

2.8.2 Cause

In Ref. 2.7 the Board determines that the probable cause of this accident was: "an undetected accretion of ice on the horizontal stabilizer which, in conjunction with a specific airspeed and aircraft configuration, caused a loss of pitch control."

In 1963 a very similar accident occurred involving a Continental Airlines Viscount. For that reason, Ref. 2.7 is a revision of the Board's original report which was released on April 15, 1959. It is of interest to observe that Ref. 2.7 was not released until February 17, 1965, about seven years after the accident.

It is well known that if there is undetected icing on a horizontal tail with the airplane in trim, lowering of the flaps tends to increase the (negative) angle of attack on the tail. This change in tail angle of attack can cause the tail to stall if sufficient ice contamination is present on the tail. The following is quoted from Ref. 2.7, page 10:

"During the investigation of another Viscount accident in 1963 new information became available to the Board regarding the behavior of aircraft with a concave build-up of ice on the horizontal stabilizer. Included was information concerning two Viscount incidents.

In one case the Viscount operated in an icing condition an estimated 10 to 12 minutes. Because of the short time the aircraft was in clouds and a surface temperature of 37 degrees, airframe anti-icing was not used. When clear of clouds no ice was visible on the leading edge of the wings. When the landing flaps were lowered to 40 degrees at a reported airspeed of 135-140 kts, the nose dropped and the pilot was unable to arrest this action with the elevator control. Flaps were retracted to 32 degrees, control regained, and the aircraft landed safely. Four minutes after the touchdown, an examination of the aircraft revealed that the wings were clear of ice but a concave build-up was observed on the leading edges of the tail-plane and the vertical stabilizer. This ice was about ¾ inch thick at the center and 1.5 inches thick at the edges extending 3 inches above the horizontal tail.

A similar incident occurred to another aircraft which had been exposed to icing conditions an estimated two minutes after the leading edges of the wings were seen to be clear of ice. In this case the aircraft was recovered from severe pitching oscillations when the airspeed decreased to 130-135 kts. The oscillations began when the flaps were lowered to 40 degrees on the final approach. Examination of the flight recorder tape showed that the aircraft lost 200 ft during the pitching oscillations with peak accelerations of -0.76 and -2.3g before control was regained. When the aircraft was inspected after landing light rime ice was found on the wings and radome, the propellers were clean and dry, but the horizontal and vertical stabilizers had a concave build-up of rough rime ice. This cup-shaped ice was approximately 1 inch thick with horns extending diagonally upward and downward, approximately 1.5 inches into the air stream.

Additionally, there have been three aircraft accidents of which one was a Viscount, which the Board has determined were caused in whole or in part by an accretion of ice on the horizontal stabilizer that disrupted the airflow over and under the horizontal tail surfaces."

2.8.3 Solution

Ref. 2.7 does not indicate what (if any) action was recommended to the FAA.

2.8.4 Lessons

1. Operating airplanes in icing conditions requires very special care and knowledge. Anti- or de-icing equipment should be used to the extent possible.

2. If uncertainty exists about the ice-state of the horizontal tail, a flaps-up landing may be the safest solution. This prevents exposing the horizontal tail to a sudden (negative) change in angle of attack which might drive it into the stalled regime.

3. Pilots should be given a means to verify in flight whether or not there is ice on any lifting surface, especially the tail. Video cameras could offer such a solution.

2.9 Unsweeping Wings with the Flaps Down

2.9.1 Problem

In 1964, to celebrate the first flight of the F-111A (Figure 2.10), the management of the Fort Worth division of General Dynamics invited some top brass for a special demonstration.

The brass and other guests were seated on a grandstand erected on the ramp. The idea was for the airplane, after landing, to taxi and park in front of this grandstand. There were quite a number of high level guests who had arrived in their squadron hacks. Those airplanes had been parked on the ramp as well.

The F-111A made a successful demonstration flight and landed, as usual, with the wings swept forward and the big Fowler flaps deployed. When approaching the ramp area the test pilot thought that there was insufficient space between two of the parked airplanes for him to taxi through with the wings swept forward. Therefore, he engaged the wing-sweep system to sweep the wings aft. Crunch, crunch, crunch: the Fowler flaps were crushed against the side of the fuselage. All this in front of all the brass. Very embarrassing.

Figure 2.10 General Dynamics F-111A (Not accident aircraft, Courtesy NASA)

2.9.2 Cause and Solution

The cause was not that the pilot did what he did. The cause was the design engineers not foreseeing this event and designing a simple device which disables the wing sweep mechanism when the flaps are down.

2.9.3 Lesson

The designers might have expected that, if it is possible to sweep the wings aft with the flaps deployed, someone would do it. Always keep in mind Murphy's Law.

2.10 Center of Gravity too Far Forward

2.10.1 Problem

In September of 1966 an Airlift International Douglas DC-7C (Figure 2.11) did not respond to a pilot control input to rotate at the proper rotation speed. The attempted take-off abort maneuver was not successful and the airplane was partially destroyed during the overrun. Two of the four crew members were injured.

Figure 2.11 Douglas DC-7C (Not accident aircraft, Courtesy www.prop-liners.com and American Airlines)

2.10.2 Cause

The cause was determined to be improper loading of cargo so that the actual center of gravity was at 15.2% m.g.c. while the forward limit for this airplane is 18.8%. According to Ref. 2.8 this occurred because the loadmaster responsible for loading the airplane was neither experienced nor familiar with aircraft weight and balance techniques. As a result of the c.g. being too far forward, the airplane did not have adequate longitudinal control power to rotate at the rotation speed consistent with the take-off weight. Figure 2.12 illustrates the situation.

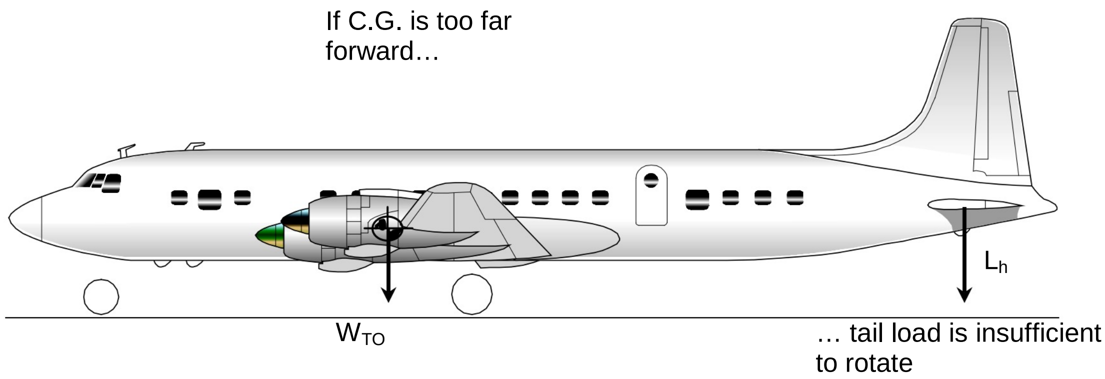


Figure 2.12 Forward Allowable C.G. Limited by Maximum Available Horizontal Tail Download

2.10.3 Solution

Tighten up loading as well as weight and balance procedures. Also, use qualified personnel when dealing with flight crucial aspects of airplane dispatching.

2.10.4 Lesson

See Section 2.2.4: a very similar problem, albeit at aft c.g.

2.11 Loss of Electrical Power Leading to Loss of Attitude Instrumentation

2.11.1 Problem

In January of 1969 a United Air Lines Boeing 727 (Figure 2.13) crashed into the Pacific Ocean following a night, instrument departure. There were no survivors.

Figure 2.13 Boeing 727 (Not accident aircraft, Courtesy www.al-airliners.be)

2.11.2 Cause

According to Ref. 2.9 the probable cause was: “loss of attitude orientation during a night, instrument departure in which the attitude instruments were disabled by loss of electrical power.” The NTSB was not able to determine why all generator power was lost, or why the standby electrical power system either was not activated or failed to function.
The following findings reported in Ref. 2.9 are of interest from a design viewpoint:

> Prior to the accident flight the airplane had been operating for 42 flight hours with the No.3 generator inoperative. This was allowed by the Minimum Equipment List (M.E.L).
>
> The discrepancy which caused the No.3 generator to be rendered inoperative had not been corrected and probably was associated with its electrical control panel.
> The flight experienced a fire warning on the No.1 engine during climb-out and the engine was shut down.
>
> Shortly after shutdown of the No.1 engine, electrical power from the remaining No.2 generator was lost.
>
> Following loss of all generator power, the standby electrical system was not activated or failed to function.
>
> The pilots were without reliable attitude indication from the point in time the No.2 generator was lost until impact.

2.11.3 Solution

The NTSB recommended that the FAA require that automatic switching of essential power to standby power be made mandatory for all turbine powered airplanes.

2.11.4 Lessons

1. Designers might have foreseen this situation in view of the M.E.L. allowing continued revenue flights with one generator inoperative. Consistently applying a "what-if" analysis surely would have led the designers to conclude that either the M.E.L. should be revised or automatic switching to alternate power be incorporated. This should have been a certification issue but was not.

2. Pilot workload under instrument conditions plus any emergency becomes very high. Designers should not assume that pilots will have the opportunity to trouble-shoot and perhaps figure out what to do in high workload cases such as this.

2.12 Flight Characteristics with one Engine Inoperative II

2.12.1 Problem

In June of 1969 a Japan Airlines Convair 880 (Figure 2.14) was on a training flight. The airplane crashed when a flight instructor reduced power on the No.4 engine shortly after lift-off.

Figure 2.14 Convair 880 (Not accident aircraft, Courtesy Mel Lawrence)

Following the reduction in power of the No.4 engine, the airplane began yawing to the right, the right wing dipped and the airplane crashed and burned. Of the five crew members on board, two survived with serious injuries.

2.12.2 Cause

According to Ref. 2.10 the probable cause was: “the delayed corrective action during a simulated critical-engine-out take-off maneuver resulting in an excessive sideslip from which full recovery could not be effected.”

It is important to review the physics of this problem. When a jet transport experiences a critical OEI condition the tendency of the airplane is to begin to yaw in the direction of the inoperative engine. In this case engine No.4 was brought back to flight idle right after lift-off. The yaw to the right produced a negative sideslip angle. Because a swept wing transport, at low speed (i.e. relatively high lift coefficient) develops a large negative rolling moment due to sideslip the airplane also began rolling to the right. If the pilot flying the airplane delays applying almost full rudder deflection (push on the left pedal to force the rudder trailing edge to the left) recovery before contacting the ground becomes questionable.

Swept wing transports are much more critical in this regard than transports with un-swept wings, the difference being the magnitude of the rolling moment due to sideslip. These types of training maneuvers were pretty much standard on the older DC-7, Stratocruiser and Constellation airplanes. Rarely did these maneuvers cause serious problems in transports with un-swept wings.

2.12.3 Solution

In Ref. 2.10 the NTSB observed that this was the fifth such accident during a four year span (1965-1969). The NTSB recommended to the FAA that the following action be taken:

1. re-emphasize to pilots the characteristics of swept wing transports during critical OEI maneuvers
2. assure that flight instructors, trainees, and line pilots are well aware of safe and proper critical OEI procedures, the limits of sideslip angles, rudder availability and sideslip limits for vertical tail stalls

3. caution instructor personnel to emphasize that corrective actions should not be delayed.

The NTSB referred to a document written by Captain A.P. Wilson, a Convair Production test pilot. In this document Wilson indicates that in controlling critical OEI no more than 5 degrees of sideslip should be allowed. He advocates banking 5 degrees into the operating engine side. He also warns that if sideslip is allowed to build up to 16 degrees the vertical tail may stall. In addition, he warns that at around 15 degrees of sideslip the rudder will begin to "blow back".

In the modern FAR rules a specified amount of time must be allowed to pass before corrective action may be taken by the pilot.

It is noted that neither the NTSB nor the FAA, as a result of these five accidents advocated a systems solution. It is the author's view that they should have. Several modern transports employ an automatic system which feeds in rudder and aileron deflection as a result of a detected power asymmetry. In many of these transports a critical OEI situation is now a "non-event".

2.12.4 Lessons

1. It does not seem to make much sense to conduct critical OEI training maneuvers at low altitude in flight. Training these maneuvers should be done at high altitude when in flight and in simulators for the low altitude cases.

2. Designers of all multi-engine transports should keep in mind that pilot workload becomes very high in emergency situations. This should not be the case. A good design philosophy would be that all first emergencies which can reasonably be expected to occur should not result in a controllability problem.

2.13 Rudder Control System too Complicated

2.13.1 Problem

In July of 1969 a Trans World Airlines Boeing 707-331C (Figure 2.15) crashed during a simulated three-engine, missed approach. Three line captains, an instructor pilot and a flight engineer were fatally injured.

Figure 2.15 Boeing 707-331C (Not accident aircraft, Courtesy geminijets.com)

2.13.2 Cause

According to Ref. 2.11 a fatigue failure of the left outboard spoiler hydraulic actuator down-line occurred when the aircraft was on the final segment of an ILS simulated instrument approach with the No.4 engine at flight idle.

Failure of the hydraulic down-line caused the loss of the hydraulic fluid in the aircraft utility system. The captain, instructor pilot and flight engineer were not aware of the hydraulic fluid loss until it was called to their attention by the third pilot in the cockpit.
All hydraulic pumps were then turned off in accordance with the existing emergency procedures for hydraulic fluid loss. There was no failure of the auxiliary hydraulic systems providing power to the rudder.

Power was not restored on the No.4 engine. The aircraft was not accelerated to the 180 knot, three-engine minimum control airspeed upon discovery of the hydraulic fluid loss. A high degree of sideslip angle resulted in loss of lateral control (remember, swept-aft wing aircraft have a large negative rolling moment due to sideslip), causing a rapid roll and loss of altitude.

According to Ref. 2.11 “the probable cause was a loss of directional control, which resulted from the intentional shutdown of the pumps supplying hydraulic pressure to the rudder without a concurrent restoration of power on the No.4 engine. A contributing factor was the inadequacy of the hydraulic fluid loss emergency procedure when applied against the operating configuration of the airplane.”

It is of interest to review some fundamental aspects of the design of the rudder control system of this airplane. The following is also taken from Ref. 2.11.

“Directional control of the aircraft is provided by the rudder, rudder control tab, and rudder control system. Rudder positioning may be accomplished hydraulically through the rudder control hydraulic power unit or mechanically through the rudder control tab and balance panels. Rudder trim is available to provide a means of maintaining directional heading without applying continuous pressure on the rudder pedals. The rudder trim system is a cable-operated linkage that functions through a power trim gearbox during rudder operation in the power mode, **or** through a manual trim gearbox during rudder operation in the manual mode.

Under normal operating conditions, the rudder is hydraulically positioned through full travel (up to 25º of rudder deflection angle). With hydraulic power available, the rudder pedal motion is transmitted by the control linkage to the power control unit actuating valve. An artificial feel unit is incorporated in the powered rudder configuration to provide the pilot with a sensing of the amount of rudder pressure that is being applied. In the power mode the rudder control tab is made to operate in anti-balance direction through the incorporation of a tab linkage in the hydraulic power unit. The tab is thus deflected in the direction of rudder motion on a ratio of 0.8º for each 1.0º of rudder travel. Rudder pedal motion, with rudder power available, is 2 inches fore and aft of neutral with zero trim. In the power mode, the three-engine minimum control speed for TWA 5787 was approximately 117 knots.

In the mechanical mode, the rudder pedal motion (4 inches fore and aft with zero rudder trim) is transmitted by cables and pushrods directly to the rudder control tab. The tab is moved in balance direction to position the rudder aerodynamically through deflection angles to approximately 17º. In the mechanical mode, the minimum three-engine control speed for TWA 5787 was 177 knots (180 knots according to the flight handbook).

With the rudder in the streamlined position, reversion to the mechanical mode is accomplished automatically upon turning off the auxiliary system pumps, or in the event of any failure of the auxiliary hydraulic systems. However, if the rudder has been deflected hydraulically to maintain a directional heading with an asymmetrical power configuration, reversion to the mechanical system will not occur automatically. The tab lock will not release unless the rudder pedals are operated to streamline the rudder or the power switch is turned to "off", thus relieving the internal pressure. In the absence of either action the tab lock will not release until the internal leak rates in the power unit result in aerodynamic streamlining of the rudder, and a degrading hydraulic pressure."

When reading this description of the rudder control system one is left to wonder whether pilots can be expected to know all this in an emergency situation and act accordingly.

2.13.3 Solution

The NTSB recommended a series of immediate and follow-up corrective actions, none of which (according to the author) addressed the fundamental issue of a rudder control system which is simply too complex.

2.13.4 Lesson

The entire scenario could have been predicted if a systematic fault tree analysis had been conducted. To be fair, such a systematic analysis was not required at the time of certification of this airplane.

2.14 Minimum Unstick Speed II

2.14.1 Problem

In September of 1972 a Spectrum Air Sabre Mark 5 (Figure 2.16) crashed on take-off after failing to safely lift-off the runway twice. Twenty persons on the ground were killed. The pilot and twenty-eight other persons on the ground were injured.

Figure 2.16 North American F-86 Sabre (Not accident aircraft, Courtesy NASA)

2.14.2 Cause

According to Ref. 2.12 the probable cause was: "over-rotation of the aircraft and subsequent deterioration of the acceleration capability. The over-rotation was the result of inadequate pilot proficiency in the aircraft and misleading visual cues."

The pilot reported that he felt and heard a vibration shortly after initial lift-off. Apparently he was not sufficiently concerned to reject the take-off. The pilot stated that when he lowered the nose, acceleration seemed normal again and he continued the take-off. The NTSB believes that the vibration experienced was precipitated by disturbed airflow, because of excessive nose-high attitude during the take-off roll. The excessive nose-high attitude was documented by the analysis of 8-mm movies taken by a ground spectator. The attitude was more than three times higher than that of a Sabre Mark 5 test aircraft flown by a designated test pilot off the same runway.

According to Ref. 2.12 the over-rotation was undoubtedly a function of a lack of familiarity of the pilot with the aircraft. It also turns out that the Sabre 5 fighter was very sensitive to over-rotation as a result of its very high elevator effectiveness.

2.14.3 Solution

The NTSB suggested a series of recommendations establishing the need for more training and experience before private individuals should be allowed to fly ex-military aircraft.

2.14.4 Lesson

1. It should be a matter of common sense that a pilot should not fly an airplane at an airshow unless he or she has received appropriate training and checkout on such an airplane.

2. Military jet airplanes, when operated in a civilian environment, should meet the same minimum un-stick requirement which came into being after the accident discussed in Section 2.4.

2.15 Electrical System Failed during Take-off Emergency

2.15.1 Problem

In June of 1973 an Overseas National Airways Douglas DC-8-63 (Figure 2.17) experienced an aborted take-off (at 105-110 kts) as a result of a series of tire failures on the right main landing gear.

Figure 2.17 Douglas DC-8-63 (Not accident aircraft, Courtesy www.al-airliners.be)

The airplane was brought to a stop and a fire ensued in the area of the right main landing gear. Emergency evacuation did not begin until 3 minutes after the aircraft stopped. Because of confusion during the evacuation 34 passengers were injured, 3 seriously.

2.15.2 Cause

According to Ref. 2.13 the cause of the mishap was the undetected deflation of a right main landing gear tire as the aircraft was taxiing for take-off. This increased the loads on the other tires which then overheated and failed.

The cause of the confusion during the emergency evacuation was due to the fact that when the crew shut down the numbers 1 and 2 engines all electrical power was cut off and the public address system became inoperative. The senior cabin attendant used hand signals and shouted directions which were not understood by many passengers. Two available, portable megaphones were not used.

2.15.3 Solution

The NTSB recommended improvements in the cabin attendant training procedures.

The board also suggested that the FAA require that public address systems in passenger aircraft be equipped with an independent power source.

2.15.4 Lesson

The public address system in passenger airplanes should not fail with the shutdown of any engines(s). The electrical system should be designed to prevent this.

2.16 Aft Center of Gravity and Stabilizer Mistrim during Take-off

2.16.1 Problem

In January of 1983 a United Airlines DC-8-54F (Figure 2.18) went out of control after take-off and crashed. The three cockpit crew members were killed.

Figure 2.18 Douglas DC-8-54F (Not accident aircraft, Courtesy Moe Bertrand)

Witnesses noted that the aircraft take-off roll was normal. After lift-off the pitch attitude steepened abnormally. The airplane then rolled to the right and crashed.

2.16.2 Cause

According to Ref. 2.14 the probable cause was that the flight crew failed to follow procedural checklist requirements and to detect and correct a mistrimmed stabilizer. The stabilizer setting used was 7.5 units ANU (Airplane Nose Up) which was the setting used during the landing on the previous flight. After take-off the crew did not have enough nose-down elevator control to prevent the airplane from stalling after which it rolled into the ground. The airplane also was loaded with a more aft center of gravity than was indicated in the dispatch papers although this probably did not contribute to the accident.
Contributing to the accident was the captain allowing the second officer, who was not qualified to act as a pilot, to occupy the seat of the first officer and conduct the take-off.

The flight data recorder did not function and information that would have been useful in the investigation was not recorded.

2.16.3 Solution

Perhaps crews should be warned before taxiing that required check-lists be completed.

2.16.4 Lessons

1. This type of accident occurs again and again. Perhaps a system which does not allow a crew to take-off without performing the required take-off check list should be considered. What is really critical in many airplanes is that the stabilizer trim position be properly set.

2. Commercial aircraft (passenger and cargo) should not take off without a functioning flight data recorder.

2.17 Reverse Propeller Mode in Flight

2.17.1 Problem

In May of 1987 an American Eagle CASA C-212 (operated by Executive Air Charter) crashed while on final approach. Figure 2.19 shows an example of this airplane.

Figure 2.19 CASA C-212 (Not accident aircraft or airline, Courtesy Gerard Helmer)

The crew was killed but four passengers escaped with minor injuries.

2.17.2 Cause

In Ref. 2.15 the NTSB determined as the probable cause: “improper maintenance in setting propeller flight idle blade angle and engine fuel flow resulting in a loss of control from an asymmetric power condition. A contributing factor to the accident was the pilot's unstabilized visual approach."

As part of the investigation it was uncovered that this airplane did not in fact meet several airworthiness standards relating to flying qualities, flap blow-back and fire blocking materials on seats. Somehow these facts had escaped attention during the certification process carried out through the FAA European Region Office in Brussels, Belgium.

2.17.3 Solution

The NTSB made a number of recommendations to the FAA concerning fire blocking materials on seats, surveillance of propeller overhaul facilities, turbo-propeller flight idle blade angle maintenance, design of propeller pitch controls, flight crew training and on the bilateral aircraft certification program.

2.17.4 Lessons

1. It is important that airplanes have benign flying qualities, particularly in their take-off and landing configurations. This airplane was certified under the bilateral aircraft certification program. Although several FAA pilots actually flew the airplane during the original certification program, no thorough flight tests were conducted.

2. Years after the fact (in 1984) several FAA designated pilots in flying this airplane discovered a number of deficiencies in areas such as: stall warning and flap blowback due to the poor design of the flap actuation system. This reflects serious shortcomings in the certification process.

2.18 Center of Gravity too Far Aft and Seat Design

2.18.1 Problem

In November of 1987 a Ryan Air Service Beech Model 1900C (Figure 2.20) crashed short of the runway. There was no fire. Both flight crew members and 16 passengers were fatally injured, 3 passengers were seriously injured.

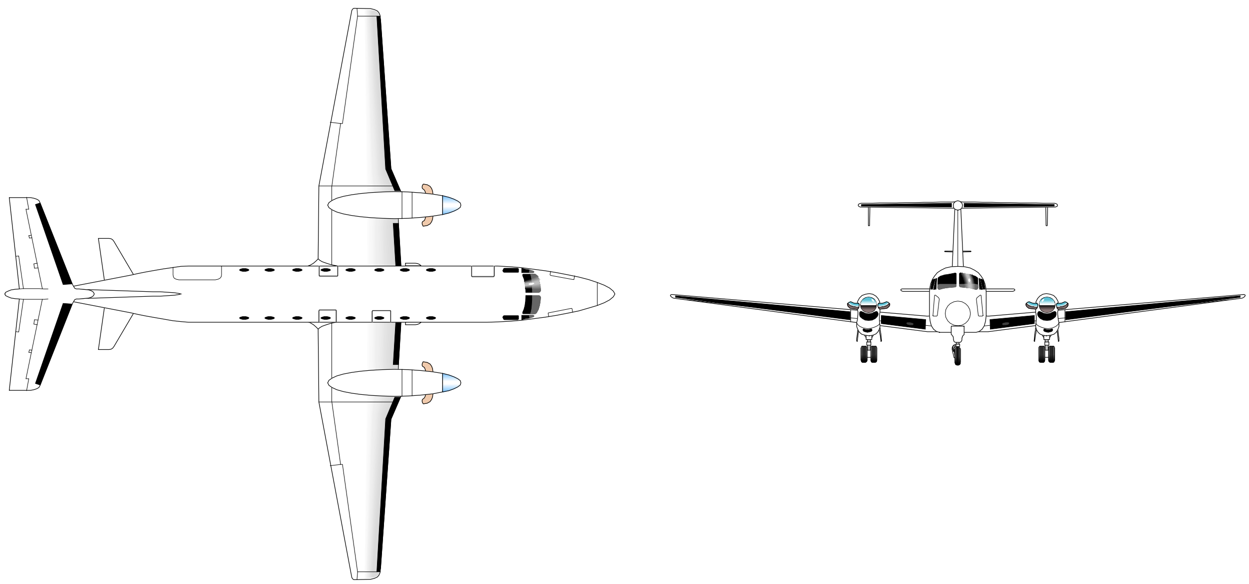


Figure 2.20 Beechcraft 1900C

2.18.2 Cause

Ref. 2.16 states that the probable cause was "the failure of the flight crew to properly supervise the loading of the airplane which resulted in the center of gravity being displaced to such an aft location that airplane control was lost when the flaps were lowered for landing." Post-accident calculations based on the load manifest showed that the center of gravity was 8-11 inches aft of the allowable limit.

One NTSB Board member also indicated that: "Contributing to the severity of the occupant's injuries was the inability of the aircraft's seats to withstand the crash forces; had these seats been designed to the standards which the Board has advocated for over twenty years, the severity of the occupant's injuries may have been reduced and more passengers could have survived."

Contributing factors to the many deaths and injuries were:

1. The fact that the rescue crew could not locate the aircraft master switch to shut off electrical power, and
2. The fact that the rescue crew did not know where to cut into the fuselage to access passengers.

2.18.3 Solution

The NTSB noted in its conclusions that the crew failed to comply with company and FAA procedures in determining the c.g. The Board also recommended (again!) rule-making to increase seat design standards and to publish and disseminate information regarding airplane access points as well as the location of master switches and batteries.

2.18.4 Lessons

1. Airplanes continue to be loaded beyond the aft allowable c.g. One human failure to verify the c.g. location before take-off can, and does often cause a fatal result. This is not in keeping with the thought that one single failure should not result in a catastrophe. It should be presumed that humans will fail in checking the c.g.

2. The airplane seats were designed in accordance with the standards which applied at that time. However, the NTSB had been advocating a revision of these seat design standards for a long period of time.

3. Design engineers should keep in mind that just because the FAA may be slow in revising these standards does not eliminate the ethical obligation of design engineers and their employers to make voluntary adjustments when events indicate the need to do so.

2.19 Icing on Take-off I

2.19.1 Problem

In November of 1987 a Continental Airlines DC-9-14 (Figure 2.21) crashed right after take-off from Stapleton International Airport near Denver.

Figure 2.21 Douglas DC-9-14 (Not accident aircraft, Courtesy geminijets.com)

Both pilots, a flight attendant and 25 passengers were fatally injured. Two flight attendants and 52 passengers survived.

2.19.2 Cause

In Ref. 2.17 the NTSB determined that the probable cause of the accident was: "the failure of the captain to have the airplane de-iced a second time after a delay before take-off that led to upper wing surface contamination and a loss of control during rapid take-off rotation by the first officer."

Several surviving passengers on board this airplane reported seeing some ice on engine inlets or in patches on the wing after de-icing (using a 38% glycol solution).

Had the airplane been anti-iced with 100% glycol, following the de-icing procedure, ice accumulation would have been delayed about 2.8 times longer thereby affording a safe take-off. During the public hearing on this accident a representative from McDonnell-Douglas stated that small amounts of upper wing ice may severely degrade the lifting capability of the wing and lead to loss of roll and pitch control on DC-9-10 series airplanes.

These airplanes do not have leading edge slats which tend to provide some protection in case of minor ice contamination of the upper wing surface.

It is estimated that granular ice of only 0.030 inch (equivalent to 30-40 grit sandpaper) would degrade the maximum lift capability of the DC-9-10 wing by about 20%. This would translate into an increase of stall speed of about 11% therefore providing very little margin when the airplane begins rotation.

In this accident the problem was aggravated by the aggressive rotation applied by the first officer who was flying the airplane. The flight data recorder indicated a rotation rate of 6 degrees per second which is twice what is recommended. With a normal rate of rotation the angle of attack reached at lift-off would have been about 9 degrees. In the accident airplane this was about 12 degrees which is only 2 degrees below the stall angle of attack of the uncontaminated DC9-10 wing.

The board also found that the flight experience of both pilots was marginal on this type of airplane.

2.19.3 Solution

In Ref. 2.17 the NTSB made a number of recommendations. The most important of these were:

1. Until such time that guidelines for detecting upper wing surface icing can be incorporated into the flight manual, all operators of this type of airplane should be directed by the FAA to require that these airplanes be anti-iced with a maximum effective strength glycol solution when icing conditions exist.
2. Require that all DC-9-10 operators establish detailed procedures for detecting upper wing ice before take-off

3. Establish minimum experience levels for each pilot-in-command and second-in-command pilot, and require the use of such criteria to prohibit the pairing on the same flight of pilots who have less than the minimum experience in their respective positions.

2.19.4 Lesson

Icing continues to be a problem in all aircraft operations. Designers should use airfoils which are less sensitive and/or consider the use of slats as standard on all jet airplanes certified for flight into known icing conditions.

2.20 Take-off without Flaps

2.20.1 Problem

In 1988 a Delta Airlines Boeing 727-232 (Figure 2.22) took-off without flaps from the Dallas/Fort-Worth airport in Texas.

Figure 2.22 Model of Boeing 727-232 (Courtesy geminijets.com)

The airplane stalled immediately after lift-off and then crashed. Twelve passengers and two crew members were killed. Twenty-one passengers and five crew members were seriously injured and 68 passengers sustained minor or no injuries.

2.20.2 Cause

According to Ref. 2.18 the NTSB determined the probable cause to be:

1. Inadequate cockpit discipline which resulted in the wing flaps and slats not having been properly configured for take-off, and
2. The failure of the take-off configuration warning system to alert the crew that the airplane was not properly configured for take-off.

The NTSB found that the take-off warning system had an intermittent failure problem which was not corrected during the last maintenance activity.

2.20.3 Solution

The NTSB recommended that a directed engineering study be performed on the take-off warning system on B-727 airplanes. Another recommendation was that the FAA issue an airworthiness directive to require modification of the take-off warning systems.
According to the NTSB the FAA was well aware of the fact that Delta Air Lines flight crews had exhibited deficiencies that needed to be corrected.

2.20.4 Lesson

If a take-off warning system exhibits failure problems this should be corrected before revenue flights are conducted. This particular airplane had intermittent failure problems with its take-off warning system which were not corrected before the flight.

A take-off warning system should be considered as a no-go item when not functional.

2.21 Sterile Cockpit and Rudder Trim Switch Location

2.21.1 Problem

In September of 1989 a US Airways Boeing 737-400 (Figure 2.23) experienced a rejected take-off from LaGuardia airport (NY) and came to rest in the water in Bowery Bay.

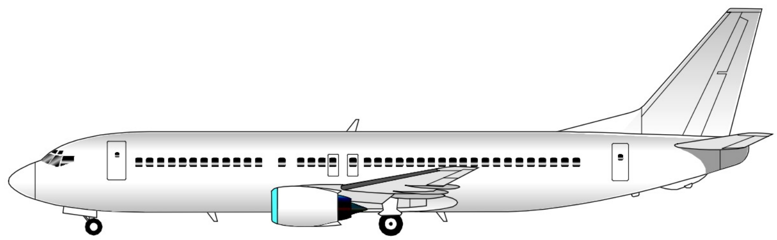


Figure 2.23 Boeing 737-400

Two passengers were killed and three received serious injuries. Six crew members and twelve passengers received minor injuries. Thirty-seven passengers escaped without injuries.

2.21.2 Cause

Ref. 2.19 establishes as the probable cause: "the captain's failure to exercise his command authority in a timely manner to reject the take-off or take sufficient control to continue the take-off, which was initiated with a mis-trimmed rudder. Also causal was the captain's failure to detect the mis-trimmed rudder before take-off was attempted."

During the time leading to push-away from the gate, several persons entered the cockpit and papers were placed on the center pedestal. It is not clear whether any person placed his or her foot on the center pedestal. The reason this is important can be seen from Figure 2.24 where it is shown that the rudder trim knob is located at the end of the pedestal where it can be easily engaged unintentionally. This rudder trim knob is spring loaded to return to its center position. The rudder trim actuator runs only when the knob is displaced from its center position.

In the accident airplane the rudder trim moved full left before push-away from the gate and this was not detected by the flight crew. Full rudder trim causes the rudder pedal to be offset by 4.25 inches and that was not noticed as unusual.

During the take-off run the crew experienced directional control problems which eventually led to the decision to abort. The board did establish the fact that with appropriate control actions the crew could have maintained control and could have stopped the airplane on the runway.

2.21.3 Solution

In Ref. 2.19 the NTSB made several recommendations regarding improvements in cockpit discipline and use of checklists.

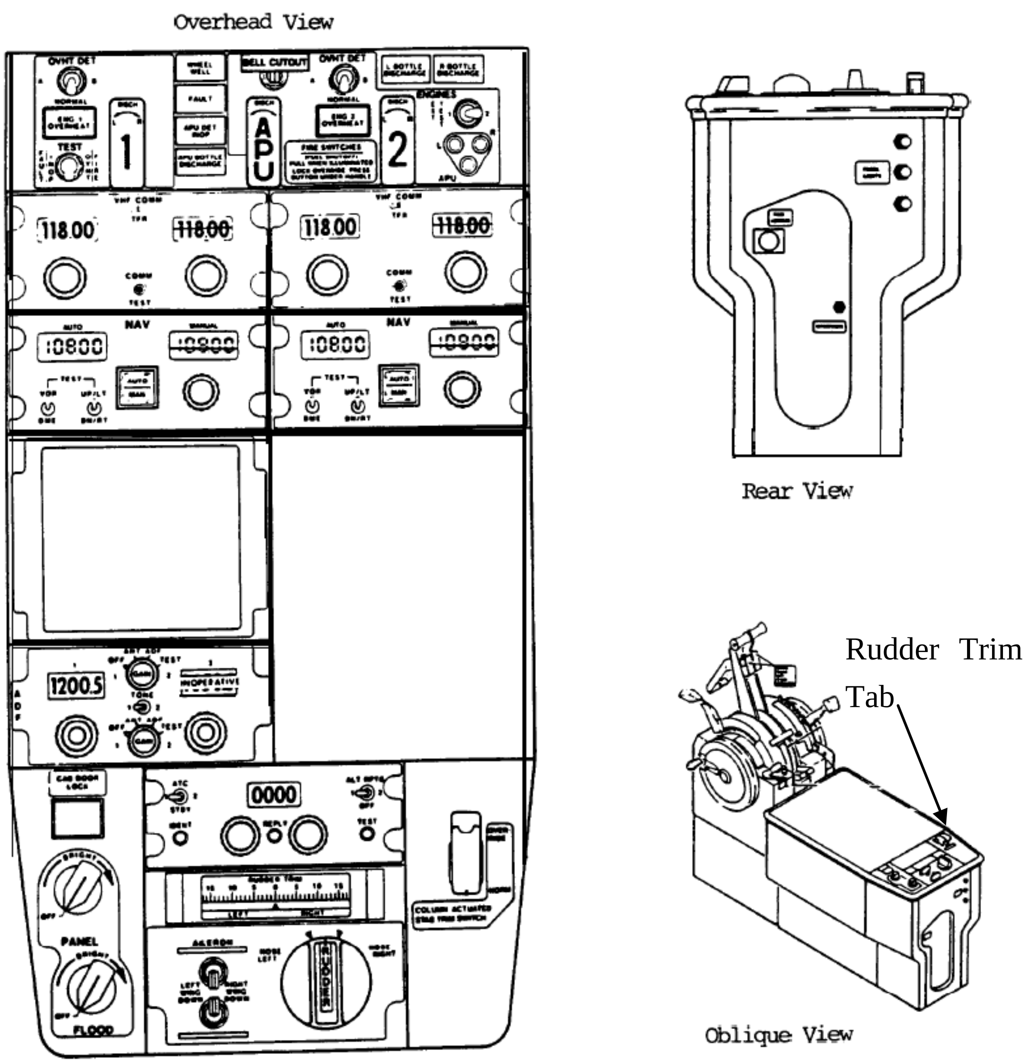


Figure 2.24 Boeing 737-400 Center Pedestal

During the investigation of this crash it became clear that several other instances of inadvertent rudder trim knob actuations had occurred on other B-737 airplanes. The author believes that the NTSB should have also recommended that these controls be placed in a less vulnerable spot.

2.21.4 Lesson

The several precursor incidents of inadvertent rudder trim actuation should have made design engineers wonder about the decision to locate the rudder trim knob in such a vulnerable spot. And that is in fact what happened. Boeing, in May of 1989 initiated studies to curtail inadvertent disturbance of rudder trim controls.

The author believes that these critical controls should not have been placed in that location in the first place: too vulnerable to unpredictable movement of limbs.

2.22 Icing on Take-off II

2.22.1 Problem

In February of 1991 a Ryan International Airlines DC-9-15 (Figure 2.25) crashed on take-off from Cleveland, Ohio. There were no passengers on board. Both crew members were killed.

Figure 2.25 Douglas DC-9-15 (Not accident aircraft or airline, Courtesy G. Helmer)

2.22.2 Cause

In Ref. 2.20 the probable cause was determined to be the failure of the flight crew to detect and remove ice contamination from the wings prior to take-off. The NTSB judged this to be largely due to a lack of appropriate response by the FAA, Douglas and Ryan to the known critical effect that a minute amount of contamination has on the stall behavior of the DC-9 series 10 airplanes (See Section 2.18). The ice contamination led to wing stall and loss of control during the attempted take-off.

2.22.3 Solution

The NTSB issued an extensive series of recommendations and procedures with emphasis on procedures to be used during winter operations, to prevent this from happening again: see Ref. 2.20, pp 52-53.

2.22.4 Lessons

There have been (and are) other airplanes with severe sensitivity to icing contamination. The real lessons here are twofold:

1. Design engineers should run tests or CFD simulations to determine the sensitivity of an airfoil to those icing situations which can occur based on previous operational experience. A popular solution to mitigate the effect of certain types of ice on airfoil aerodynamic behavior is to use leading edge slats to protect the wing from stalling at a low angle of attack.

2. Crews have to be brain-washed into believing that icing can be deadly and therefore never attempt a take-off without verifying that wings and tails are free from ice and snow.

2.23 High Speed Descent to Avoid Icing

2.23.1 Problem

In 1994 a BAE Jetstream of Atlantic Coast Airlines (Figure 2.26) crashed on final approach to Columbus, Ohio. The pilot, co-pilot, flight attendant and two passengers were fatally injured. Two passengers received minor injuries and one was not injured.

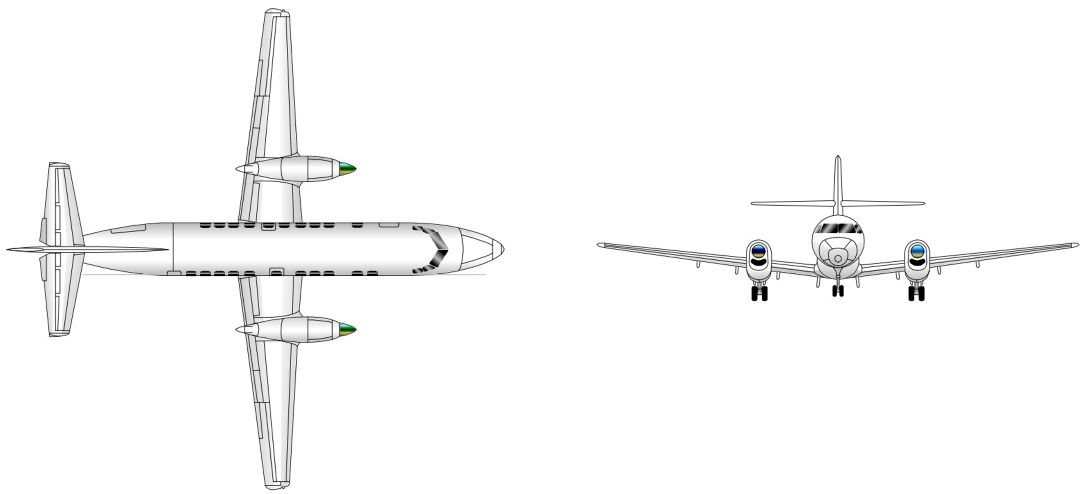


Figure 2.26 BAE Jetstream 4101

The pilot was conducting a high speed approach to avoid icing conditions. This left little time to configure the airplane for a stabilized final approach which was flown on autopilot. To slow the airplane down power was pulled back to flight idle. The autopilot trimmed the airplane nose up which, together with the low thrust caused the airplane speed to decay below the minimum approach speed of 130 knots. When the airplane slowed down to 104 knots the stick shaker activated and the autopilot disconnected. The flight data recorder indicated that the captain applied nose-up elevator without adding power. This action caused the descent rate to increase to 2,400 ft/min which continued until impact. The situation was aggravated by the fact that the first officer raised the flaps at the captain’s command without challenging this.

2.23.2 Cause

In Ref. 2.21 the NTSB determined the probable causes to be:

1. An aerodynamic stall that occurred when the flight crew allowed the airspeed to decay to stall speed following a very poorly planned and executed approach characterized by an absence of procedural discipline;
2. Improper pilot response to the stall warning, including failure to advance the power levers to maximum, and inappropriately raising the flaps;
3. Flight crew inexperience in "glass cockpit" automated aircraft, aircraft type, and in seat position, a situation exacerbated by a side letter of agreement between the company and its pilots; and
4. The company's failure to provide adequate stabilized approach criteria, and the FAA's failure to require such criteria.

2.23.3 Solution

The board issued a series of recommendations regarding flight training of air crews and the pairing of inexperienced flight crew members. Both the captain and first officer had limited experience in this airplane.

2.23.4 Lesson

Design engineers can do little to prevent flight crews from really violating proper cockpit procedures.

2.24 Center of Gravity too Far Aft II

2.24.1 Problem

In January of 1999 a Channel Express Fokker F-27 cargo transport (Figure 2.27) crashed when control was lost on final approach. Both pilots were killed.

When the flaps were lowered during the final stage of the approach the nose of the aircraft rose and attempts by the crew to lower the nose were not successful. The airplane stalled, entered a spin and crashed into a house. The sole occupant of the house escaped injury.

Figure 2.27 Fokker F-27-600 (Not accident aircraft, Courtesy D. Schulman)

2.24.2 Cause

According to Ref. 2.22 the causes of this accident were:

1. The aircraft was operated outside the load and balance limitations
2. Loading distribution errors went undetected because the load sheet signatories did not reconcile the cargo distribution in the aircraft with the load and balance sheet
3. The flight crew received insufficient formal training in load management

2.24.3 Solution

The British Department of Transport made seven safety recommendations dealing with training and proper load and balance management.

2.24.4 Lesson

This type of accident happens again and again. One more time: manufacturers should provide a simple system with any transport airplane which alerts the crew when either the weight or the center of gravity location is outside the limits.

2.25 Center of Gravity too Far Aft, Overloading and Misrigged Controls

2.25.1 Problem

In 2003, an Air Midwest Beech Model 1900D (Figure 2.28) crashed shortly after take-off from Charlotte, NC. Both flight crew members, 19 passengers and one person on the ground were killed.

Figure 2.28 Beech Model 1900D (Not accident aircraft, Courtesy T. Perkins)

2.25.2 Cause

According to Ref. 2.23 the probable causes of this accident were:

1. Loss of pitch control on take-off
2. Incorrect rigging of the elevator flight control system
3. The center of gravity was substantially aft of the allowable limit.

2.25.3 Solution

The NTSB noted a number of serious problems at Air Midwest, Raytheon (Beech) and the FAA:

- Lack of oversight in one of the Air Midwest maintenance stations
- Lack of proper maintenance procedures and documentation
- Questionable weight and balance program at Air Midwest
- Failure of the Raytheon Aerospace quality assurance inspector to detect the incorrect rigging of the elevator control system
- The FAA average weight assumptions (these were revised upward after the crash)

- The FAA's lack of oversight of Air Midwest's maintenance program and its weight and balance program.

2.25.4 Lessons

1. From a design engineering viewpoint it is incomprehensible that flight crucial control systems require extensive rigging procedures. These should be designed so that this is not necessary.

2. It is also incomprehensible why, after maintenance of a flight crucial aspect of a flight control system, no functional checks were performed by the maintenance organization or by the flight crew. According to the NTSB these checks were not even required (sic!).

2.26 Center of Gravity too Far Aft and Overloading

2.26.1 Problem

In December of 2003 a Boeing 727-200 of UTAG (Figure 2.29) crashed during a take-off attempt. Of the 160 persons on board, 140 were killed.

Figure 2.29 Model of Boeing 727-200 (Courtesy geminijets.com)

2.26.2 Cause

According to Ref. 2.24 the airplane was 22,000 lbs over its allowable take-off weight although this did not cause the crash. The center of gravity was "well aft of the allowable limit" according to the preliminary French investigation report.

2.26.3 Solution

Another repeat of a familiar problem. The formal accident report is not yet available. However, proper weight and balance procedures and requirements to obey these seem to be fundamental.

2.26.4 Lesson

This type of accident happens again and again. One more time: manufacturers should provide a simple system with any transport airplane which alerts the crew when either the weight or the center of gravity location is outside the limits.

2.27 Old Habits Return in Emergencies I

2.27.1 Problem

The Cessna Model 336 Super Skymaster (Figure 2.30) is a centerline thrust twin designed to avoid one engine inoperative (OEI) control problems. With both propellers on the centerline, losing one engine does not result in a yawing moment and avoids losing control at too low an airspeed. Aeronautical engineers and multi-engine pilots refer to this as the minimum control speed problem (for a mathematical discussion of this problem, see Ref. 2.25). The airplane also has retractable landing gears. The main gears retract rearward into a cavity below the rear engine. To make this possible, required rather large landing gear doors (Figure 2.30). That in turn would result in high drag with the gear down and therefore it was decided to retract most of the doors after the gear extends. That is a design trick also used on many jet transports.

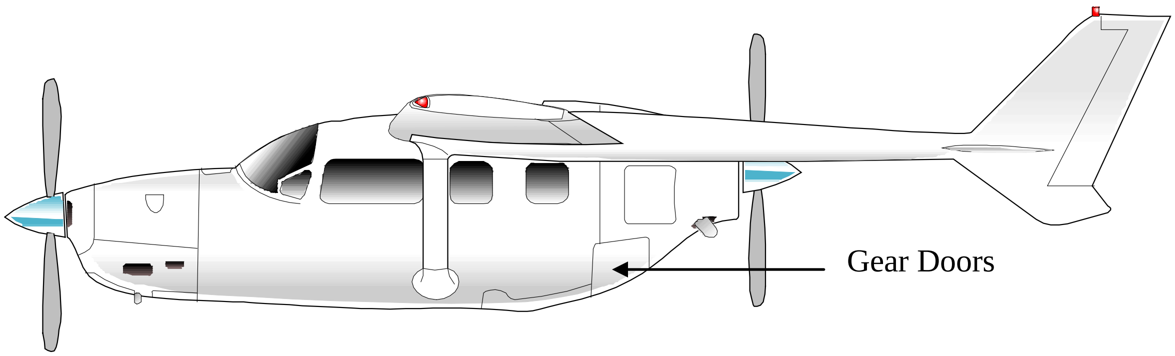


Figure 2.30 Cessna Model 336 Super Skymaster (Note Large Gear Well Doors in Closed Position

Pilots who transition from conventional twins to the Model 336 have to learn that in case of OEI right after lift-off the landing gear should be left in the down position to reduce drag on climb-out. However, the typical OEI drill in conventional twins is to retract the landing gear following an engine failure.

Several serious accidents occurred when, following engine failure after lift-off the pilot reverted to old habits and retracted the landing gear. In the Model 336 what happens first is the large landing gear doors come down. In that position, the drag is much higher which compromises the climb capability of the airplane with OEI. The lawsuits against Cessna following several of these accidents caused the airplane to be taken out of production.

2.27.2 Lesson

Sometimes seemingly good ideas can have unforeseen negative consequences. Basically the Cessna 336 was a safer airplane to fly than conventional twins. But, because some pilots revert to old habits in the case of an emergency and because of the cost of tort liability the airplane had a relatively short production life.

2.28 Old Habits Return in Emergencies II

2.28.1 Problem

Manufacturers, for a variety of reasons, decide to change the layout of the cockpit of an airplane. Such a decision often turns out to be a real trap for pilots experienced in that model. A case in point is the Beechcraft A-36 (Figure 2.31).

Figure 2.31 Beechcraft A-36 Bonanza (Not accident aircraft, Courtesy Digimicra@airliners.net)

A change which caused a lot of operational problems was the reversing of the flap and gear switches from the way these were in most retractable gear airplanes. This led to pilots raising the gear instead of the flaps while taxiing to their ramp position.

The following is taken from Ref. 2.26, page 138:

"To compound the issue earlier models had just one squat switch and it was located on the right main gear. As weight comes off the gear during take-off the strut extends slightly, closing a circuit and delivering power to the gear motor if the gear switch is selected to retract. While the squat switch prevented the gear from retracting while the airplane was at rest on the ground, the same couldn't be said if the switch was selected to retract while the airplane was taxiing. For example, if the right main gear became unweighted when turning off the runway onto the taxiway, the gear might come up. Beechcraft addressed the issue by placing a squat switch on each main gear on newer model Bonanzas, starting in the mid-1970's."

Another change that caused problems was the reversing of the left-right sequence of throttle, mixture, and propeller levers.

2.28.2 Lesson

When making changes to the cockpit, designers should carefully evaluate the potential consequences. In this regard it would be helpful if the people making these changes are themselves licensed pilots.

2.29 Postlude

Operational experience with airplanes shows a series of recurring events which many times have serious consequences. The most frequently occurring mishaps are:

1. Taking off with the center of gravity too far forward or too far aft
2. Taking off with the weight above the maximum allowable take-off weight
3. Reversing propellers in flight
4. Stalling and/or losing control in icing conditions
5. Taking off without proper flap setting
6. Flying unstabilized approaches
7. Taking off with gust locks engaged
8. Striking the rear fuselage during take-off rotation

In many operational situations the pilot task is made more difficult by problematic instrumentation, diversion of attention due to other problems, poor visibility, moderate or severe turbulence. Another way of putting this is: the pilot workload is too high. The author believes that automating these tasks would help reduce accidents and incidents.

Design engineers and design management should be familiar with the operational history of whatever airplane type they are working on. That might help to reduce the severity and frequency of occurrence of many of these problems.

Chapter 3

Lessons Drawn from Structural Design

"Design structures for inspectability and repairability"

Dr. Jan Roskam, 1990

3.1 Introduction

In this chapter a series of problems which arose due to problems with structural design are reviewed. Where applicable, causes and solutions are described and lessons learned are stated. Connections with other areas of design are identified in Appendix A.

3.2 Fatigue Failure of Wing Fitting I

3.2.1 Problem

In March of 1948 a privately owned Vultee V-1A (Figure 3.1) crashed nine miles north of Somerset, Pennsylvania.

All eight occupants were killed and the aircraft was destroyed.

Figure 3.1 Vultee V-1A (Not accident aircraft, Courtesy Royal Aeronautical Society Library)

3.2.2 Cause

Ref. 3.1 lists as the probable cause: "the failure from fatigue of the steel wrap-around plate of the steel attachment lower fitting at the rear spar of the right wing, causing separation in flight of the right outer wing panel from the center panel."

The outer wing panel was found 210 feet away from the rest of the wreckage. From inspection it was found that the fatigue failure occurred in the steel wrap-around sheet of the bottom steel lug of the attachment fitting of the rear spar. It was clear from this construction that evidence of fatigue in this case could not be disclosed in the course of the usual inspection.

3.2.3 Solution

To avoid fatigue failures, fatigue cracking must be found before it is too late. Detection of fatigue cracks is possible only if the affected components are designed to be easily inspectable.

3.2.4 Lesson

By modern standards a primary structure which is not completely inspectable by normal inspection procedures would not be certifiable. Designers must keep this in mind. All primary structure of an airplane should be readily inspectable.

3.3 Fatigue Failure of Wing Fitting II

3.3.1 Problem

In 1948 a Northwest Airlines Martin 202 (Figure 3.2) lost a wing while flying close to a thunderstorm over the state of Minnesota. All 37 persons on board were killed.

Figure 3.2 Martin 202, Before Dihedral Modification (Not accident aircraft, Courtesy Stan Piet)

Another Northwest Martin 202 that flew through the same storm about one hour after the accident airplane landed with a crack in a wedge shaped forging which was used in the structural connection between the inboard wing and the outboard wing.

3.3.2 Cause

In Ref. 3.2 the probable cause of this accident was: "the loss of the outer panel of the left wing which separated from the aircraft as a result of a fatigue crack in the left front outer panel attachment fitting which had been induced by a faulty design of that fitting, the fatigue crack having been aggravated by severe turbulence encountered in the thunderstorm."

The reason for the existence of this fitting will be explained. Figure 3.3 shows a front view of the Martin 202 with the characteristic ten degrees of geometric dihedral.

Figure 3.3 Martin 202 (Not accident aircraft, Note the Large Dihedral of the Outboard Wing)

During early flight testing of the airplane (Figure 3.2) it was found that there was insufficient dihedral effect (i.e. negative rolling moment due to sideslip). This was corrected by increasing the geometric dihedral angle of the outboard wings. To enable this from a structural viewpoint, wedge shaped, forged fittings were inserted at the spars of the wing. The fittings were designed with fairly sharp edges in areas of high stresses leading to local stress concentrations which in turn led to early cracking. Figure 3.4 shows a perspective view of this fitting and its installation in the wing.

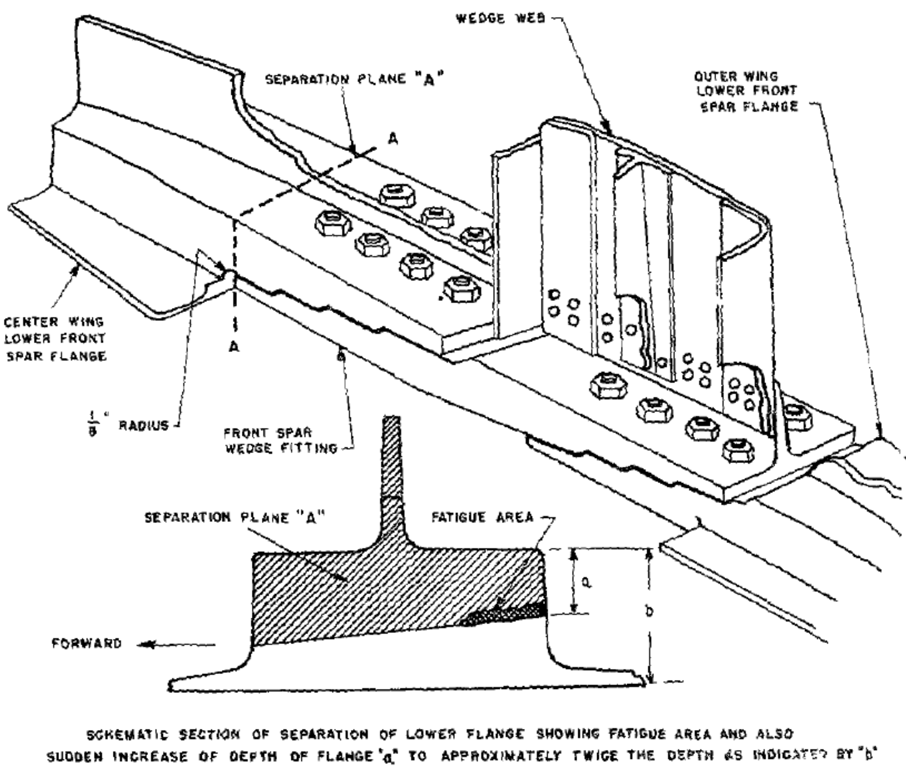


Figure 3.4 Sketch of the Forged Fitting (Courtesy Civil Aeronautics Board)

3.3.3 Solution

The fittings had to be redesigned and replaced on all existing Martin 202 transports.

3.3.4 Lessons

1. Avoid stress concentrations in highly loaded areas of flight crucial structures.

2. Sharp edges and sharp corners in highly loaded structural components should not be allowed. A critical design review should have caught this situation.

3.4 Canopy Loads Must Be Watched

3.4.1 Problem

In 1951, during early test flying of the Fokker S-14 jet trainer (Figure 3.5) it was discovered that the canopy attachment frame deformed significantly in flight such that the resulting gaps caused pressurization to be lost.

Figure 3.5 Fokker S-14 Mach Trainer (Not accident aircraft, Courtesy Joop de Groot)

3.4.2 Cause

The cause of the problem was that the suction pressure loads on the canopy were much larger than expected. In addition, the number of latching pins was too small which allowed the canopy frame to distort under loads.

3.4.3 Solution

An extensive series of pressure survey flight tests were conducted on a steel canopy to determine the actual loads. After that, the canopy attachment structure and latching pin system were completely redesigned.

3.4.4 Lesson

In aircraft with large canopies the pressure loads on these canopies can be very extreme. It is of interest to observe that similar problems were experienced in 1955 by Cessna in the development of the Cessna T-37 (Figure 3.6) and by English Electric in 1956 during test flying of the P1B Lightning (Figure 3.7). In fact, the P1B lost its canopy twice during early flight testing.

Figure 3.6 Cessna T-37 (Courtesy San Diego Aerospace Museum)

Figure 3.7 English Electric P1B Lightning (Courtesy S. Petch)

An interesting question is why this lesson has to be re-learned so many times by different design teams in three different countries. In modern aircraft design the canopy loads can be estimated with the help of CFD. This has the potential of saving a lot of money.

3.5 Verification in Structural Design

3.5.1 Problem

During the summer of 1953 the author worked at Percival Aircraft Ltd. in Luton, England as an engineering apprentice. My first assignment was to verify the structural design, load-paths and stress calculations which had been done on the wing tip extension of the new Percival Pembroke (Figure 3.8).

To do this I was given copies of all appropriate drawings, which had already been released to prototype manufacturing. The Pembroke was a slightly larger version of the Prince, a twin-engine, propeller-driven, high wing, utility transport airplane.

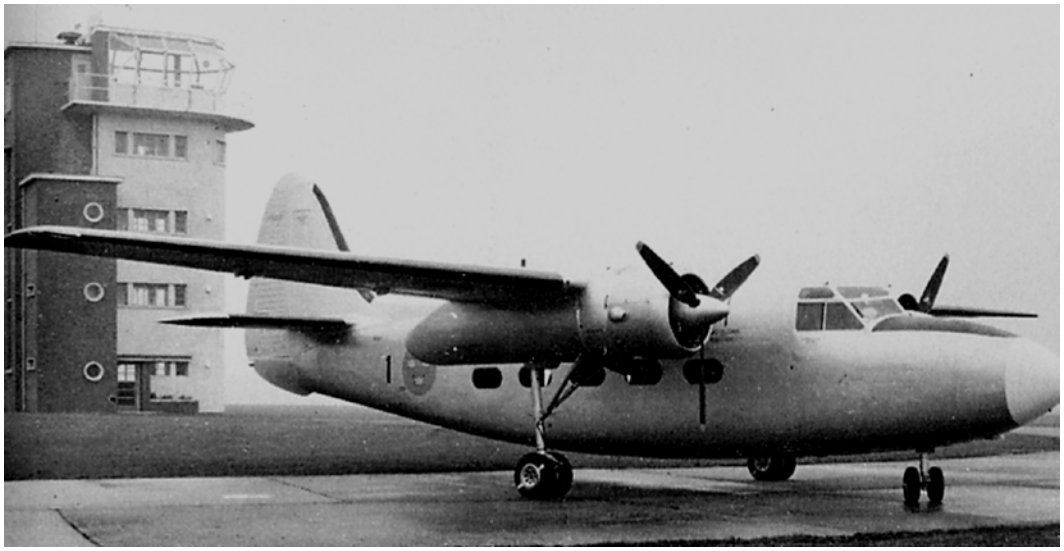


Figure 3.8 Percival Pembroke (Not accident aircraft, From Ref 3.3 Courtesy Mrs. B. Silvester)

I discovered a major flaw in the design which would have caused the wing tip to come off on the very first flight. As it turned out, this flawed design had already been installed on the first prototype.

3.5.2 Solution

I proposed a redesign which was approved and installed.

3.5.3 Lesson

All flight crucial structural design aspects of an airplane should be cross-checked by an engineer other than the original design engineer. This is a very important lesson which also speaks to the organization of structures design and analysis departments.

3.6 Fatigue Failure due to Pressurization Cycles

3.6.1 Problem

In January of 1954 a DeHavilland Comet 1 (Figure 3.9) crashed into the sea near the island of Stromboli, Italy. There were no survivors.

Figure 3.9 DeHavilland Comet 1 (Not accident aircraft, Courtesy M. West)

All Comets were temporarily grounded. After an intensive inquiry into the structural design criteria, no major problems were found, and the airplanes were released for flight in April of 1954. Two weeks later, still in April of 1954 a similar fate befell another Comet, this time near the island of Elba, Italy. The certificate of airworthiness was withdrawn and the entire fleet was grounded.

3.6.2 Cause

The finding of the cause is discussed in Ref. 3.4 (pp 15-21). Even though fatigue had been suspected it was not really considered the culprit, because none of the Comets were even close to the design life of 18,000 flights. Nevertheless, after the first accident the oldest Comet in the fleet was taken to the RAE at Farnborough. There, the fuselage was submerged in a large water-tank, and a systematic series of pressurization-depressurization cycles were initiated. After the equivalent of 9,000 flight hours the test fuselage failed. The tank was drained and it was found that the fuselage had developed a very large split, similar to that found in the wreckage of the first and second Comet. The cause was established: a fatigue crack initiated at a corner of a roof window for the ADF antenna. A hairline fatigue crack had started at a rivet hole in the corner of that window and rapidly progressed to split a large section of the fuselage.

3.6.3 Solution

As a result of these accidents and the ensuing very detailed investigation, changes were made not only in the way fuselages of high altitude jet transports are designed but also in the manner in which they must be tested.

Modern fuselages carry a heavier skin, particularly in the areas of windows and doors. Also, windows are made smaller and well-rounded to avoid stress concentrations. In addition, crack-stopping features are designed into all fuselage structure to prevent the type of massive failure that occurred in the Comet 1 airplanes.

Finally, water-tank fatigue testing is now mandated for all newly designed transport fuselages.

3.6.4 Lesson

Whenever an airplane can be expected to operate in a new type of environment design engineers should ask all appropriate "what-if" questions to determine whether the new design is really adequate.

3.7 Vertical Tail Flutter

3.7.1 Problem

In 1959 a prototype of the Convair 880 (Figure 3.10) experienced a failure of the vertical tail during a dive test. The airplane was safely landed.

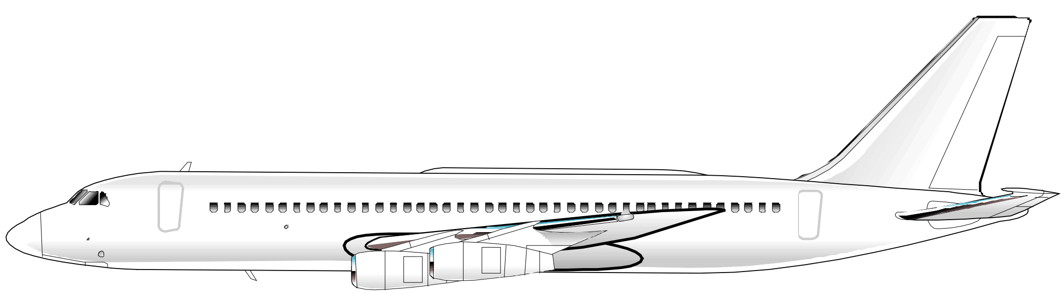


Figure 3.10 Convair 880

3.7.2 Cause

The cause turned out to be flutter. An error in the engineering mathematical model of the vertical tail was identified as the culprit.

3.7.3 Solution and Lesson

Flutter predictions, to be accurate require very careful attention to the mathematical modeling of the torsional and bending stiffness distribution along the span of any lifting surface. The mass distribution relative to the elastic axis of the structure must also be properly modeled.

3.8 Whistling Swan Downs Viscount

3.8.1 Problem

In November of 1962 a United Air Lines Vickers Viscount (Figure 3.11) crashed near Ellicott City, Maryland. All 13 passengers and the crew of 4 were fatally injured.

Figure 3.11 Vickers Viscount Model 745D (Not accident aircraft, Courtesy D. Schulman)

3.8.2 Cause

Ref. 3.5 states as the probable cause a loss of control following separation of the left horizontal stabilizer which had been weakened by a collision with a Whistling Swan.

Examination of the wreckage indicated bird strikes on both horizontal stabilizers. Bird strike damage to the right stabilizer was superficial. However, the left stabilizer clearly had been penetrated by a large bird which was determined to have been a Whistling Swan. These birds easily weigh 16 lbs. Figure 3.12 shows the path of the swan through the structure which was weakened so that it failed down and aft under its normal load.

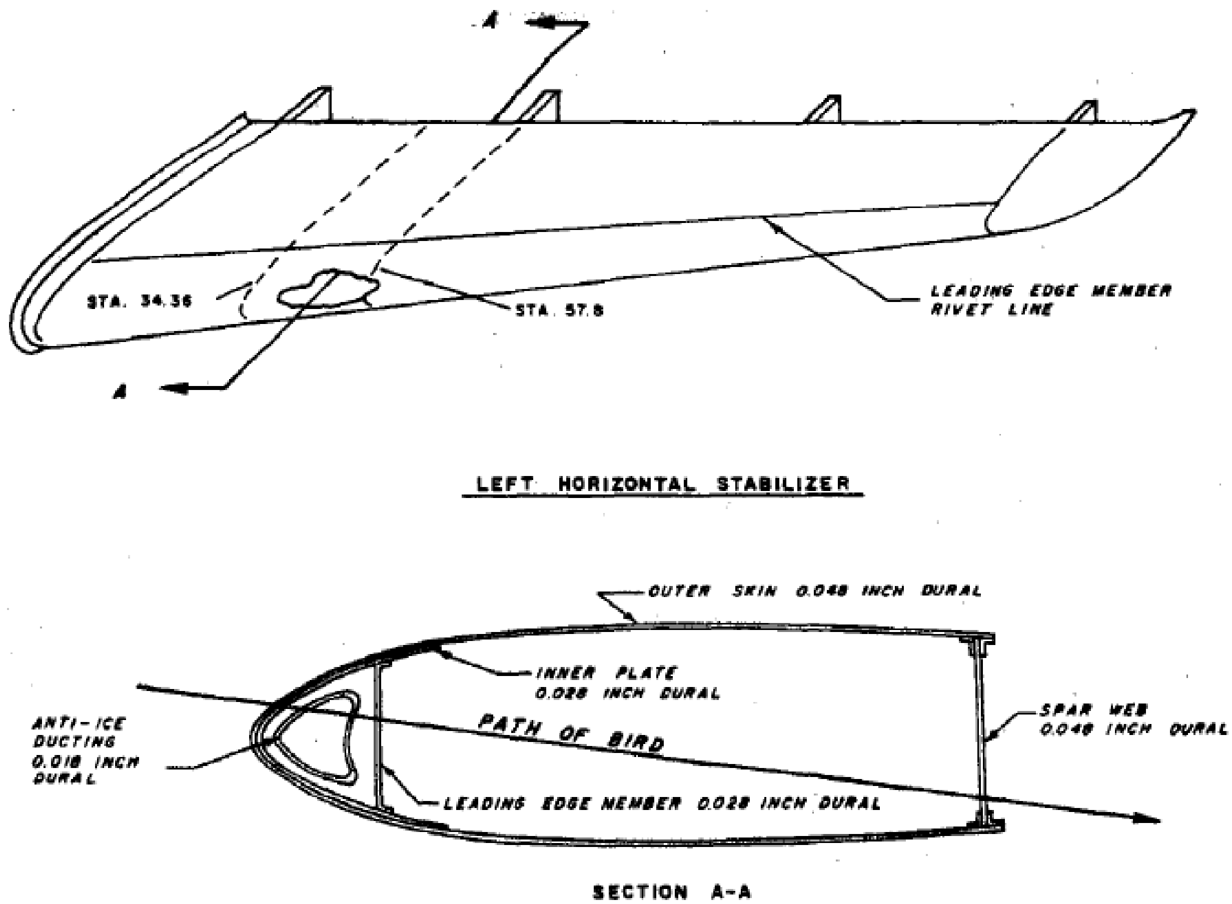


Figure 3.12 Damage Path of Whistling Swan in Collision with the Left Stabilizer (Courtesy Civil Aeronautics Board)

3.8.3 Solution

This accident occurred in 1962. Modern airworthiness regulations assume that a 6 lbs bird strike at 250 kts aimed at the wind-screens of a commercial airplane shall not penetrate the wind-shield. A similar regulation exists with regard to primary structure.

It is still an open question what constitutes a reasonable size bird, striking an airplane at what speed.

3.8.4 Lesson

Bird strikes have and continue to cause many problems in aeronautics. One thing is clear: even relatively small birds would penetrate a leading edge of a wing or tail. As long as the damage done to the structure is local in nature, a safe landing can probably be made.

The following comment is not relevant to this case but it is important for designers to keep in mind. If flight crucial hydraulic lines, electrical conduits or cables are located in such areas, loss of control is a possibility. Therefore, as a general rule, such systems should not be placed in leading edges.

3.9 A New Flutter Mode

3.9.1 Problem

When the F-111A (Figure 3.13) was in flight test, it was discovered that at high dynamic pressures a coupling of thee vibration modes could occur which might lead to flutter.

Figure 3.13 General Dynamics F-111A (Not accident aircraft, Courtesy NASA)

3.9.2 Cause

Because of the aft engine configuration of the F-111A, the rear fuselage, from a structural viewpoint, had a T-cross section. Figure 3.14 illustrates this.

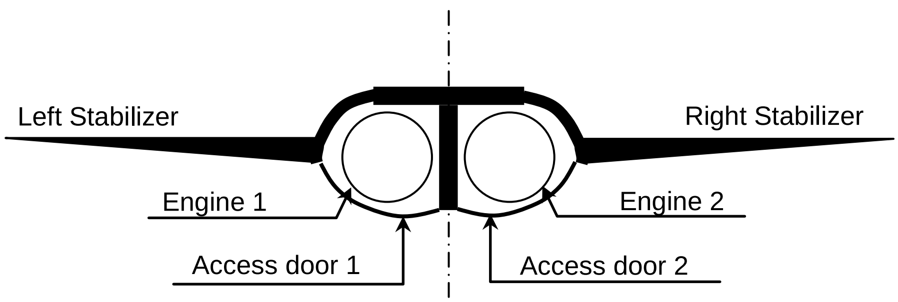


Figure 3.14 Schematic of the Aft Fuselage Cross Section of the F-111A

It does not take a structures expert to realize that this can lead to severe torsional stiffness problems. Because of the obvious lack of torsional stiffness, a new type of flutter mode arose: asymmetric horizontal tail bending driving fuselage torsion and, in turn, asymmetric wing bending, or vice versa.

3.9.3 Solution

To solve the problem a very significant increase in torsional stiffness was needed. This in turn increased the weight of the aft fuselage. As a result the center of gravity of the airplane moved aft and a large amount of lead ballast had to be added into the nose compartment. Both factors drove up the empty weight of the airplane to the point that its mission effectiveness was significantly eroded.

This problem should have been discovered during early design. In fact, that was the case at Boeing which competed for this fighter-bomber contract when it was still called the TFX. Boeing built a flutter model and found this flutter mode in the tunnel. To eliminate this flutter mode Boeing engineers decided to use a primarily titanium structure in the aft fuselage for the USAF airplane. The Navy version (which was not required to fly at such high dynamic pressures) was to have a conventional aluminum structure. This cost a certain amount of commonality, but allowed a reasonably low weight in the US Navy version.

General Dynamics engineers were not aware of this problem until it showed up during early flight testing. The increase in empty weight made the airplane too heavy for carrier operations and the US Navy cancelled its part of the contract for the F-111B. The Navy then developed its own fighter which became the Grumman F-14A: see Figure 3.15.

The USAF was stuck with the F-111A and, to stay within reasonable development costs, decided to degrade the mission requirements: the supersonic, low altitude requirement was dropped. The US taxpayers ended up with a much less capable airplane.

Figure 3.15 Grumman F-14A, Tomcat (Not accident aircraft, Courtesy Royal Aeronautical Society Library)

3.9.4 Lesson

When a contract goes to the least informed company and technology transfer between companies is not required, the taxpayer ends up with a much less capable airplane.

3.10 Corrosion Fatigue

3.10.1 Problem

In May of 1971 a DeHavilland Model 104-7AXC (Figure 3.16) crashed near Coolidge, AZ. The airplane was observed to enter a shallow dive which steepened as it approached the ground. Two crew members and all ten passengers were killed.

Figure 3.16 DeHavilland 104 Dove (Not accident aircraft, Courtesy F. Duarte Jr.)

3.10.2 Cause

According to Ref. 3.6 the probable cause of this accident was: "the in-flight failure and subsequent separation of the right wing. This failure was the result of a fatigue fracture in the lower main root joint fitting which propagated from an area of corrosion and fretting damage which, in turn, was probably caused by design deficiencies. These design deficiencies remained undetected, because surveillance of the supplemental type certification process and the modification programs was not adequate to assure compliance with design and inspection requirements."

The reader should understand how the FAA approval process of aircraft modifications works. To that end, the following is taken from Ref. 3.6:

"The right lower wing fitting was subjected to an eddy current inspection 1,651 hours before the accident. This inspection was performed in compliance with Airworthiness Directive (AD) 70-15-6. That AD, which resulted from a prior accident involving a standard Dove aircraft, required inspection of all DH-104 wing fittings at 2,500 hour intervals. The last visual inspection was performed 2 weeks prior to the accident. This inspection did not require removal of the attachment bolt.

This particular aircraft had been modified with engineering approved by Supplemental Type Certificate (STC) SA1747WE. The STC, dated July 23, 1968, was issued to Von Carlstedt Corporation, C-Air, Long Beach, CA.

The aircraft modification consisted of the installation of two AiResearch TPE 331 series engines, an increase in fuselage length and relocation of the wing fuel tanks. Von Carlstedt subcontracted the engineering associated with this modification to Strato Engineering Co., Burbank, CA. The heat treatment of various fittings was subcontracted to Comet Steel Treating Co., Signal Hill, CA.

A significant aspect of the modification was the redesign of the wing lower main root joint fittings to accommodate the new engine installation and the relocation of the fuel tanks. The new fitting, part number CPD-2004, was structurally similar to the original fitting. This similarity was the basis upon which design approval was issued without a requirement for substantiating fatigue tests.

The fatigue life of the CPD-2004 fitting was predicated upon the life of the original DeHavilland fitting, provided that the new fitting maintained the same precise tolerances and joint sealing procedure employed in substantiating the life of the original DeHavilland fitting. The critical nature of these procedures and tolerances was reported by DeHavilland in 1964 after that company failed a lower wing fitting at less than 25% of its predicted life during fatigue tests. DeHavilland established that this premature fatigue failure was caused by corrosion fretting of the fitting.

The stress analysis submitted to the FAA by Strato regarding the CPD-2004 fitting noted that the service life of the fitting was predicated upon maintenance of the DeHavilland tolerances. However, the engineering drawing which was prepared, checked and released by Strato and subsequently approved by the FAA as part of the STC data, specified a tolerance which could result in 0.0022 inches greater diametrical clearance than that specified in the fatigue analysis.

The fatigue life of the CPD-2004 fitting was also predicated, in part, upon the use of a material with a higher allowable ultimate tensile strength than that used for the original fitting. Accordingly, the design drawing specified that the fitting be constructed from 4130 alloy steel, heat treated to a tensile strength from 180,000 to 200,000 p.s.i. The drawing did not, however, specify the process by which this heat treat was to be accomplished. According to Military Handbook 5A, which was used in the design of this modification, a part fabricated from 4130 alloy steel with the size and geometry of this fitting could not consistently be hardened throughout the section thickness to attain the specified tensile strength. Tables in the 5A handbook indicate that 4340 alloy steel would be preferred in order to attain the desired strength level.

Because of its interest in the types of aircraft currently in use in air taxi operations, the NTSB not only reviewed the modification of this aircraft but also the process by which the aircraft was certificated. Supplemental type certification is used when changes to the existing type certificate are not considered significant enough to require a new type certificate (TC); the STC is considered an amendment to the original TC.

The applicant for an STC must show that the altered product meets the applicable airworthiness requirements. However, the responsibility for assurance that the modification meets the standards of the Federal Aviation Regulations (FAR's) rests with the FAA and is accomplished by FAA Engineering and Manufacturing personnel in the regional offices.

In actual practice, most of the review of an STC program is accomplished by employees of the applicant who act as representatives of the FAA, and who are titled Designated Engineering Representatives (DER's). DER's are appointed at the convenience of the FAA; they are guided by the same requirements, instructions and procedures as FAA employees; and the amount of review of their work is dependent, in part, upon the confidence the FAA regional personnel have in their capability.

The duties and responsibilities delegated to a DER are outlined in FAA Handbook 8110.4, "Type Certification." That handbook notes that a DER has the authority either to approve specific data (subject to spot review by the FAA) or to recommend that FAA approve the data. The handbook also notes that, in approving data, the DER must completely satisfy himself that all pertinent FAR requirements are complied with. He must accept the responsibility for approving the technical data as complying at least with the prescribed minimum airworthiness standards. However, the Chief Engineer of Strato Engineering Co., who functioned as a DER in the structures and flight test areas, testified that in one case his signature on technical data merely indicated that he had reviewed the data and that he thought it was a proper document. In arriving at this conclusion, he approved the general approach used in the calculations, but he did not check the numerical accuracy. He felt that actual approval of the data was the responsibility of the FAA. He also noted that, although he initialed drawing CPD-2004 as a DER, he did not check it for material allowables.

Another DER on this project testified that, with the exception of Handbook 8110.4 he had not been provided guidance regarding his duties and responsibilities as a DER.

In addition to its responsibility for design adequacy, the FAA has a responsibility to assure that the modified aircraft conforms to the design drawings. The conformity inspections of the aircraft were performed by FAA Manufacturing Inspectors from the local district office. The inspector who performed the majority of these inspections said that these inspections were done on a sampling basis. He also said that he had no instructions from the regional engineering personnel as to what he should inspect or check.

Although the discrepancy in the material selection/heat treatment criteria remained undetected, the NTSB noted that the manufacturing inspector rejected the fitting on the basis of its strength. This part was rejected because a hardness test on another part from the same heat treat lot was not within its hardness specifications and the entire lot was rejected.

The inspector did not, however, follow up to assure compliance with his request for a subsequent inspection to determine that this part was properly heat treated. Although the procedures used for the ultimate acceptance of this part were never determined, the fitting was subsequently installed, and the aircraft was certificated."

The following is again taken from Ref. 3.6:

"In reviewing the design of this modification the NTSB noted two errors which affected the fatigue life and load carrying capability of the CPD-2004 fitting:

The failure to transfer information regarding dimensional tolerances from the design data to the engineering drawing from which the parts were manufactured. This omission seems particularly significant to the NTSB in view of the known premature failure of the DeHavilland fatigue test specimen, and that company's finding that the failure was related to bolt tolerances. Although deformation of the failed fitting in the accident aircraft precluded the determination of the actual diameter of the hole, the hole tolerance callout on the engineering drawing was considerably larger than that specified in the fatigue data. Excessive clearances could have caused high bearing stresses at the hole wall. The NTSB, therefore, concludes that this increase in clearance may have contributed to the initiation of the fracture.

The selection of an alloy (4130) that did not harden uniformly in the various sections of the fitting when the part was heat treated. This resulted in a fitting which had lower average tensile strength than the value used in the stress analysis. The NTSB believes that this lower strength may also have contributed to the premature failure of the fitting.

In addition to the influence of these two design errors on the cause of this accident, other facets of the certification program must be considered significant. For example, both of the errors discussed might have been detected if the DER's had properly reviewed the design data and engineering drawings which they, in effect approved by affixing their signatures or initials thereto. However, the NTSB noted that the DER's involved were not fully aware of the responsibilities associated with that position. Also, erroneous heat treatment callout on the design drawing might well have been detected by the Manufacturing Inspector if he had followed up on his rejection of the entire lot in which the fitting was heat treated.

Thus, the factors which permitted certification of this aircraft seem to derive from the general nature of the implementation of the STC program. In theory the system may work well, but, as implemented in this case, it allowed this problem to develop. In retrospect, it is quite clear that adequate communication among all parties concerned, and increased surveillance by the FAA of the STC process and of the parties implementing this program, might have prevented this accident."

3.10.3 Solution

It is the opinion of the author that this sequence of events reflects a breakdown and failure of the certification process. In this case, with obviously tragic results.

The NTSB, in Ref. 3.6 made the following recommendations to the FAA:

1. Conduct a one-time metallurgical inspection on an expedited basis by approved methods of all lower main spar root fittings PIN CPD-2004 on all DeHavilland Model 104 “Dove” airplanes that were modified under STC No. SA1747WE.

2. Review the adequacy of Airworthiness directive 70-15-6 and revise as necessary to assure adequate service limits on this fitting.

The FAA agreed with these recommendations and also suspended the airworthiness certificate of these airplanes until remedial action was taken. For the affected aircraft this remedial action consisted of the installation of a steel reinforcing strap on the lower front spar cap, and replacement of the upper wing fittings with identical parts fabricated from 4340 steel.

The NTSB also concluded that the FAA re-evaluate its STC program to ensure continuity in quality control in the supplemental type certification process.

3.10.4 Lessons

1. For design engineers the lesson is that all flight crucial components of an airplane structure should receive double scrutiny of:

   - Any calculations made
   - Any assumptions made leading up to those calculations
   - Any drawing callouts with regard to manufacturing tolerances
   - Any callouts made in regard to special treatments

2. For manufacturing personnel the last two items certainly require their special scrutiny.

3. Clearly, in this case, there was a problem with the DER system. DER’s, just like pilots, hold the lives of people in their hands. Qualifications, training, continuing education and understanding of ethics should be made part of the process of appointing a DER. There should also be periodic reviews of whether a DER still qualifies to hold that designation.

3.11 Rear Pressure Bulkhead Failure I

3.11.1 Problem

In October of 1971 a Vickers Vanguard (Figure 3.17) crashed over Belgium. There were no survivors.

Figure 3.17 Vickers Vanguard (Not accident aircraft, Courtesy G. Helmer)

3.11.2 Cause

The cause was established: the rear pressure bulkhead had failed at cruise altitude (Ref. 3.7, pp 176-184). High pressure air from the cabin had entered the un-pressurized aft fuselage cone and from there entered the horizontal stabilizer box. Several upper skin panels of the stabilizer box, not having been designed to take high pressure loads from the inside, peeled open. This caused extensive flow separation over the tail which in turn made the airplane pitch up. Major structural breakup ensued.

It was also found that the reason for the pressure bulkhead rupture was a corrosion induced fatigue crack which opened up rapidly. The corroded area was not easily detectable during normal inspections of the airframe. Figure 3.18 shows a sketch of the pressure bulkhead installation.

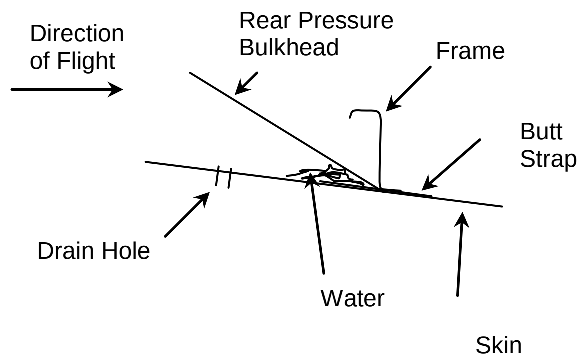


Figure 3.18 Rear Pressure Bulkhead with Water Trap

Note the poor location of the drain hole. As a result water could accumulate which caused the corrosion. During normal structural inspections this particular area was very difficult to see and thus the corrosion remained undetected.

3.11.3 Solution

Solutions which suggest themselves are:

1. arrange for positive drainage of areas where water can collect (always in bottom areas)
2. arrange for easy inspectability of all primary structural components from all sides
3. design a pressure venting system in the aft fuselage cone to prevent high pressures from building up due to whatever cause

One result from this accident has been the incorporation of pressure ventilation devices in the aft fuselage cones of all Airbus airplanes, starting with the A300. This design decision has paid off.

In 1973 a Thai Airways A310 experienced a rear pressure bulkhead failure due to a hand grenade which was exploded by a gangster who was trying to commit suicide in the rear lavatory. The explosion blew a hole in the rear pressure bulkhead. The pressure ventilation devices in the rear fuselage cone prevented further damage to the airplane which was landed safely. Even the gangster survived the incident.

3.11.4 Lessons

1. All primary structure should be designed so that inspection can be easily carried out.

2. If a failure of a rear pressure bulkhead occurs it is not acceptable for high pressure cabin air to cause such damage as to render an airplane uncontrollable.

3.12 Crack Propagation I

3.12.1 Problem

In May of 1974 a Lockheed L-382 (Figure 3.19) of Saturn Airways crashed near Springfield, IL.

Figure 3.19 Lockheed L-382 (Not accident aircraft, Courtesy www.al-airliners.be and Saturn Airways)

Three crew members and a route supervisor were killed. The outboard section of the left wing, including the No.1 engine, separated in flight from the remainder of the wing.

3.12.2 Cause

In Ref. 3.8 the NTSB determined the most probable cause to be the undiscovered, pre-existing fatigue cracks, which reduced the strength of the left wing to the degree that it failed as a result of positive aerodynamic loads created by moderate turbulence.

From Ref. 3.8, page 9:

“Metallurgical examination of the fractured surfaces of the left wing at Outer Wing Station (OSW) 162 revealed that the lower front spar cap fractured completely in fatigue. The spar cap was deformed at the primary origin area of the fatigue fracture. Hardness and electrical conductivity of the spar cap material were normal for 7075-T6511 aluminum alloy.
The lower portion of the front spar web contained an approximate 4.9 inch fatigue crack with intermittent tensile tearing.

The lower wing skin fracture stemmed from pre-existing fatigue cracks at the first fastener hole which was located ¾ inch outboard of the primary origin area at which the spar cap failed. Deformation and multiple cracks were noted at the origin of the skin fatigue fracture in the lower wing.

Further study showed that if the above cracks existed before the accident, a wing loading of about 60% of limit load would be required to make the 4.9 inch crack progress. When the crack progressed beyond the 4.9 inch mark, the stress intensity factor in the lower skin panel would approach the critical value and would trigger crack instability in the panel. From there on, the crack propagation in both the panel and the web would have been simultaneous. With this condition, total failure of the wing section would be expected.”

A fatigue crack monitoring program had been in effect for USAF Hercules aircraft since 1969. About 700 aircraft which had accumulated over 6,000 flight hours were and are being surveyed. As a result, Lockheed developed Engineering Change Proposal (ECP) 954 in October of 1971. This ECP resulted in fatigue preventive modification kits which were applied to all USAF aircraft. The record of cracking of the forward spar near engines No.1 (left wing) and No.4 (right wing) disclosed 36 cases of which 11 cases were between OWS 156 and 162.

The accident aircraft did not have this wing modification kit installed.

That fatigue cracks are not always easy to identify may be seen from Figure 3.20. Some crack origins may be hidden by other items of the structure.

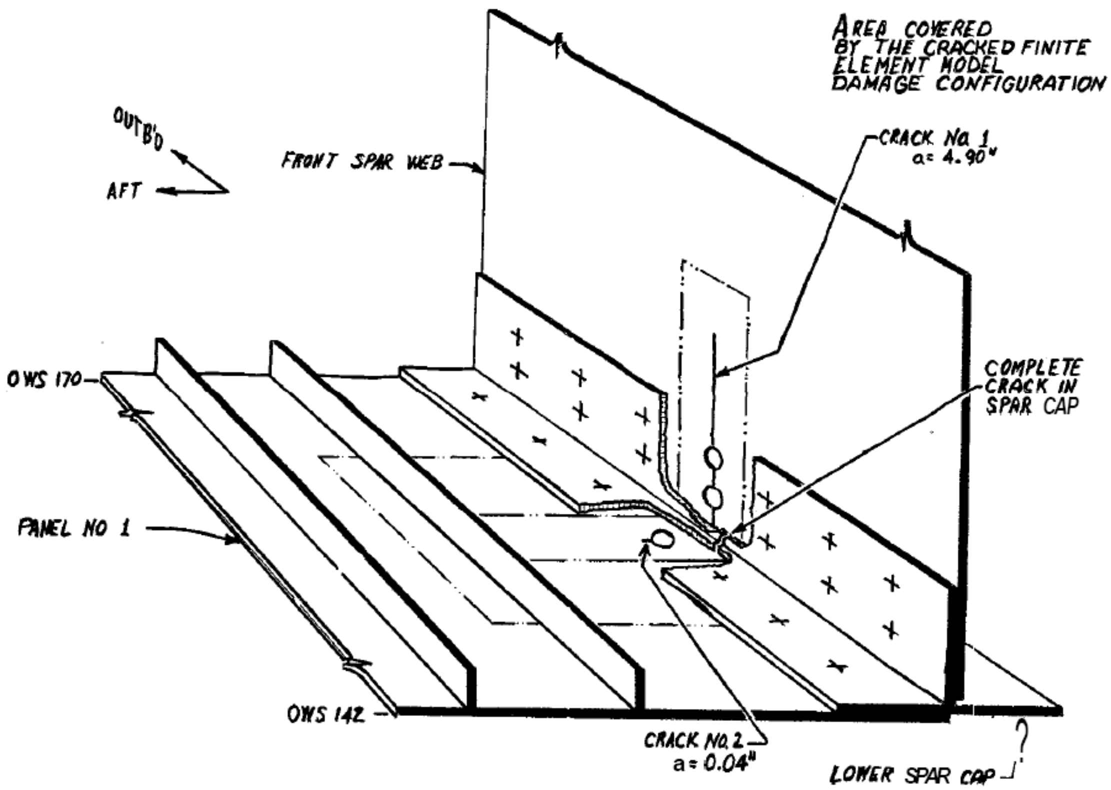


Figure 3.20 Area of Fatigue Cracks in N14ST (Courtesy NTSB)

3.12.3 Solution

As a result of the NTSB investigation the FAA issued Air Worthiness Directive 74-12-06 LOCKHEED: Amendment 39-1867. This AD required that all applicable aircraft be inspected. If cracks were found in the designated inspection areas, repairs were required in accordance with Lockheed Service Bulletin 382-169A. Aircraft with no cracks were required to be inspected at 1,000 hour intervals. Aircraft with the repair installed were exempt from the periodic inspections.

The inspection of all U.S. registered L-382 aircraft revealed one aircraft with the lower forward spar cap cracked through the entire cross section at OWS 160. The flight hours on that aircraft were over 16,000.

3.12.4 Lessons

1. In the days prior to the use of finite element methods for structural analysis it was very difficult to predict all areas where fatigue might become an airworthiness issue. Today, identifying the actual stress levels in critical areas of the structure is relatively easy, given certain user load spectra. Therefore it should probably be required by the FAA that all new aircraft be analyzed in this manner.

2. All areas of the structure which are considered flight crucial should be inspectable. Designers, should verify that this can be done at the earliest opportunity. With today's CAD methods this task has also become easier.

3.13 Horizontal Stabilizer Failure

3.13.1 Problem

In 1977 a Dan Air Boeing 707 (Figure 3.21) crashed while on final approach to the Lusaka Airport in Zambia. There were no survivors.

Figure 3.21 Boeing 707 (Not accident aircraft, Courtesy www.al-airliners.be and Dan Air)

3.13.2 Cause

The cause was established to be an in-flight structural failure of the starboard horizontal stabilizer due to simultaneous crack nucleation and propagation from co-linear holes, which operated at or near the same stress level.

3.13.3 Solution

This particular structural failure case caused a major revision to be made in the way aircraft structures must be designed and certified. It is of interest to quote the following from p.30 of Ref. 3.9:

"Before 1978, CAR 4b.270 and 14 CFR §25.571 required that airplane structures whose failure could result in catastrophic failure of the airplane be evaluated under the provisions of either fatigue strength or fail-safe strength requirements. If the structure was not demonstrated to withstand the repeated loads of variable magnitude expected in service, it had to be fail-safe. A fail-safe structure is one in which catastrophic failure or excessive structural deformation that could adversely affect the flight characteristics of the airplane are not probable after fatigue failure or obvious partial failure of a single primary structural element (PSE). After these types of failures of a single PSE, the remaining structure must be able to withstand static loads corresponding to the required residual strength loads. If the concept of fail-safe was impractical, structures were certified using the safe-life concept. Most common examples of structures that have been certified to safe-life were landing gear components and structure associated with control surfaces."

The Lusaka accident proved that these concepts were inadequate since the 707 stabilizer had been certified under the fail-safe rule.

It turned out that the structural damage that had accumulated in the Lusaka case was not readily detectable. Therefore, on October 5, 1978, Amendment 25-45 of the rule was issued. This amendment added the concept of damage tolerance. Quoting from Ref.3.9:

"Damage tolerance is the attribute of the structure that permits it to retain its required residual strength for a period of use after the structure has sustained a given level of fatigue, corrosion, or discrete source damage. Amendment 25-45 requires that a damage tolerance assessment of the structure be accomplished to determine the most probable location of damage and to provide an inspection program that requires directed inspections of critical structure. The damage tolerance assessment philosophy essentially replaced the fail-safe and safe-life design philosophies.

3.13.4 Lesson

The lesson for structures designers is that all critical areas of the structure (i.e. those areas where the most probable damage will occur) must be readily inspectable.

After the Lusaka accident Boeing and the FAA warned other operators to inspect 707's with a similar service history (flight cycles and flight hours). Several operators did and found stabilizers that were also close to failure. This action prevented further serious accidents.

3.14 Elevator Structural Failure

3.14.1 Problem

In December of 1984 a Provincetown-Boston Airlines Embraer EMB-110P1 Bandeirante (Figure 3.22) crashed near Jacksonville, FL. All 11 passengers and two crew members were killed.

Figure 3.22 Embraer 110P1 Bandeirante (Not accident aircraft, Courtesy of Ellis M. Chernofft)

3.14.2 Cause

Ref. 3.10 identifies as the probable cause: "the malfunction of either the elevator control system or the elevator trim system which resulted in an airplane pitch control problem. The reaction of the flight crew to correct the pitch control problem overstressed the left elevator control rod which resulted in asymmetrical elevator deflection and overstress failure of the horizontal stabilizer attachment structure."

The NTSB did not find it possible to determine why the elevator trim tab deflected to its full trailing edge up position, only that it did.

The result was a sudden nose-down pitching motion of the airplane which the pilots attempted to correct by pulling aft on the elevator controls with high pull forces. This action produced a compression load in the left elevator control rod which exceeded the design strength of the rod and caused it to fracture. Figure 3.23 shows a perspective view of the elevator controls.

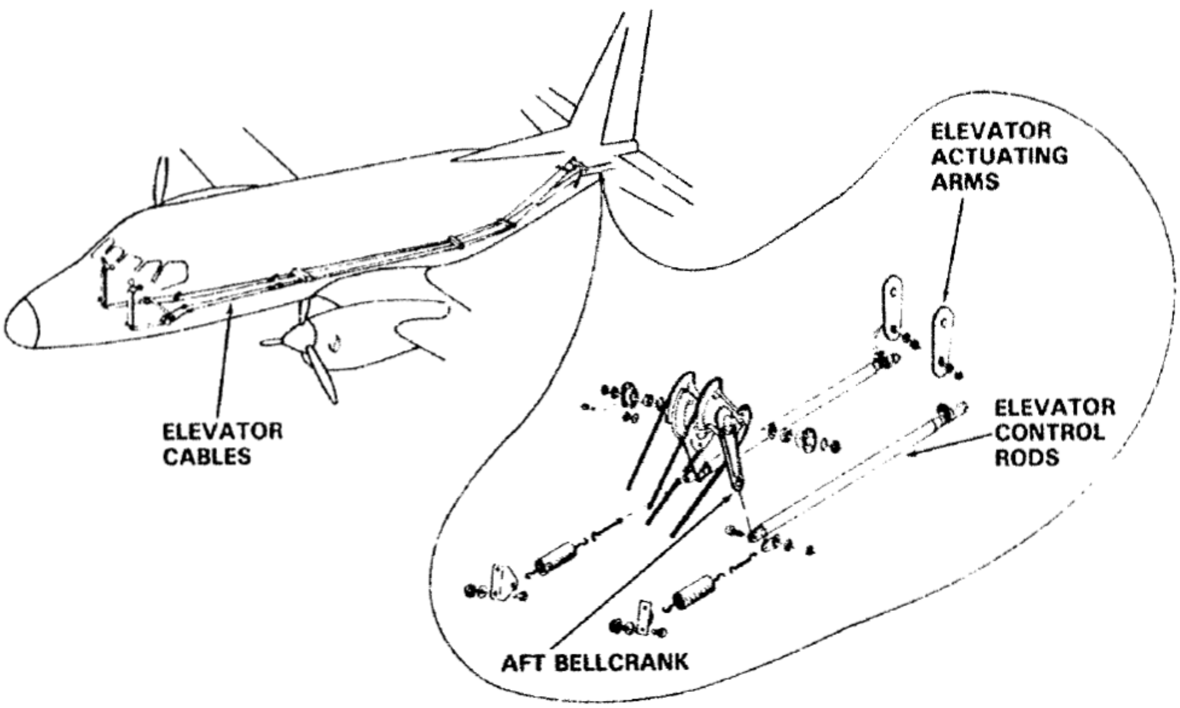


Figure 3.23 Perspective View of the Elevator Controls in the EMB-110P1 (Courtesy NTSB)

With the restraint of the left control rod removed, the left elevator instantaneously reacted to the aerodynamic load produced by the fully deflected trim tab and moved rapidly trailing edge down. Simultaneously, the fracture of the left control rod caused the high pull forces on the pilot control column to transfer fully to the intact right elevator control rod, which rapidly forced the right elevator to move trailing edge up. The combination of about 170 knots airspeed and the differential elevator deflection produced high asymmetrical aerodynamic loads on the horizontal stabilizer which exceeded the strength of the forward attachment structure.

Figure 3.24 shows the stabilizer attachment structures. The horizontal stabilizer then separated from the airplane in a clockwise twisting motion.

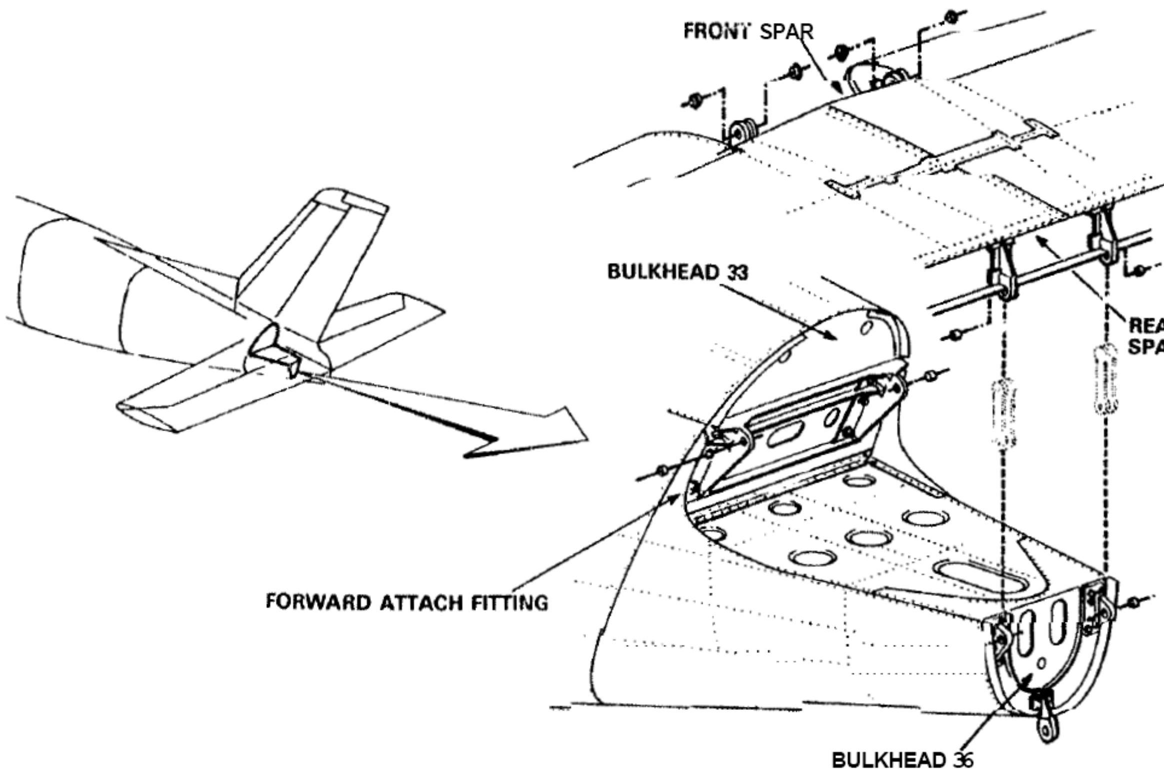


Figure 3.24 Horizontal Stabilizer Attachment Structure of the EMB-110P1 (Courtesy NTSB)

3.14.3 Solution

The NTSB, in Ref. 3.10, determined that the airplane had been properly certified according to the loads and strength criteria in effect in 1969 when the airplane was certified.

The NTSB also found that neither the design criteria nor the certification requirements included a structural design load consideration for anti-symmetric aerodynamic loading of the horizontal stabilizer. The Safety Board agreed that because it is not possible to achieve such a loading condition absent other failures which could render the airplane uncontrollable, an anti-symmetric loading condition is not a reasonable design consideration.

This author disagrees. Since the elevator control rod failure was caused by pilot control forces it could and should have been predicted and, of course, been corrected.

Because in-flight structural failures always raise questions with regard to the certification process a Special Certification Review team was set up to evaluate design load criteria.

On page 39 of Ref. 3.10 the NTSB notes:

"The Special Certification Review Report did not specifically address the certification of the airplane as it related to *control* system strength or to *trim* system run-away protection. The Safety Board is concerned since the accident that a failure of a primary part of the airplane flight control system could be achieved by a pilot-applied load, notwithstanding that the load was applied by two pilots, both pulling at near maximum strength on their control wheels. Although the total load resulting from the efforts of both pilots far exceeds the reacting aerodynamic loads achievable within the airplane flight envelope, such a load might be required to overcome a jammed flight control condition. The FAR addressing flight control system design strength has remained unchanged since the certification of the EMB-110P1 and P2 and specifies that the flight control system strength be designed to withstand the maximum effort of the pilot applied to the system; this maximum effort is defined as a 238-pound force applied to the elevator control wheel. The strength of the EMB-110P1 and P2 flight control system, including the elevator control rods far exceeded this requirement. In further consideration of the design strength of the systems, the load applied to the aft bell crank is normally divided between the left and right elevator control rods, each of which is capable of withstanding the maximum control system force which can be applied by one pilot. Furthermore, the left and right elevator control rods are considered to be redundant because an in-flight failure of either rod will result in free elevator only on the side of the failure. The airplane can be controlled in pitch by the remaining elevator. The fallacy of the redundancy consideration, however, is the effect of a highly deflected trim tab on a free elevator which, as demonstrated in this accident, can cause anti-symmetric loading of the stabilizer. The Safety Board acknowledges that the EMB-110P1 and P2 flight control system design strength complied with the certification standards. Further, the conditions of this accident were unique in that the elevator trim tab was fully deflected, and the pilots were applying maximum force to achieve a desperate maneuver. However, the Board believes that the elevator control system should be of sufficient strength to withstand the maximum efforts of both pilots."

Since the accident, both the CTA (Canadian Transport Authority) and the FAA have required operators to install the higher strength elevator control rods in EMB-110P1 and P2 airplanes.

3.14.4 Lesson

In control system design of airplanes with two pilots the system itself should always be designed to withstand both pilots simultaneously applying maximum force to the cockpit controls. This just seems the "right" thing to do, regulations or not.

3.15 Rear Pressure Bulkhead Failure II

3.15.1 Problem

In 1985 a Japan Air Lines Boeing 747 (Figure 3.25), carrying 509 passengers and a crew of 14, crashed in Japan. There were only four survivors. A detailed description of the event may be found in Refs. 3.11 and 3.12.

Figure 3.25 Boeing 747 Prototype (Not accident aircraft, Courtesy Boeing)

3.15.2 Cause

The cause was established. The rear pressure bulkhead had failed in cruise flight and high pressure air flowing into the rear fuselage cone caused enough damage to cause massive flow separations. This resulted in severe lateral vibrations which broke all four hydraulic system lines serving the longitudinal and directional flight controls. The vertical fin also separated from the airplane. The hydraulic system then lost all fluid. As a result the airplane was essentially uncontrollable and crashed into terrain.

The cause of the pressure bulkhead failure turned out to be a faulty repair which led to rapid fatigue. Figure 3.26 shows three sketches of a cross section of the rear pressure bulkhead: a) original, b) intended repair due to crack in original bulkhead and c) (faulty) repair carried out on the accident airplane.

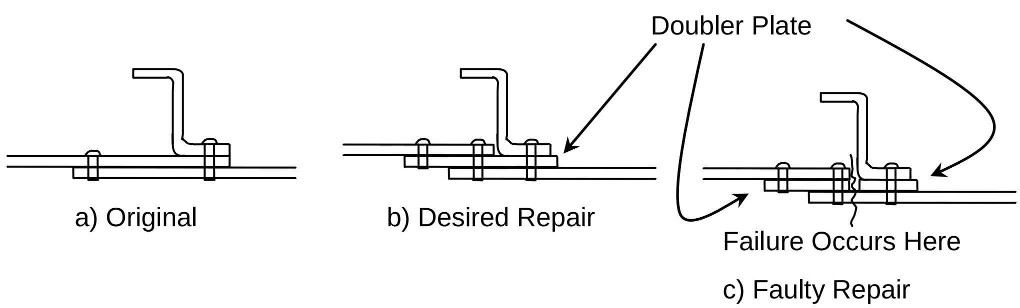


Figure 3.26 Cross Section of Rear Pressure Bulkhead

3.15.3 Solution

All 747 airplanes had to be modified to include:

- non-return valves in the hydraulic system in the rear fuselage cone
- pressure ventilation devices in the rear fuselage cone

3.15.4 Lessons

1. It would seem desirable for design engineers to become familiar with past accidents and accident reports. Had the report of the accident described in Section 3.10 been read by the

designers (which happened in 1971, a full fourteen years earlier!) the pressure ventilation devices would have been installed voluntarily and the airplane probably would have remained controllable.

2. Also, since pressure bulkhead failures seem to occur with some regularity designers would do well to keep this in mind and assure themselves that the consequences of such failures are benign

3.16 Crack Propagation II

3.16.1 Problem

In April of 1988 an Aloha Airlines Boeing 737 (Figure 3.27) suffered an explosive decompression of the fuselage. One flight attendant was sucked through the hole and presumably died. The airplane was safely landed.

Figure 3.27 Model of Boeing 737 (Courtesy geminijets.com)

3.16.2 Cause

The cause was determined to be the undetected presence of widespread fatigue damage (WFD) to the upper fuselage skin panels. The airplane had been used for inter-island service and accumulated as many as 16 pressurization cycles per day for a grand total of 89,680 flights.
Refs. 1.3, and 3.13 – 3.15 contain detailed information on this accident.

According to Ref. 3.13, p. 49, one of the passengers had seen a crack in the fuselage while boarding the airplane. She had not mentioned this to anyone. After the accident she was asked to point out this area in another not affected 737. The area she pointed out turned out to have been the area where the failure had begun.

Quoting from Ref. 3.15 (p.48-49):
"Where a 737 might normally undergo one or two pressurization and depressurization cycles in a working day, the short flights between islands meant that Aloha aircraft were undergoing this pattern of repeated stresses and relaxations six or seven times as often. Moreover, the combination of warm, humid, salt air around these tropical islands was the most corrosive atmosphere possible. Moisture was penetrating the joints between panels, and when this loosened the bonding between some panels, the entire stress of pressurization and depressurization was taken by the rivets fixing the panels together.

The investigators checked the rivets more closely, and found the holes were drilled with a beveled profile so that the countersunk rivets could be fitted flush with the top panel for minimum aerodynamic drag. However, the problem was that the beveled section extended all the way through the top layer of skin, leaving a sharp circular edge at the bottom of each hole (Author's comment: see Figure 3.28).

As the rivets were stressed and relaxed with each pressurization-depressurization cycle, these sharp edges provided perfect starting points for fatigue cracks at each rivet.

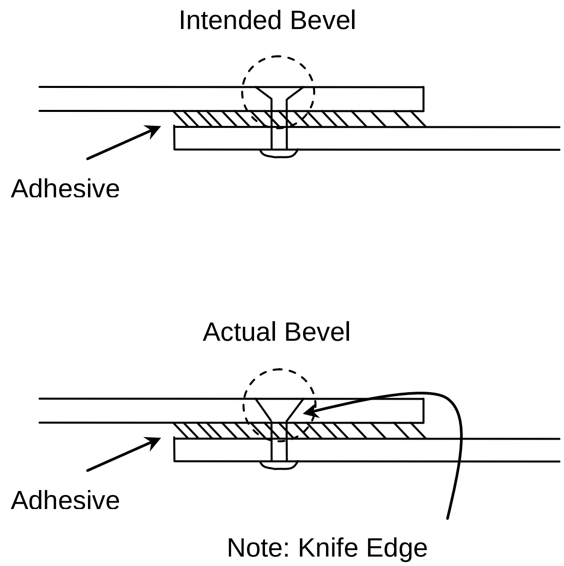


Figure 3.28 Sketch of an Actual and an Intended Rivet Installation

Normally, fatigue cracks are only able to extend for a short distance before they reach a reinforcing member designed to limit their travel. They can then only extend at right angles to their original direction, so that if the problem is uncorrected and a decompression occurs, a small flap opens up and the rush of air out through the hole will not damage the structure. In this case, it seemed as if fatigue cracks had appeared along the whole row of rivets, creating a fault line which crossed a whole succession of reinforcing members. As a result, when the cracks caused one section of paneling to fail, the failure was able to jump from one fuselage section to the next, until the whole forward cabin roof was torn away in the slipstream."

3.16.3 Solution

From Ref.1.3, p.31: "As a result of this accident a number of 'geriatric' airplane initiatives were launched by the FAA and the industry:

- Publication of select service bulletins describing necessary modifications and inspections
- Development of inspection and prevention programs to address corrosion
- Development of generic structural maintenance program guidelines for aging airplanes
- Review and update of supplemental structural inspection documents (SSIDS) that describe programs to detect fatigue cracking
- Assessment of damage tolerance of structural repairs
- Development of programs to preclude WFD in the fleet.

3.16.4 Lessons

1. Again, all primary structural features of an airplane (and that includes the skin of the pressure cabin) should be readily inspectable.

2. To some extent the external painting scheme used by Aloha made visual detection of cracks very difficult. The lesson therefore is: from an inspectability viewpoint, the less paint, the better.

3.17 Cargo Door Hinge Design

3.17.1 Problem

In February of 1989 a United Airlines 747 (Figure 3.29) cargo door opened in flight (see Ref. 3.16, pp 31-47 and Ref. 3.17).

Figure 3.29 Boeing 747-122 (Not accident aircraft, Courtesy Ellis Chernoff)

A large hole was torn into the passenger deck sidewall structure. Nine passengers, eight still strapped into their seats were sucked out of the airplane. They were never found. Structural debris "fodded" engines number 3 and 4 and the crew had to secure them. Structural debris also caused significant damage to the airplane in several other areas. The crew made a satisfactory emergency landing in Honolulu.

3.17.2 Cause

An investigation showed that the cargo door blew open because it was not properly latched in place before departure. More detailed investigation showed various undesirable aspects of the design of the door latching mechanism (Ref. 3.17).

When a picture (Figure 3.30) of the damage done to the side of the airplane is examined, one is drawn to conclude that the door hinge structure, once the door opened in flight, tore a large part of the passenger cabin skin open, more or less like a sardine can.

Figure 3.30 Damage Done by a Cargo Door Opening in Flight (Courtesy NTSB)

3.17.3 Solution

The FAA at the recommendation of the NTSB issued a number of airworthiness directives dealing with the latching mechanism to prevent the cargo doors from opening again. In the author's view, the NTSB and the FAA both failed to question the very design of the door hinge structure.

3.17.4 Lesson

No matter how careful cargo door latching mechanisms are designed and inspected there continue to occur incidents with such doors. It appears to the author that a design criterion for the door hinge structure should be to fail without opening the skin of the airplane. Designers should consider this in future designs.

3.18 Vertical Tail Fatigue due to Vortex Shedding

3.18.1 Problem

Early in the service life of the F-18A (Figure 3.31) it was discovered that the vertical tail attachment fittings to the fuselage were showing serious signs of fatigue.

Figure 3.31 McDonnell-Douglas F-18A (Courtesy NASA)

3.18.2 Cause

The cause was found to be vortex shedding from the strakes running from the wing to the fuselage. Figure 3.32 shows a sketch indicating the path of the vortices.

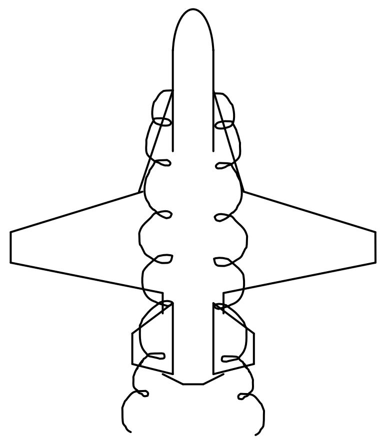


Figure 3.32 Paths of Strake Vortices toward the Vertical Tail

These vortices were found to impose large dynamic loads on the bottom of the stabilizers during maneuvering flight. Ref. 3.18 contains CFD generated depictions of such vortices.

3.18.3 Solution

The solution was a rather extensive redesign and remanufacturing of the vertical tail attachment structure.

3.18.4 Lesson

This event was predictable. From water-tunnel studies it can be determined under what conditions of angle of attack and sideslip vortices will hit the vertical tail. Knowing the intensity of maneuvering experienced by fighters in training and in combat a dynamic load spectrum should have been predicted. The results could then have been accounted for in the structural design.

3.19 Design Instructions Ignored

3.19.1 Problem

In September of 1995 a Magal Cuby II ultralight (Figure 3.33) crashed near Legal, Alberta.

Figure 3.33 Cuby II Ultralight (Not accident aircraft, Courtesy A. Presterud)

Witnesses observing the airplane heard a loud report and saw pieces falling from the airplane. They also observed that the outer section of the left wing was missing. The student pilot and his instructor were fatally injured.

3.19.2 Cause

Ref. 3.19 contains a description of how this accident occurred. Figure 3.34 shows a designer's sketch of typical front and rear spar cross sections indicating the type of wood and the grain direction which should be used. Figure 3.35 shows the factual arrangement used by the builder.

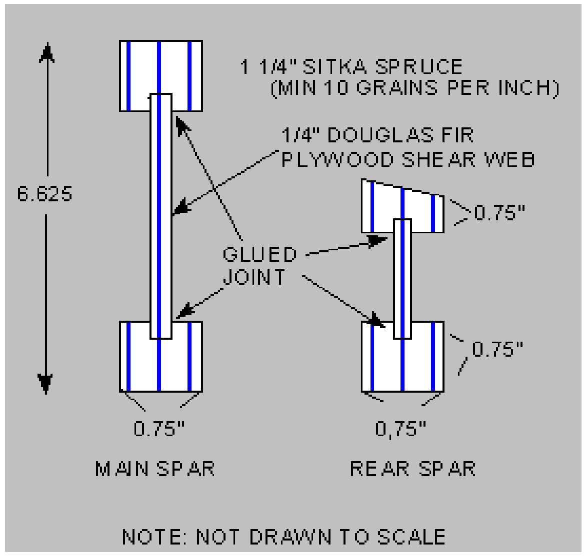


Figure 3.34 Designer's Sketch of Spar Cross Sections
(Courtesy Transportation Safety Board of Canada)

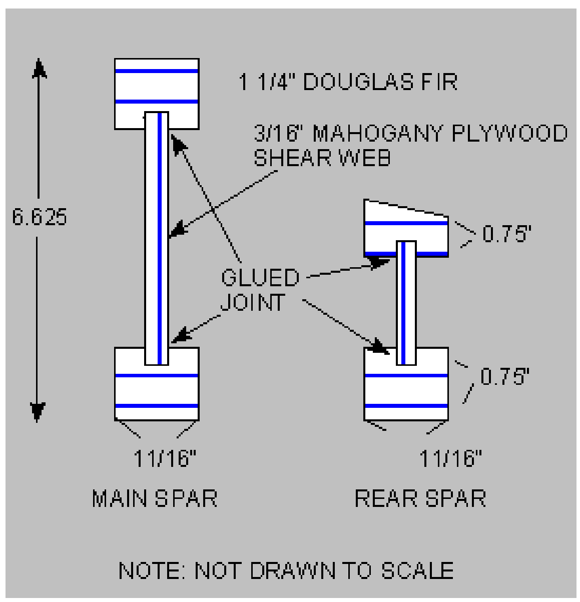


Figure 3.35 Actual Spar Cross Sections in the Accident Aircraft
(Courtesy Transportation Safety Board Canada)

The following is taken from Ref. 3.19:

"Examination of the aircraft's failed left wing spar indicated that they were not constructed in accordance with the designer's sketch. The wood grain orientation of the failed spar was found to be at 90 degrees to the direction recommended and was unsatisfactory for straightness. The spar caps and webs were under-dimensioned; 3/16 inch mahogany had been substituted for the ¼ inch fir plywood shear web called for in the sketch. In addition, the spar cap wood material was fir and not sitka spruce, as specified. Further examination reveals that the structural stability of the aircraft's wing was questionable. Any sort of aerobatic maneuver, particularly ones requiring positive high angles of attack for entry, would be hazardous. There was also evidence of previous damage to the left wing in the form of a left wing tip spar and fabric repair. There were no wing inspection ports to allow for adequate periodic inspections of the internal wing structure."

3.19.3 Solution

Ultralight airplanes are exempt from airworthiness certification requirements. The author believes they should not be exempt.

3.19.4 Lessons

1. It would probably be a good idea to add notes to structural drawings of uncertified airplanes that the designer instructions, material specifications and dimensions must be adhered to.

2. Designing or building airplane structures which cannot be inspected periodically is ethically wrong

Chapter 4

Lessons Drawn from Flight Control System Design

"If control or electrical wires can be installed the wrong way, they will be"

Variation of Murphy's Law

4.1 Introduction

In this chapter a series of problems which arose in flight control system design are reviewed. Where applicable, causes and solutions are described and lessons learned are stated. Connections with other areas of design are identified in Appendix A.

4.2 Heat Source Close to Flight Controls

4.2.1 Problem

In 1944, during a test flight on the Westland Whirlwind I (Figure 4.1) the starboard aileron failed (Ref. 4.1, page 264). The aileron floated up and the airplane began to roll right. The test pilot managed to land the airplane by using the port side aileron which remained operative.

Figure 4.1 Westland Whirlwind I (Not accident aircraft, Courtesy Royal Aeronautical Society Library)

4.2.2 Cause

The lateral control push-rods were routed close to exhaust ducts on the Westland Whirlwind I. During that particular flight the exhaust duct failed, causing hot gasses to weaken the aileron push-rod which then failed in compression. It was also found that the part of the exhaust duct which failed was close to a main wing spar. Failure of that spar would have meant catastrophic failure of the wing. Luckily that did not happen.

4.2.3 Solution

The exhaust ducts were routed externally to avoid the controls and the spar.

4.2.4 Lessons

1. A good design rule is to never route flight controls close to heat sources.

2. Also, never expose primary aircraft structure to heat sources unless the structure is specifically designed to withstand high temperatures.

4.3 Ailerons Reversed I

4.3.1 Problem and Cause

In August of 1946, during the certification flight test program of the Avro Tudor 2 (Figure 4.2) the company was under a lot of calendar time pressure due to competition from Lockheed and Douglas (with the Constellation and DC-4 respectively).

Figure 4.2 Model of AVRO Tudor 2 (Courtesy www.collectorsaircraft.com)

During the first test flights it was discovered that the airplane needed a considerably larger vertical tail and rudder. The airplane also required significant modifications to its flight control systems. During a night shift new control cables were installed. This included new aileron control cables.

Although hard to believe, these modifications were made **without** the benefit of drawings and procedures (Ref. 4.2, p.12). Moreover, when the work was done, no functional control system checks were made and the airplane signed off as ready for flight.

The next day upon taxiing out for a test flight the crew did not notice that the aileron cables had been inadvertently crossed. Consequently, after take-off the airplane went out of control and crashed. There were no survivors.

4.3.2 Solutions

Before flying an airplane which has just undergone a major revision to its configuration and flight control system, common sense dictates that the control system functionality be checked as a matter of standard procedure.

Also, a flight crew should always verify the functionality of the flight control system before taxiing out on the first flight after repairs or modifications.

4.3.3 Lesson

It is essential that procedures for checkout of a new airplane be developed, recorded and adhered to before releasing the airplane for flight.

4.4 Gust Lock Engaged in Flight

4.4.1 Problem

In October of 1947 an American Airlines Douglas DC-4 (Figure 4.3) executed a violent maneuver. As a result, 5 crew members and 30 of the 49 passengers received minor injuries, and the aircraft received minor damage.

Figure 4.3 Douglas DC-4 (Not accident aircraft or airline, Courtesy G. Helmer)

4.4.2 Cause

According to Ref. 4.3 the probable cause was the engaging and disengaging of the gust lock in flight, which occurred without the pilot's knowledge.

There were three captains in the cockpit, one of them occupied the jump seat. The flight was uneventful until, without a control input, the airplane started to climb. To correct the climb the pilot-in-command (PIC) rolled the control for the elevator trim tab forward which would normally induce a nose-down attitude; however, the airplane continued to climb. The PIC continued to roll the elevator trim tab control forward, which resulted in increasing the nose-high or climbing attitude of the airplane. The PIC then attempted to return the elevator trim tab control to its former position. Before he could accomplish this, the aircraft pitched downward violently, executing part of an outside loop and actually becoming inverted.

The PIC and the pilot in the jump seat did not have their seatbelt fastened, and they were thrown to the top of the cockpit and accidentally struck the feathering control thereby feathering propellers Nos. 1, 2 and 4. The captain in the right hand seat had his seatbelt fastened and remained in his seat, managed to roll the aircraft out of its inverted position and regained control about 300 to 400 feet above the ground. Propeller Nos. 1, 2 and 4 were then unfeathered and the aircraft made a normal, albeit unscheduled landing at El Paso, Texas.

As a result of the violent dive, and the roll, many passengers who did not have their seatbelts fastened were thrown about the cabin. Parts of the wing de-icer boot were damaged and the interior cabin linings were torn.

Statements from the pilots to the investigators indicated that the captain seated in the jump seat had engaged the gust lock while the aircraft was in level flight. The other pilots stated that they were not aware of this action. The airplane started to climb and while rolling the elevator trim tab control nose-down, the PIC asked: “Is the automatic pilot on?” Upon receiving a negative reply, he thought of the possibility of the gust lock having become engaged in flight and reached for the trim tab control to neutralize it. Before this could be accomplished, the pilot in the jump seat released the gust lock lever, and it being spring loaded permitted the gust lock to return to the unlocked position. The elevator was then free to be moved by the trim tab which had been placed in an extreme upward or airplane nose-down position. The sudden and violent movement of the elevators to a down position caused the airplane to pitch down violently as previously described.

4.4.3 Solution

The captain seated in the jump seat was fired and his pilot license revoked.

4.4.4 Lessons

1. From a design viewpoint the question should be asked: "Should it even be possible to engage the gust locks while in flight?" Hopefully the answer is "No."

2. Not wearing a seatbelt when seated in the cockpit or passenger cabin is a bad idea.

4.5 Propeller Blade Severs Controls I

4.5.1 Problem

In February of 1948 an Eastern Air Lines Lockheed L-649 Constellation (Figure 4.4) en route from La Guardia to West Palm Beach and flying over the ocean, experienced a failure of the No. 3 propeller which resulted in a blade severing control cables, electrical wires and engine controls and fatally injuring a cabin crew member.

Figure 4.4 Lockheed L-649 Constellation (Not accident aircraft, Courtesy www.prop-liners.com and Eastern Airlines)

The airplane depressurized rapidly, heavy vibration was felt and all flight and engine instruments became impossible to read. The No. 4 engine kept running but could not be controlled. With controlled power only from engines No. 1 and 2 the airplane diverted for an emergency landing at Bunnell, Florida. During the emergency evacuation several passengers received minor injuries.

4.5.2 Cause

Ref. 4.4 states as the probable cause the failure of a propeller blade due to high stresses induced by accumulative engine malfunctioning.

The investigation showed that damage within the fuselage was confined to the galley section. This section is roughly in the plane of the inboard propellers.

Under the right side of the galley floor section pass three bundles of electrical wiring containing some 500 individual wires, cables controlling throttle and mixture settings, etc. of all four engines, and the cables controlling the elevator and rudder trim tabs. Nearly all of the wires were severed as were all the tab control cables and the throttle control of the No. 4 engine.

Examination of the No. 3 engine showed that the complete propeller and propeller shaft with connected stationary reduction gear, pinion and forward front sections were missing and had fallen into the sea.

The operational history of the No.3 engine was reviewed in detail. It was found that in its operational history of some 1,186 hours (461 since the last overhaul) it had a large number of reported irregularities. In fact three times more than comparable engines.

This persisting series of engine malfunctioning should have constituted a warning that the engine later might develop more serious trouble. (note from the author: *these are now called: precursors*).

Inspection of the engine revealed six serious problems, including damage to bearings, balancers, faulty ignition and missing stop nuts in the engine mounts.

Tests showed that all these factors would serve to increase the stress levels in the propeller beyond what would normally be expected.

4.5.3 Solutions

1. The engine manufacturer altered the design of the balancers of this model engine.

2. The investigation brought to light enough additional knowledge concerning the potential hazard of accumulated engine malfunctioning, that, coupled with the known history of the No. 3 engine, the Safety Board found that these particular propeller blades (Hamilton-Standard 2C13) are marginal when used with this type of engine (Wright 74 9C18BD-1).

4.5.4 Lessons

1. Engine, flight control cables and flight crucial electrical wires should be protected from propeller blade failures or, redundantly located.

2. In service, frequent engine malfunctions should be reported and fixed before more serious problems occur.

4.6 Design for One-way Fit

4.6.1 Problem

In March of 1954 a Continental Air Lines Convair 340 (Figure 4.5) experienced serious control difficulties after take-off and made a wheels-up emergency landing.

Figure 4.5 Convair 340 (Not accident aircraft, Courtesy G. Helmer)

There were no fatalities but two of eight passengers were seriously injured while the crew and several other passengers received minor injuries. The aircraft was severely damaged.

4.6.2 Cause

Ref. 4.5 states that the probable cause of this accident was: "the loss of control due to a failure of the right elevator trim tab push-pull rod caused by a reversed installation of the right elevator trim tab idler as a result of the carrier's reliance on the Manufacturers Illustrated Parts Catalog as a maintenance reference."

Right after take-off the crew noted a slight vibration but attributed it to the spinning main gears. The captain applied brakes during retraction but the vibration only increased in severity. At about 75 feet above the ground the vibration suddenly stopped and the airplane assumed a nose-down attitude. It took both pilots, applying their maximum strength, to keep the airplane from hitting the ground. Nose-up trim was applied to no avail. The captain decided to make a straight ahead wheels-up landing which was successfully executed.

Examination of the right horizontal stabilizer and elevator revealed no external damage. However, it was noted that the right elevator trim-servo tab was jammed in a 24-degree up or aircraft nose-down position. Over-travel marks and notching were found on the leading edge skin of the tab at its hinge points. Opening of the lower surface access door disclosed that the forward push-pull rod, which normally extends from the jack assembly to the elevator hinge-line idler, had failed. The failure occurred adjacent to the rear rod-end fitting. The free stub end attached to the idler was wedged against the bottom edge of the elevator spar cutout hole in such a manner as to hold the trim tab rigidly in a full-up (i.e. aircraft nose-down) position. A comparison of the assembly, as installed, with the appropriate Convair drawing disclosed both the idler and the forward push-pull rod were installed in reverse. Interference between the idler and push-pull rod was caused by the reversed idler.

Continental Airlines maintenance records showed the right elevator trim tab assembly had been removed, re-installed and inspected by company maintenance personnel. This work was done about 15 flight hours prior to the accident, for purpose of removing excessive play from the assembly. In the process of reassembly and re-installation both the Company Convair Maintenance Manual and the Manufacturers Illustrated Parts Catalog were used as references.

A figure in the Maintenance Manual first referred to shows the idler as a straight designed component whereas the actual part is curved, and depicted the forward and rear push-pull rods incorrectly in their inboard and outboard relationship.

To determine which way the curved idler had to be installed reference was made to the Manufacturers Illustrated Parts Catalog. This reference correctly showed an exploded view of the complete left-hand elevator trim tab idler assembly including its left idler. Since the right idler for the right elevator trim tab assembly was of different design than the left, it appeared alone and below the left assembly but on the same plate. It was shown curved which correctly depicted the actual design. Thus, for the right-hand assembly it was necessary to substitute the right idler in place of the left. It was stated by the company that by conventional interpretation of this illustration the left assembly would be correctly installed. However, upon substituting the right idler as required for the right assembly and following the same conventional interpretation, the result would be, and was, a reversed idler installation.

Upon completion of the installation the mechanic told the inspector how he had installed the idler. The inspector, using the same references agreed with the mechanic's interpretation and approved the work. The assembly was functionally tested in accordance with prescribed procedures and the results were normal. Subsequent tests revealed that the normal indications would be obtained with the idler in reverse. Had the check procedure required the trim tab be moved through its travel with the elevator full-up, an interference would have been noted.

4.6.3 Solutions

1. An accelerated inspection was conducted of all Convair 340 airplanes. Four aircraft were found to be in service with the idlers installed in a reversed condition. Two of these were alleged to come straight from the manufacturer. One other aircraft was found with a forward push-pull rod bent evidencing a reversed idler installation some time prior to the inspection. The total flight time on these aircraft varied from 1,600 hours to 3,000 hours.

2. The CAA was encouraged to re-enforce CAR 18.30 which reads:

"Standard of performance: general. All maintenance, repairs, and alterations shall be accomplished in accordance with methods, techniques, and practices approved by or acceptable to the administrator."

4.6.4 Lessons

1. Design all flight crucial components for a one-way fit. Murphy's Law in this case states that if a component can be installed in a reverse manner, it will be.

2. After maintenance and repair of flight crucial components it should be recommended that the system be moved through its operational range to make sure there is no interference.

4.7 Ailerons Reversed II

4.7.1 Problem

In June of 1953 a Western Air Lines Douglas DC-3A (Figure 4.6) crashed shortly after take-off while on a routine test flight following a major overhaul. The two flight crew members were injured while a company inspector on board was killed.

Figure 4.6 Douglas DC-3A (Not accident aircraft, Courtesy Mel Lawrence)

4.7.2 Cause

Ref. 4.6 states the probable cause to be: "reversed installation of aileron control cables and pulleys, and failure of the inspection department to detect this mistake."

Examination of the control system revealed that the aileron control cable within the control column housing had been reversed. Specifically, the replacement pulleys, one aluminum and one micarta, located at the elbow of both control columns, had been transposed during assembly.

Apparently the error resulted from the mechanic assuming that the diagram in the Overhaul Manual showed the captain's side looking forward. Although this diagram was ambiguous in that it did not illustrate graphically which wheel was depicted or the direction from which it was viewed, instructions applicable to the diagram indicate that it referred to the copilot's wheel looking aft. The result was a reversed motion of the ailerons. The mechanic, unaware of his mistake signed the work as satisfactorily completed.

Both control columns were installed in the airplane and the inspector (the same person killed in the accident) signed off the Plane Overhaul record indicating that he was satisfied with the work. Finally the full travel of the wheel controls against the surface controls was checked. All controls moved freely and with full travel but the improper direction of aileron motion was not noticed.

When the pilot who was to perform the test flight boarded the airplane a controls "free" check was made but the direction of motion was not noted.

4.7.3 Solution

The Safety Board concluded that the company's maintenance procedures should have been more explicit and that the proper direction of the controls should have been part of the final checks. Western Air Lines included the appropriate changes in their DC-3 Overhaul Manual.

4.7.4 Lessons

1. Design all flight crucial components for a one-way fit. Murphy's Law in this case states that if a component can be installed in a reverse manner, it will be.

2. Verify the correct operation of flight controls after maintenance or overhaul of the flight control system.

These lessons are apparently hard to learn. In 2006 the prototype Spectrum VLJ crashed right after take-off. The cause was: roll control cables were improperly installed.

4.8 Propeller Blade Severs Controls II

4.8.1 Problem

In August of 1957 an American Airlines Douglas DC-6A (Figure 4.7) experienced a blade failure of the No. 3 propeller during a take-off roll.

Figure 4.7 Model of Douglas DC-6A (Courtesy geminijets.com)

The failed blade struck the fuselage, severing many control cables, hydraulic lines, and electrical conduits, causing almost complete loss of mechanical and directional control. The take-off was aborted. The aircraft received substantial damage but there were no injuries to the crew. The airplane was on a cargo flight.

4.8.2 Cause

Ref. 4.7 indicates that the probable cause of this accident was: “the failure of a propeller blade precipitated by cold bending.”

A substantial portion of the broken propeller blade entered and passed through the lower part of the fuselage from right to left, severing 38 control cables and making 74 others inoperative. The broken blade then struck a blade tip of the No. 2 propeller and the propeller dome, breaking it and causing oil to be released from the dome.

The severing or otherwise damaging of the control cables made all throttle and mixture controls inoperative, as well as all main and auxiliary fuel selectors. Firewall shut-off cables to engines Nos. 1 and 2 were cut through. Ignition switches of Nos. 1 and 2 engines were also made inoperative. Most of the electrical instrument and warning circuits to engines Nos. 1 and 2 were severed. The hydraulic and emergency brakes could not function because of damaged hydraulic and air lines.

A detailed analysis of the reason for the propeller blade failure was made. The No. 3 propeller blade exhibited fatigue markings. It was established that residual stresses in the area of the fracture had been disturbed by cold bending during manufacturing.

4.8.3 Solution

The Safety Board concluded that inadequate precautions were being taken to protect against blade failures due to disturbance of residual stresses by cold bending loads.

4.8.4 Lessons

1. Ref. 4.7 does not anywhere question the design integrity of the engine control systems. Imagine what would have happened if this had been a passenger flight and if the blade failure had occurred after the airplane became airborne. One failure would then probably have caused serious loss of life.

2. The certification standards for the engine control systems used in the design of this airplane were marginal at best.

4.9 Elevator Boost System Bolt Backs Out

4.9.1 Problem

In September of 1961 a Trans World Airlines Lockheed L-049 Constellation (Figure 4.8) crashed about nine miles west of Midway Airport, Chicago, Illinois.

Figure 4.8 Lockheed L-049 Constellation (Not accident aircraft, Courtesy www.prop-liners.com and Trans World Airlines)

While climbing on its intended course, five minutes into the flight, the airplane experienced loss of longitudinal control and crashed. All 78 persons on board were killed. The airplane was completely destroyed.

4.9.2 Cause

Ref. 4.8 determines that the probable cause of this accident was: "the loss of an AN-175-21 nickel steel bolt from the parallelogram linkage of the elevator boost system, resulting in loss of control of the aircraft."

The airplane is equipped with a shift mechanism to allow the crew to have mechanical control over the elevator in case of a hydraulic system failure of the elevator boost system. It was found that to operate this shift system in the presence of large forces on the control column is extremely difficult. Because of the backed out bolt in the parallelogram linkage the elevator would have moved upward to which a pilot would normally react with a push force on the control column. Under that scenario shifting to mechanical control would become questionable.

It was clear from the wreckage that the horizontal tail had separated from the airplane in flight. This occurred due to the elevator being deflected about 40 degrees trailing edge up. That the elevator was deflected to 40 degrees up at some point during the empennage failure was clear from the fact that the right outboard closing rib was crushed by, and left its impression on, the right fin and rudder.

Following several accidents involving Navy R7V and Air Force C-121 aircraft (military versions of the same airplane) the Air Force conducted an analysis of the Constellation elevator system. Several tests indicated that the shift operation could not always be accomplished when 100 lbs or more of stick force was applied.

The Safety Board believed that the design philosophy of the emergency shift system was questionable.

The most probable reason that the bolt backed out (although this could not be positively established) is that a cotter key was omitted at the time of the last parallelogram installation and that the bolt had gradually backed out. Despite a detailed search of the area (including sifting dirt at the accident site) the bolt could not be located.

4.9.3 Solution

The FAA had the Constellation Flight Manual amended to include “procedures for turning off the elevator boost with an uncontrollable elevator.” It was felt that to require a design change of the flight control system was not warranted in view of the “excellent service history of this airplane since certification in 1946.”

4.9.4 Lessons

1. One maintenance error resulted in a fatal crash (three, counting the military experiences with the same problem). Design engineers should consider this in the design of flight crucial systems.

2. Also, if a mechanical back-up is designed, switching to it should be possible even when, in the case of an emergency, cockpit control forces are applied to the system.

4.10 Elevator Control Forces to Overcome Electric Trim Tab Failure Become too High

4.10.1 Problem

In September of 1962 a Lockheed L-18 Lodestar (Figure 4.9), operated by Ashland Oil & Refining Co., lost its right wing in flight and crashed and burned in a field near Lake Milton, Ohio. The pilot, co-pilot and 11 passengers were killed.

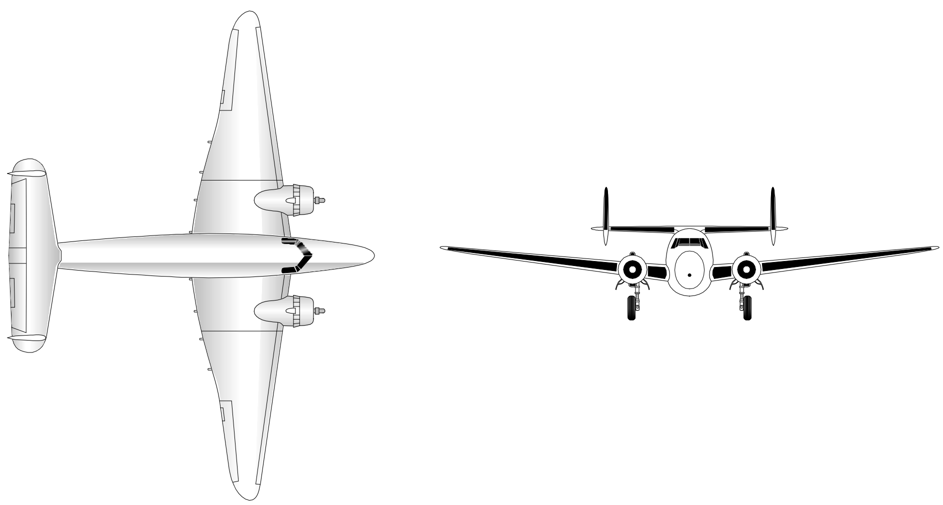


Figure 4.9 Lockheed L-18 Lodestar

4.10.2 Cause

In Ref. 4.9 the probable cause is stated as the: "malfunction of the electric trim tab unit which resulted in aircraft uncontrollability and subsequent failure of the wing."

The elevator trim drive and both elevator trim cable drums were found in a position that corresponds to a full aircraft nose-down setting. The failed wing panel was found to have been subjected to a large negative air load.

According to Ref. 4.9 the failure of the right wing panel and the full nose-down trim indication appeared to be interrelated.

The electric longitudinal trim system installed in this airplane would require a large control column force to override unwanted elevator trim. The amount of force would be approximately 30 lbs per degree of aircraft nose-down trim, at aft c.g. and 170 kts.
The rate at which the elevator tab moves, if unchecked, is approximately 5 degrees per second, requiring about 5.5 seconds from the neutral position to its full aircraft nose-down setting of 25 degrees.

Relating control column force to time, a pilot would be required to exert a force of approximately 150 lbs to override the first second of unwanted nose-down trim. At the end of two seconds the override force required would be about double that.

If, prior to this accident, longitudinal trim was applied in a nose-down direction (intentionally or inadvertently) and a run-away trim condition developed, the pilot's natural reaction would probably be to resist the nose heaviness with a pull force on the control column or by reversing trim or both. The time involved in any corrective action would have to be about one second; in this time interval the control forces would have reached the limit of human capability.

The time from unwanted trim initiation to catastrophic failure of the right wing was probably a matter of a few seconds.

4.10.3 Solution

Following a recommendation of the Safety Board to the FAA administrator an order for immediate de-activation of Spartan electric longitudinal trim systems in L-18 aircraft was issued. This order was followed by a modification of Supplemental Type Certificate (STC) SA2-183 which now requires a driving motor rated at 0.12 hp at 4,000 rpm as opposed to the original 0.17 hp motor at 8,500 rpm. In addition, this modification limits the travel of the elevator trim tab to between 5 degrees nose-down and 10 degrees nose-up. All L-18 airplanes with the Spartan electric longitudinal trim system, must comply with the provisions of this modification.

4.10.4 Lessons

1. What happened in this accident was predictable and should have been prevented by proper design. Calculations of control column forces were well known at that time. Limits of human capability were also well known. It is hard to understand how this STC could have been granted without such an analysis.

2. Whichever regulations were applicable in the STC SA2-183 era, it is not ethical to pursue an STC which has this (entirely predictable) consequence.

3. This case represents a serious breakdown of the DER system.

4.11 Pitch Trim Failure Reverses Elevator Stick-force-speed-gradient

4.11.1 Problem

In February of 1964 an Eastern Air Lines Douglas DC-8 (Figure 4.10) crashed in Lake Pontchartrain, Louisiana. All 51 passengers and the crew of 7 were fatally injured.

Figure 4.10 Model of Douglas DC-8 (Courtesy geminijets.com)

4.11.2 Cause

Ref. 4.10 states that the probable cause of this accident was the degradation of aircraft stability characteristics in turbulence, because of abnormal longitudinal trim component positions.

To understand how this accident came about it will be necessary to describe several aspects of the longitudinal flight control system of the DC-8. Figure 4.11 shows a schematic of that system.

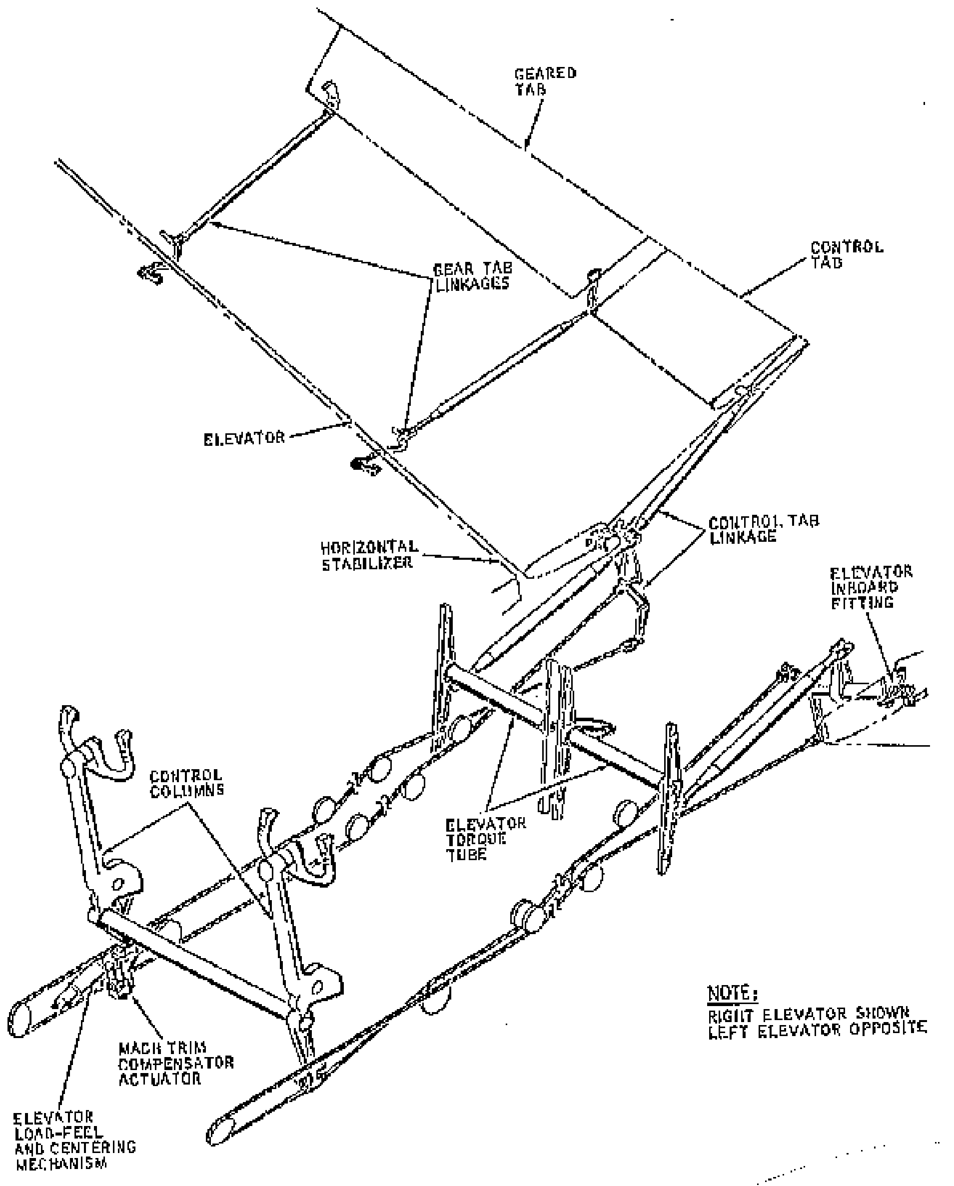


Figure 4.11 Elevator Control Schematic (Courtesy of NTSB)

The following material has been taken/adapted from Ref. 4.10.

The airplane can be controlled longitudinally with elevators or a variable incidence horizontal stabilizer. The elevators are operated by movement of either (left or right) control column through two independent cable systems to elevator control tabs (servo tabs). The elevators are connected together by a torque tube at the rear spar of the horizontal stabilizer. The friction bandwidth in this system is between 5 and 6 lbs. Separate tabs (geared tabs) on the elevator trailing edge provide aerodynamic boost to pilot control inputs.
Most of the pilot's stick-force (actually column force in this case) is provided by a load feel mechanism with two opposing springs which establish a neutral point.

In high speed flight there is a tendency toward an increasing nose-down pitching moment. This is mainly due to an aft shift of the aerodynamic center of the wing. This characteristic is quite common to swept-aft wing airplanes operating in the high Mach number range. In the DC-8 this tendency is counteracted by the Pitch Trim Compensator (PTC) system. Operation of the PTC is also required in the low altitude, high dynamic pressure regime to improve the stick-force-speed gradient of the airplane.

The PTC consists of an analog computer, an electrical actuator, spring loaded linkages, and a mechanical indicator. The computer senses Mach effects at high altitude and dynamic pressure at low altitudes (below 20,000 ft). The computer provides electrical signals to the actuator which actually moves the co-pilot's control column. The actuation begins at M=0.7 (or 310 kias) and increases in displacement and rate up to M=0.88 (or 410 kias). The maximum input is 36 lbs of stick-force. Actuation of the PTC is indicated to the flight crew by the extension of a plunger from a flexible cable housing attached to the left side of the co-pilot's control column. There is no measurable correlation between the amount of indicator showing and the degree of PTC actuator extension. A three-position switch located on the left side of the control pedestal permits normal operation, testing of the system in the spring-loaded test position, and an override position which may be used to retract the actuator in the event of a malfunction.

Figure 4.12 shows schematics illustrating the operation of the feel mechanism (elevator load feel spring) and the PTC trim actuator. The diagrams do not show the workings of the PTC indicator system.

Figure 4.13 shows the PTC actuator control laws with Mach number and with dynamic pressure (indicated airspeed).

Figure 4.14 shows the effect of the PTC on the stick-force-speed gradient with Mach number only (above 20,000 ft). Note the strong effect of the PTC on the stick-force-speed gradient above M=0.7 and particularly above M=0.8. Without the PTC in operation the airplane would exhibit an unstable stick-force-speed gradient which is not acceptable according to flying quality regulations.

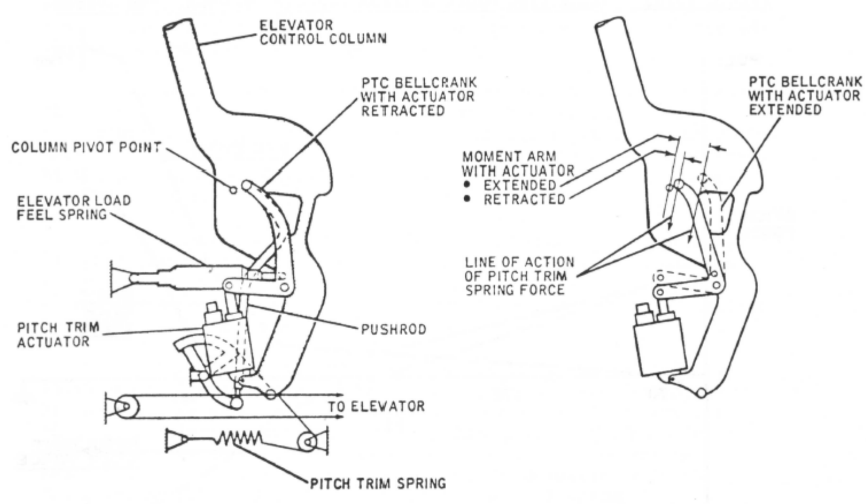


Figure 4.12 PTC Operating Schematic (from DC-8 Flight Study Guide)

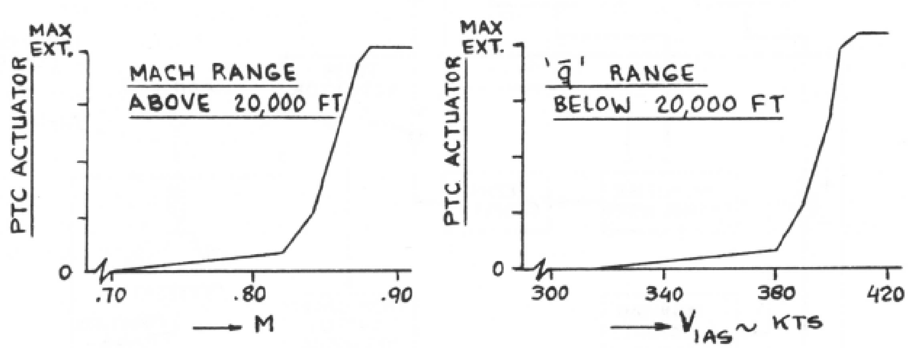


Figure 4.13 PTC Actuator Control Laws with Mach Number and with Indicated Airspeed (from DC-8 Flight Study Guide)

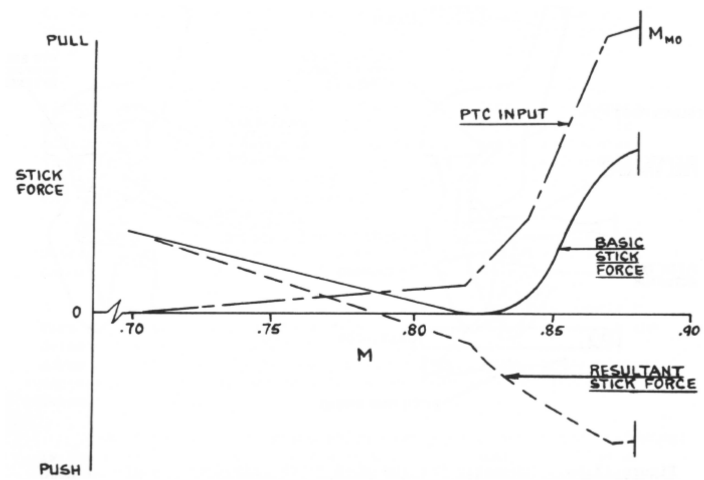


Figure 4.14 Effect of PTC on Stick-Force-Speed Behavior for Altitudes above 20,000 ft (from DC-8 Flight Study Guide)

Therefore, with the PTC inoperative the airplane may not be operated at Mach numbers above 0.8. A similar situation prevails (but not shown here) with regard to dynamic pressure at altitudes below 20,000 ft. The airplane then may not be operated at indicated air speeds above 310 kias.

Longitudinal trimming in the DC-8 is accomplished by hydraulic or electrical actuation of the horizontal stabilizer. The hydraulic motor trims at a rate of 0.5 deg/sec through a range of -10 degrees (trailing edge up or Aircraft Nose Up (ANU)) to +2 degrees (trailing edge down or Aircraft Nose Down (AND)).

The hydraulic trim motor is actuated by manipulation of dual toggle switches on either control column, or by split “suitcase” handles mounted side by side on the center console.

The electric trim motor trims at a rate of 1/17 deg/sec, and is actuated by dual toggle switches on the center console, or by the autopilot.

Both motors provide power through differential gearing to a drive shaft on which a dual sprocket assembly is mounted. The sprockets are connected to the common drive shaft by shear rivets, and each transmits rotation of the drive unit through roller chains to an irreversible jackscrew. Failure of either set of shear rivets freezes the stabilizer in the last position, and further operation is impossible. The indication of stabilizer position is provided by fore and aft movement of a small “bug” along a scale on the left side of the center console.

Longitudinal control of the airplane may also be accomplished through the autopilot which utilizes elevator displacement to initially retain the selected pitch attitude. An automatic trim coupler senses elevator torque information and generates stabilizer trim commands when torque of a given value or time interval is encountered.

A "runaway" or contradiction in the system results in the interruption of power to the autopilot and the illumination of a warning light.

Pitch attitude information on this particular DC-8 was provided by a Collins 105 Approach Horizon through movement of the "miniature airplane" in reference to an all-black face of the instrument. There are no indices for the degree of pitch attitude, and the displayed rate of pitch change varies as follows:

Attitude Range	Display Ratio
0-20 degrees	0.033 inch/deg change
20-70 degrees	0.012 inch/deg change
70-85 degrees	0.006 inch/deg change

It is therefore possible for the instrument to indicate a reduced rate of pitch when attitude changes through 20 degrees of pitch, even though the actual rate of pitch is constant. In a corresponding manner if the attitude has exceeded 20 degrees, the displayed rate of aircraft response to control inputs will be slower than the actual response.

So far the description of the flight control system. Now back to the accident.

The airplane was being dispatched with an inoperative PTC (known to the flight crew) and for a reduced cruise speed. There was moderate to severe turbulence during the climb-out and the flight was conducted in Instrument Meteorological Conditions (IMC).

A review of the aircraft records showed that the PTC had been replaced eight times, four during the last week of operation. Seven days before the accident flight the PTC was reported as operative although the indicator failed to show extension.

The PTC computer installed at the time of the accident, had been removed from various aircraft 15 times since 1960. Following the accident it was found that functional tests by Eastern Air Lines and other DC-8 operators could not even detect certain computer malfunctions.

The flight maintenance log also revealed eleven autopilot malfunctions in the last 30 days of operation. Two discrepancies involved yaw, six referred to longitudinal control problems, and three reported automatic disconnects.

From the wreckage it was clear that the stabilizer jackscrews were within one turn of the full AND (Airplane Nose Down) trim setting.

In flight testing it was revealed that the DC-8 when trimmed at 300 kts in an aft c.g. climb configuration, with maximum continuous thrust, and inoperative PTC, has essentially zero stick-force-speed gradient. (Author's note: this is contrary to what is required in the regulations)

During another flight test conducted by an FAA test pilot it was discovered that, with the c.g. at 24% (about 1% forward of the calculated c.g. location of the accident aircraft): "..…during maneuvering with a fully extended PTC at a velocity of approximately 220 kts and the airplane trimmed to its previous extreme of full AND (+2 degrees). It was observed that any attempt at maneuvering the airplane with the elevator system resulted in sharp reversals in the airplane's maneuvering stability."

Another test pilot reported that flight testing of the DC-8 handling characteristics under abnormal conditions, i.e. PTC extended to offset a 0.5 AND stabilizer setting, in a cruise configuration at 220 kts revealed that: "….the aircraft exhibited no stick force stability. This lack of stick force stability is caused by a shifting of the stick neutral position to a very flat portion of the load feel spring when PTC is extended. The low gradient of the load feel spring in this area is masked by the control system friction which necessitates flying the aircraft by stick position only. The aircraft is neutrally stable at small airspeed increments about the trim point in any normal attitude, including 45 degrees turning flight, and would maintain a 45-degree coordinated turn hands off until the speed was changed. With a change of 10 kts, the aircraft exhibited classic instability and would continue to increase or decrease, whichever the case may be, until restrained."

It is now well known that airplane response rate to elevator deflection has a profound effect on the behavior of the aircraft from a pilot's viewpoint. Couple this with flying in IMC, an inaccurate pitch attitude indication system, an inoperative PTC and a failed stabilizer drive (almost AND) and it is very likely that a pilot would lose control over the airplane.

The Safety Board also came to the conclusion that a support bushing in the stabilizer drive unit had been installed in a reversed manner. This would allow the drive unit to eventually fail which it did during the accident flight.

4.11.3 Solution (Sic!)

Ref. 4.10 does not contain specific recommendations for change despite the fact that the report includes the following findings (page 57 of Ref. 4.10):

1. The attitude indicator, which was small with a solid background, was difficult to interpret at night.
2. The pitch indication of the attitude indicator was "geared down" but not indexed as to degrees.
3. The aircraft exhibited marginal to non-existent speed stability and a stick-force-per-g characteristic which test pilots have interpreted as unstable.

4.11.4 Lessons

1. Airplane flying qualities must be benign under VMC and IMC conditions with any part of the flight control system inoperative if it is legal to dispatch the airplane in that manner.

2. Flight tests must be conducted before certification is granted, under conditions which, realistically can be expected to occur, with any part of the flight control system inoperative if it is legal to dispatch the airplane in that manner.

3. It appears that there were many precursors to this tragic event. The "safety oversight system" must do a better job of heeding the warnings which emanate from such precursors.

4. It is hard to understand how this airplane was certified in view of item 3 in Section 4.11.3.

4.12 Reversing Polarity in a Pitch Damper

4.12.1 Problem

During May of 1964 a Lockheed C-141 (Figure 4.15), military transport was climbing out of McCord Field, near Tacoma, Washington.

Shortly after retracting the flaps the pilot engaged the pitch-damper. This is part of configuring the airplane for climb-to-cruise operations. The airplane immediately began a series of diverging short period oscillations. After several oscillations, the g-level was building up to worrisome levels and the crew wondered what was going on. Luckily, at that point the pilot remembered what someone had taught him: If you ever get into trouble in an airplane, try undoing the last thing you did. The pilot disengaged the pitch damper and the oscillations subsided. The airplane was landed without further problems.

Figure 4.15 Lockheed C-141, Starlifter (Not accident aircraft, Courtesy Lockheed-Martin)

4.12.2 Cause

It turned out that during a previous flight the pitch damper had malfunctioned. The diagnosis had been a faulty rate-gyro. During the overnight repair a new rate gyro had been installed. However, the mechanic carrying out the repair had accidentally reversed the wiring to the gyro. As a result, the sign of the signals in the pitch-rate-to-elevator feedback loop was reversed, and the damper was driving the airplane unstable. For a detailed discussion of why this is so, see Ref. 4.11, Chapter 11.

4.12.3 Solution

Wire bundles for critical items in the flight control system should be designed for a one-way fit. Figure 4.16 suggests two ways of accomplishing this.

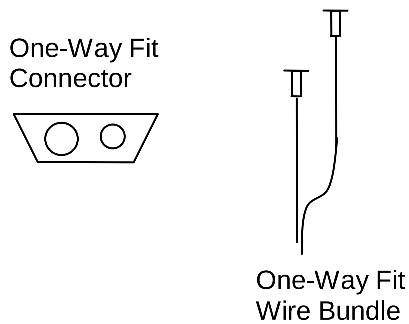


Figure 4.16 Suggested Diagram for One-way Fit of Critical Wiring

4.12.4 Lesson

Murphy's Law strikes again and again! Prevent miswiring of flight crucial systems by designing for a one-way fit.

4.13 Take-off with Locked Elevator I

4.13.1 Problem

In December of 1967 a Frontier Airlines Douglas DC-3 cargo airplane (Figure 4.17) crashed during take-off from Stapleton International Airport in Colorado. Both flight crew members were killed.

Figure 4.17 Douglas DC-3 (Not accident aircraft or airline, Courtesy H. Chaloner)

4.13.2 Cause

An investigation revealed that the take-off had been made with an external gust lock in place on the right elevator. The NTSB established that the probable cause was the failure of the crew to perform a pre-take-off control check resulting in take-off with a fixed elevator.

4.13.3 Solution and Lesson

This has been a frequently occurring problem with DC-3 and similar aircraft. One solution which designers should keep in mind is to apply the gust locks to the cockpit controls and make them very obvious.

4.14 Elastic Stop-nuts in Flight Control Systems

4.14.1 Problem

In April of 1970 a Cessna 340 (Figure 4.18) experimental airplane was observed by the chase plane pilot to pitch down, roll inverted and enter a flat spiral from which it did not recover. The test pilot did not survive.

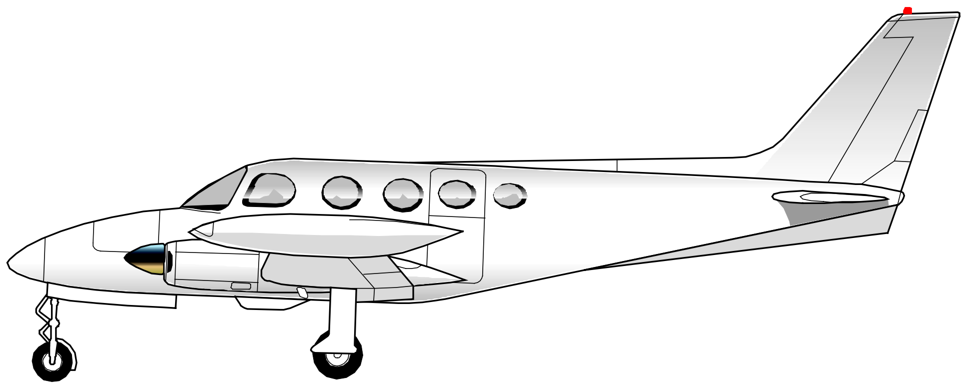


Figure 4.18 Production Version of the Cessna 340

4.14.2 Cause

The cause was found to be an elastic stop-nut in the elevator tab, which had loosened up to the point where the bolt holding the tab-actuating-rod in place had come off. That allowed the tab to oscillate a few times after which it jammed in the tab up direction. In turn this required a control column pull force of more than 300 lbs, which the pilot was not able to generate.

4.14.3 Solution

The elastic stop-nut was replaced with a castellated nut that was wired to the bolt to prevent rotation of the nut.

4.14.4 Lesson

Elastic stop-nuts should not be used in primary or secondary flight control systems. All nuts should be of such a type that they can be wired to their bolts to prevent rotation. Castellated nuts are a preferred way of achieving that.

4.15 Controls Jammed by Foreign Object

4.15.1 Problem

In September of 1970 a Trans International Airlines Douglas DC-8-63F (Figure 4.19) crashed on take-off from the Kennedy International Airport in New York, NY.

Figure 4.19 Model of Douglas DC-8-63F (Courtesy geminijets.com)

The aircraft was observed to be rotating to an excessively nose-high attitude. The airplane did become airborne but continued to rotate nose-up to more than 60 degrees. Then it rolled about 20 degrees to the right, reversed and rolled 90 degrees to the left and crashed in that attitude. All eleven crew members on board were killed.

4.15.2 Cause

In Ref. 4.11 the NTSB determines that the probable cause of this accident was: "a loss of pitch control caused by the entrapment of a pointed, asphalt-covered object between the leading edge of the right elevator and the horizontal spar web access door in the aft part of the stabilizer. The restriction to elevator movement, caused by a highly unusual and unknown condition, was not detected by the crew in time to reject the take-off successfully. However, an apparent lack of crew responsiveness to a highly unusual situation, coupled with the captain's failure to monitor adequately the take-off, contributed to the failure to reject the take-off."

To understand why this happened it is useful to revisit the longitudinal flight control system in the DC-8 series of airplanes. Figure 4.11 (in Section 4.10) shows a schematic of this system.

The system is a reversible, mechanical system. The pilot directly controls the elevator control tabs (one on the inboard side of each elevator). When in flight, whenever the pilot pulls on the cockpit control column, the tabs are deflected trailing edge down. This produces a hinge-moment about the elevator hinge-line which drives the elevator trailing edge up. The elevator is assisted in the motion by geared tabs located on the outboard side of each elevator.

A well known problem with this type of control system is that when the pilot moves the cockpit controls when parked on the ground, only the control tabs move, the elevators do not. In most airplanes with reversible controls a standard check before taxiing is to check the cockpit controls for freedom of movement. By implication, if the cockpit controls are free to move, the elevator will follow suit. Not so in the DC-8.

Therefore, a standard check in the DC-8 is to slightly activate the longitudinal controls during the take-off run (but below the rotation speed of the airplane) to verify that the airplane does respond. If the airplane does not respond to a longitudinal control input then the take-off must be aborted and since this is done below the rotation speed it is considered safe. Rejecting a take-off at higher speeds can be potentially hazardous.
It can now be understood that with the elevator blocked by some mechanical cause (such as a foreign object) the cockpit controls will still move freely.

The investigation showed that in the accident airplane the elevator was deflected trailing edge up by about 11 degrees when the airplane began the take-off roll with the airplane beginning to rotate at 80 kts. Once airborne with the elevator jammed in this position there was not adequate pitch control to correct the attitude of the airplane.

Apparently, during this take-off the lack of airplane responsiveness was not detected by the crew and the take-off was not aborted.

It is noted that in the original DC-8 series of airplanes there was no elevator position indicator in the cockpit. Such a system was added later but not in all airplanes.

4.15.3 Solution

In Ref. 4.11 the NTSB makes four recommendations to the FAA:

All DC-8 operators should be advised of the hazardous condition that can be created by foreign objects jamming the elevators.

All DC-8 operators should be advised that take-offs should be rejected when premature or unacceptable rotation occurs during take-off until adequate procedures are developed for a positive check of elevator position.

The DC-8 flight control system should be evaluated by the FAA with a view to establishing a standard procedure for checking the system from the cockpit.

Consideration should be given for a requirement to install an elevator position indicator in the cockpit of all DC-8 aircraft.

4.15.4 Lesson

Designers of flight control systems could have easily predicted that an accident of this type would, at some point, happen. It does seem unreasonable to certify an airplane with a flight control system that cannot be easily checked for functionality before take-off.

4.16 Rudder Fitting Failure

4.16.1 Problem

In March of 1971 a Boeing 720-047B (Figure 4.20) of Western Air Lines crashed on the Ontario International Airport in Ontario, CA while carrying out a simulated engine-out missed approach procedure. All five crew members were killed.

Figure 4.20 Boeing 720 (Not accident aircraft, Courtesy www.al-airliners.be and Western Airlines)

4.16.2 Cause

In Ref. 4.13 the NTSB determined that the probable cause was: “failure of the aircraft rudder hydraulic actuator support fitting. The failure of the fitting resulted in the apparent loss of left rudder control which, under the conditions of this flight, precluded the pilot’s ability to maintain directional control during a simulated engine-out missed-approach.”

One of the fitting lugs failed due to a combination of stress corrosion cracking and high tensile loading. The corrosion was initiated at the bushing in the fitting lug because of the use of dissimilar materials.

The detailed investigation reported in Ref. 4.13 found that “the ultimate strength of an intact actuator support fitting was approximately 100,000 pounds. With a single actuator attachment lug failure on a fitting, the remaining lug would sustain a tensile load of approximately 18,500 pounds. With a fully pressurized rudder hydraulic system of 3,000 p.s.i., maximum left rudder deflection (25 degrees) exerted a maximum in-flight tensile load of approximately 26,300 pounds on the support fitting. Under the asymmetrical thrust conditions established in N3166, with at least 23˚ left rudder deflection, nearly the full 23,600 pound tensile load was applied to the support fitting.”

There was a well-documented history of rudder support fitting cracking since 1967. Several of these cases had been discovered on KC-135 aircraft and reported by USAF to Boeing. Fleet inspection showed several aircraft with cracked fittings and these were restricted to the use of mechanical rudder control only until the problems were fixed.

Boeing recommended inspection of commercial 707 and 720 aircraft but no problems were found in 1967.

Boeing did issue a Service Bulletin (SB 2903) which included the following description of the fitting problem:

“… five operators have reported cracking of the upper, lower, or both lugs of the rudder actuator support fitting on five airplanes with 7,000 to 26,000 flight hours. Complete failure occurred through the actuator bolt hole and the actuator became separated from the rudder in two instances, resulting in loss of rudder hydraulic control. Uneventful landings were made in both instances. Fitting failure is attributed to cracks caused by stress corrosion which started at the bushing.”

In a revision to SB 2903, dated June 4, 1969, it was stated that one operator had discovered five fittings with cracks after inspecting a large portion of his fleet.

Boeing records contained a history of four complete failures of both lugs on 707 aircraft prior to March, 1971: three foreign carriers and one domestic carrier had experienced these failures during training maneuvers using asymmetric thrust. These incidents occurred in October of 1967, May of 1969, December of 1970 and March of 1971, all prior to the date of the N3166 accident.

The NTSB found that Boeing, the FAA and Western Air Lines did not emphasize the potential operational hazards associated with in-flight failures of the support fitting.

4.16.3 Solution

The NTSB made a number of recommendations which resulted in the FAA issuing several Airworthiness Directives and Operational Alert Notices. Among these were:

1. AD 71-9-2, effective April 27, 1971, requiring more frequent inspections of the support fitting with the lug bushings removed. This directive also required replacement of all 7079-T6 fittings with 7075-T73 fittings within the next 5,400 flight hours but no later than October 1, 1972.

2. Operational Alert Notice No, 8430 informed all 707/720 operators of the support fitting failures and advised that simulated engine failures at low altitudes not be performed until the requirements of AD 71-9-2 had been complied with.

4.16.4 Lessons

1. It would seem that the “airworthiness reporting and oversight system” was not reacting seriously to these serious problems. There were many precursors which were foretelling the events that led to the demise of N3166.
2. Design engineering departments should be kept informed about developments as outlined in this Section. Design engineers and their management should then initiate the “what-if” philosophy. Doing so would have predicted an N3166-like event which should have led to inspection, design and manufacturing action much sooner.

3. This is an example of where a single failure caused catastrophic results.

4. Finally using dissimilar materials on parts which are intact with each other, invites corrosion.

4.17 Locating Flight Control System Cables

4.17.1 Problem

In June of 1972 an American Airlines Douglas DC-10 (Figure 4.21) experienced a rapid cabin decompression due to a failure of the rear cargo door latching system which allowed the door to open in flight at 11,750 ft altitude.

The door separated from the fuselage causing substantial damage to the leading edge and the upper surface of the horizontal tail. All longitudinal and rudder controls and the center engine controls were partially jammed. The flight crew was able to land the airplane (at the Detroit airport) with great difficulty but there were no injuries.

Figure 4.21 Model of McDonnell-Douglas DC-10 (Courtesy geminijets.com)

About two years later, in March of 1974 another DC-10 (Turkish Airlines) experienced a similar event near Paris, France. This time the airplane became uncontrollable and crashed with 346 lives lost.

4.17.2 Cause

The NTSB investigation of the American Airlines DC-10 accident of June, 1972 is presented in Ref. 4.14. Additional information is presented in Ref. 4.15 (pp 159-162).

The investigation into the Turkish Airlines DC-10 accident is covered in some detail in Ref. 4.16, pp 127-144.

The investigations into both events showed the cause to be a partial “caving in” of the cabin floor under large differential pressure loads. When a cargo door leaves the airplane the pressurized air is evacuated from the cargo department very rapidly. The passenger compartment retains its cabin altitude pressure and, as a result, a large pressure load is exerted on the cabin floor for which it was not designed. Figure 4.22 shows a cross section of the fuselage indicating the approximate location of the elevator control cables.

Due to the deformation of the floor the control cables became partially jammed in the case of the American Airlines accident, making the airplane difficult to control. In the case of the Turkish Airlines accident the controls were severed and/or jammed and the airplane became impossible to control.

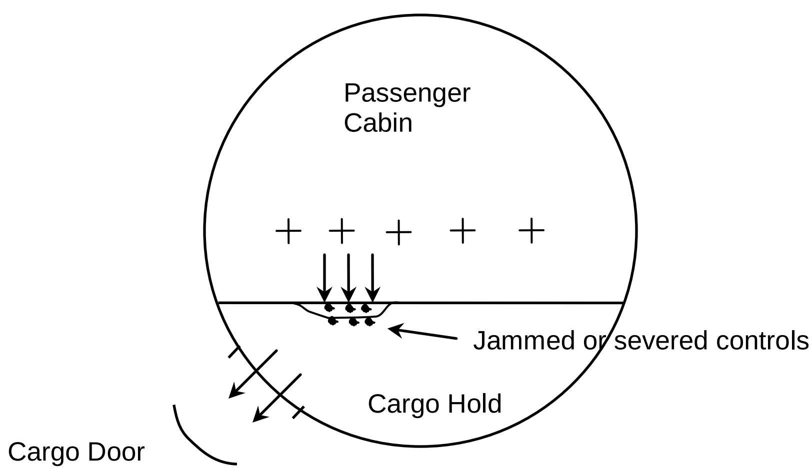


Figure 4.22 Fuselage Cross Section Indicating Location of Flight Control Cables

4.17.3 Solution

The NTSB made a number of recommendations to the FAA and most of those were implemented. These recommendations dealt with pressure ventilation devices, improved designs for cargo door latching and improved training of ground personnel.

The pressure ventilation devices that were installed in the rear cargo compartment work as follows: when the differential pressure between the passenger cabin and the cargo hold exceeds a certain value properly sized spring-loaded doors are opened to ventilate the pressure differential overboard.

The scenario leading to these failures is not only predictable but it was predicted, as discussed in Refs. 4.17 and 4.18, by engineers at Convair, Douglas and the RLD in The Netherlands.

It seems to the author that the proper design solution is to locate the control cables in such a manner that one event cannot jeopardize the entire system. This should have been done early in the design layout of the airplane. The DC-10 should never have been certified the way it was.

4.17.4 Lessons

1. A redundant system is not redundant when one single event can put the entire system out of commission. In this case the mechanical signal paths to the redundant hydraulic actuators of the flight controls were themselves not redundant. Therefore the system is not really redundant.

2. Even with the pressure ventilation system installed the system is still not redundant. Some other cause (for example a minor explosion in the control cable vicinity) can still render the airplane uncontrollable.

3. These DC-10 events bring up the important point of “precursors” first mentioned in the introduction to this book. Once a cargo door comes loose, subsequent events like it are bound to happen. Certifying agencies and their DER’s should keep this in mind.

4.18 Pilot Induced Oscillations

4.18.1 Problem

In August of 1971 a Trans World Airlines Boeing 707-331B (Figure 4.23) porpoised while descending toward the Los Angeles International Airport (LAX).

Figure 4.23 Model of Boeing 707-331B (Courtesy geminijets.com)

The oscillations persisted for about two minutes during which more than 50 oscillations occurred. Peak accelerations of +2.4g to -0.3g were measured at the airplane center of gravity. One passenger was fatally injured, one flight attendant and two other passengers were seriously injured. The airplane then made an uneventful landing at LAX.

4.18.2 Cause

In Ref. 4.19 the NTSB established as the probable cause: "a combination of design tolerances in the aircraft longitudinal control system which, under certain conditions, produced a critical relationship between cockpit control forces and aircraft response. The a-typical control force characteristics which were present in the control system of this particular aircraft were conducive to over-control by the pilot. The pilot's normal reaction to an unexpected longitudinal disturbance led to a pitching oscillation which was temporarily sustained by his subsequent application of control column forces to regain stable flight."

The problem first occurred when the crew disengaged the autopilot while at 33,000 ft to begin the descent into the LAX area. The airplane pitched up abruptly, then pitched down, and began an oscillatory, or porpoising, motion. The captain, assisted by the first officer, attempted to counteract the porpoising motion by inputs through the control column. The aircraft continued to descend while decelerating. The pitching oscillations abated and the aircraft regained stable flight at 19,500 ft and about 300 kias.

The reason for the initial pitch-up upon auto-pilot disconnect was that the airplane was slightly out of trim. The cause of the auto-pilot mistrim was found to be a burned contact on the trim control relay.

Another discrepancy which was found was that the elevator hinge line friction exceeded allowable values specified in the 707 maintenance manual.

A large number of flight tests were carried out on the accident aircraft and on other similar 707 aircraft by Boeing test pilots and by FAA and TWA pilots. The findings of these flight tests were essentially that the accident aircraft had a number of minor discrepancies in its longitudinal flight control system which together produced a lowering of the stick-force-speed gradients at higher elevator deflections. Contributing factors were:

- slight asymmetrical elevator deflections between the right and left elevators
- waviness of the left stabilizer skin panels under no airload conditions which probably caused different boundary layer behavior and therefore different elevator hinge-moment behavior
- under airloads the waviness of the left and right stabilizer were different by about 0.1 inches, again probably causing different boundary layer behavior and therefore different elevator hinge-moment behavior

4.18.3 Solution

Boeing suggested a number of corrective measures in rigging the flight controls to assure stiffer stick-force-speed gradients. Stiffer stabilizer skins were also adapted. The NTSB and the FAA essentially adopted these suggestions. No similar problems have been encountered since that time.

4.18.4 Lessons

Means to verify whether or not autopilots keep an airplane in trim should be part of each autopilot installation.

The observed oscillations correspond roughly to the short period frequency of this airplane which is about 2.6 rad/sec at mid center of gravity. The short period damping ratio of most jet transports ranges from about 0.35 at about 35,000 ft altitude to about 0.5 at about 20,000 ft altitude. The airplane will therefore appear to have relatively poor pitch damping at high altitude and much better damping at lower altitudes.

However, the airplane is self-damping at all altitudes. Therefore, the first thing a pilot should do when uncommanded oscillations begin, is do nothing.

Humans tend not to be good controllers of oscillatory motions of airplanes. Pilots therefore do have a tendency to re-enforce oscillations by trying to oppose them. This effect is known as PIO: pilot induced oscillations.

4.19 Reversing Polarity in a Yaw Damper

4.19.1 Problem

In 1973, the Learjet Model 36 (Figure 4.24) certification was being held up because the yaw damper (designed by a subcontractor) did not properly damp the Dutch roll of the airplane. This author was asked to look into this problem with a team of engineers.

After quite a bit of root-locus work it was discovered that by tilting the sensitive axis of the rate gyro aft, significant improvement in damping could be obtained. In other words, the original yaw damper had the gyro installed at the wrong angle.

Figure 4.24 Gates-Learjet Model 36 (Not subject aircraft, Courtesy Learjet)

Management was briefed on the proposed solution and the author was asked to draw up a bracket which allowed for aft tilt of the gyro and have it made by the experimental shop. The modification would then be flight tested the next morning, a Saturday. Figure 4.25 shows a sketch of that bracket. Notice the aft "tilt" of the rate gyro.

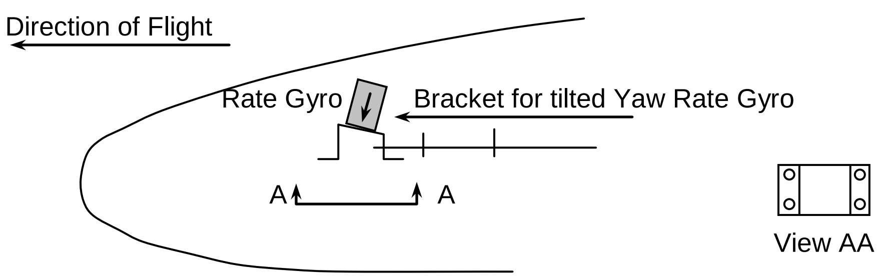


Figure 4.25 Experimental Bracket for Tilted Yaw Rate Gyro

I went home to Lawrence late Friday night, figuring that the problem had been solved. However, on Saturday morning I received a call from our test pilot, Bob Berry. "Jan, your yaw damper does not work. Things are worse than before: as soon as I turned the yaw damper on, the airplane started unstable oscillations. Please come to Wichita to help us straighten this out."

I asked Bob whether someone had reversed the wiring but he assured me that they had verified the wiring polarity.

4.19.2 Cause

On my way back to Wichita I racked my brain to figure out what might have gone wrong. Along the way it occurred to me that perhaps the tilting bracket had been installed backwards. According to our root-locus studies that would indeed undamp the Dutch roll. Therefore, as soon as I arrived at the plant I went to the experimental shop to check the bracket. Sure enough, it had been installed backwards. It can be shown that the result of "tilting" the gyro in the wrong direction results in unstable operation. A detailed explanation of this is found in Ref. 4.11, Chapter 11.

4.19.3 Solution

Prevent erroneous installation of the gyro bracket by designing it for a one-way fit.

4.19.4 Lesson

Always assume that if something can be installed the wrong way, it will be. In this case, the designer (me!) should have designed the bracket so that it could have been installed only one way. Again, when possible, design for a one-way fit.

4.20 Loss of Control due to Unwanted Extension of Ground and Flight Spoilers

4.20.1 Problem

During June of 1974 an IBM operated Grumman G-1159 (Figure 4.26) was on a training flight.

The airplane made several 360 degree rolls and then dove into a swampy area near Kline, South Carolina. The crew of three was killed.

Figure 4.26 Grumman G-1159 (Not accident aircraft, Courtesy G. Helmer)

4.20.2 Cause

In Ref. 4.20 the NTSB determined that the probable cause of this accident was: “an unwanted extension of the ground and flight spoilers, which resulted in loss of control at an altitude from which recovery could not be made. The ground spoilers probably deployed because of a hot electrical short circuit in the spoiler extend circuitry. Whereas the spoilers probably deployed symmetrically, the left ground spoiler actuator failed in flight and caused a loss of lateral control. The subsequent loss of pitch control was caused by the full nose-down elevator trim tab position and the high aircraft speed.”

The following findings in Ref. 4.20 are relevant to understanding the design alterations needed to prevent re-occurrence:

- The only evidence of an aircraft malfunction was the extended position of the ground spoiler panels at impact.
- The elevator trim tab position was full nose-down to the electrical stop. The aileron manual trim was set 9.5 degrees left wing down.
- The landing gear and the wing flaps were retracted at impact.
- The right and left ground spoilers were unlocked and extended in flight. Their exact position could not be determined. The inboard and outboard flight spoilers on each wing were extended between 24 and 25 degrees, and 24 and 35 degrees respectively. The left ground spoiler actuator was fractured, probably by high air-loads.

- The cause of the unwanted ground spoiler extension was probably a hot electrical short which bypassed the four ground spoiler interlocks installed in the system. The extension of the ground spoilers caused the flight spoilers to extend.
- This unwanted extension of the spoilers occurred at a relatively low airspeed and when the aircraft was in a landing approach configuration.
- The unwanted extension of the spoilers resulted in an upset and a rapid loss of altitude.
- The pilots probably attempted recovery from this upset by retracting the gear and flaps, increasing power, and accelerating the airplane to a speed of more than 300 kts.
- The resulting high airloads failed the actuator rod of the left ground spoiler which resulted in lateral asymmetry and high rolling moments.
- During their attempts to recover from the ensuing rolls, the pilots may have inadvertently activated the electrical trim tab to the full nose-down position.
- The pilots were unable to maintain pitch control and had insufficient altitude in which to recover from the ensuing dive.

4.20.3 Solution

In a letter to the FAA (Ref. 4.11, Appendix E) the NTSB states:

"Although the aircraft was probably certificated with the belief that the design of the ground spoiler actuation system provided sufficient redundancy to prevent in-flight deployment, the Board's review of the system design disclosed what we believe to be a potentially dangerous condition. A hot electrical short, which bypasses the redundant switches in the line to the power terminal of the solenoid, could cause the unwanted actuation of the ground spoiler system. The original configuration of the aircraft provided a switch on each main landing gear strut which completes the circuit by connecting the power source to the ground spoiler's control valve solenoid.

On August 20, 1971 the manufacturer issued Service Change No. 98, which provided additional redundancy by breaking both the power source and the ground source to the solenoid, through the landing gear switches. This change, which was not mandatory, affected aircraft serial Nos. 1 through 90. The manufacturer advised that 39 aircraft had not been changed, including the accident aircraft. We believe that incorporation of this Service Change will eliminate the danger of a similar failure, i.e. "hot electrical short," unwanted deployment of the ground spoilers in flight, and possible subsequent loss of control.

Although the incorporation of Service Change 98 may eliminate the possibility of in-flight ground spoiler deployment, we believe that a hot electrical short could possibly prevent the retraction of the spoilers on take-off from a touch-and-go landing.

For this reason, the crew should have a means available to retract the spoilers at any time. In this regard, deployment of the spoilers cannot be detected visually, and some warning system may be required to alert the crew to unwanted spoiler deployment."

These changes were made.

4.20.4 Lesson

Design engineers should have predicted the scenario that led to the demise of this airplane. As stated before: playing the what-if game is very helpful in identifying potential flight hazards.

4.21 Take-off with Locked Elevator II

4.21.1 Problem

In September of 1975 a Canadair CL-44-6 (Figure 4.27) operated by Aerotransportes Entre Rios S.R.L. crashed while attempting a take-off. The crew and two passengers survived the crash while one person on the ground was injured.

Figure 4.27 Canadair CL-44-6 (Not accident aircraft, Courtesy G. Helmer)

4.21.2 Cause

In Ref. 4.21 the NTSB determined that the probable cause was: "an attempt to take-off with an external, makeshift, flight control lock installed on the right elevator."

Quoting from Ref. 4.22, page 8: "This aircraft type, a cargo version of the Bristol Brittania passenger transport, is equipped with an internal gust lock system to prevent damage to the flight control system by wind gusts when on the ground. When any control surface is locked, a micro-switch, mounted on the locking actuator, operates the amber master caution lights and the control surface lock windows on the annunciator panels. These lights will remain illuminated until all control surfaces are unlocked. The gust lock lever is interconnected with the engine power levers so that take-off power cannot be applied to more than one engine on each side when the gust lock lever is in the "locked" position.

According to the CL-44-6 operating manual, the flight controls must be unlocked during the pre-take-off check-list and the flight crew should observe the control surface positions on the indicators. With an external gust lock installed on the right elevator, releasing the internal gust lock would permit the left elevator to droop down while the right elevator remained faired with the horizontal stabilizer. The control surface indicators would show these positions.

The external elevator control lock was not produced by the aircraft manufacturer. The manufacturer first became aware of the device during the investigation of this accident. Such a device was not a part of, nor included in, the certification.

It turned out that the external lock was carried on the aircraft just in case the internal lock system failed and was not supposed to be used normally.

4.21.3 Lesson

Here is an example where the designers did everything right but the operator found a way to cause a problem.

4.22 Take-off with Rudder and Aileron Controls Locked

4.22.1 Problem

In December of 1977 a Douglas DC-3 (Figure 4.28) operated by National Jet Services on a passenger charter service crashed at the Evansville Dress Regional Airport in Indiana.

Figure 4.28 Douglas DC-3 (Not accident aircraft, Courtesy H. Chaloner)

The aircraft had taken off under instrument meteorological conditions (IMC) weather. The crash occurred less than 1.5 minutes after take-off. All 29 persons on board were fatally injured.

4.22.2 Cause

According to Ref. 4.22 the following factors contributed to this crash:

- The external right aileron and rudder control locks were installed. The control locks were not discovered during the before-take-off check-list and the control locks were in place when the aircraft crashed.
- The aircraft center of gravity (c.g.) was aft of that shown on the load manifest, was aft of the optimum c.g. range for this aircraft but forward of the most aft allowable c.g.
- The aircraft lifted off the ground prematurely and below the take-off decision speed. This was probably because of the aft c.g.
- The airplane then entered the so-called region of reversed command from which the pilot could not escape.

In the DC-3, when loaded at aft c.g. a considerable forward push force is required on the control column to prevent premature lift-off. Unless a pilot is fully aware of this, the airplane will have a tendency to pitch up after becoming airborne. This can place the airplane in the so-called region of reversed command from which an escape at very low altitude is unlikely. For a detailed discussion of this phenomenon, see Ref.4.11 Part I, pp 186-189 (where this phenomenon is referred to as speed-instability).

In the DC-3 it is possible to move the cockpit controls a bit even with external surface locks installed. That is because of elastic deformation in the control cables. The pilot may therefore have thought that the controls were free.

4.22.3 Solution and Lesson

There have been many instances of DC-3 aircraft taking off with locked flight controls and not always with fatal results. As indicated before airplanes should be designed to prevent this from happening in the first place.

4.23 Take-off with Mistrimmed Stabilizer

4.23.1 Problem

In February of 1978 a Beech Model 99 (Figure 4.29*)* crashed near Richland, WA immediately following take-off.

Figure 4.29 Beech Model 99 (Not accident aircraft, Courtesy F. Duarte Jr.)

All seventeen persons aboard were fatally injured.

4.23.2 Cause

In Ref. 4.23 the NTSB lists as the probable cause: "the failure or inability of the flight crew to prevent a rapid pitch-up and stall by exerting sufficient push force on the control wheel. The pitch-up was induced by the combination of a mis-trimmed horizontal stabilizer and a center of gravity near the aft limit. The mis-trimmed stabilizer condition resulted from discrepancies in the aircraft trim system and the flight's crew probable preoccupation with making a timely departure. Additionally, a malfunctioning stabilizer trim actuator detracted from the flight crew's effort to prevent the stall."

Importantly, the NTSB listed as contributing factors: inadequate flight crew training, inadequate trim warning system check procedures, inadequate maintenance procedures, and ineffective FAA surveillance.

It is instructive to list several of the NTSB findings in Ref. 4.23:

- The horizontal stabilizer trim position indicator was unreliable
- The horizontal stabilizer trim-in-motion system was unreliable.
- The horizontal stabilizer out-of-trim warning system was inoperative.
- The horizontal stabilizer actuator clutch slipped.
- The aircraft was not airworthy.

Research into the service history of this aircraft indicated that it had been operated by three previous owners and that there had been many problems with the flight control and trim system. The NTSB also found that the investigation was hindered by the lack of a Flight Data Recorder (FDR) and a Cockpit Voice Recorder (CVR).

4.23.3 Solution

In Ref. 4.23, issue an Airworthiness Directive applicable to all Beech 99, 99A, A99, A99A and B99 model aircraft to require an immediate one-time inspection of the horizontal stabilizer trim system to ascertain that all components of the system and its associated position-indicating and –warning circuits are operational within specified tolerances.

Require an inspection to insure that the primary and secondary mode of the horizontal stabilizer actuator are capable of deflecting the stabilizer under specified airloads. The exact instructions should be furnished by the Beech Aircraft Corporation. The inspection should be made as soon as the Beech instructions are available and repeated at 2,000 hour intervals.

Change the minimum equipment list to make the out-of-trim warning system a mandatory requirement for flight.

The NTSB notes that:

“the investigation was difficult and time-consuming because of the lack of definitive information on the aircraft’s performance and on the flight crew’s reaction to the emergency situation which arose immediately after take-off. Information from a flight data recorder (FDR) and a cockpit voice recorder (CVR) would have provided invaluable information in both of these areas, would have significantly reduced the investigative effort, and would have provided more direct evidence of causality. The Safety Board believes that these recorders are virtually a prerequisite to improvements in safety in commuter air carrier and corporate/executive operations involving complex multi-engine aircraft. Therefore, we reiterate Safety recommendations A-78-27, -28 and -29, dated April 13, 1978 and we urge the FAA’s early action on those recommendations.”

4.23.4 Lessons

1. Design engineers should try to make trim systems as robust as possible. Aircraft in regional passenger operations see many more flight cycles per day than do many other types of airplanes. The service history of this airplane suggests that the system as designed was not robust.

2. Also, a functional out-of-trim warning system not being a mandatory requirement for flight in passenger carrying operations is not appropriate. The author’s question for designers is: “does one really need a regulation for this?”

4.24 Defunct Elevator Hard-stop

4.24.1 Problem

In August of 1978 a Piper PA-31-350 (Figure 4.30) operated by Las Vegas Airlines crashed shortly after take-off from the North Las Vegas Airport.

Figure 4.30 Piper PA-31-350 (Not accident aircraft, Courtesy Colin Zuppicich)

All persons aboard (one pilot and nine passengers) were killed.

4.24.2 Cause

In Ref. 4.24 the NTSB determined that the probable cause was: “a backed out elevator down-stop bolt that limited down elevator travel and made it impossible for the pilot to prevent pitch-up and stall after take-off. The Board was unable to determine how the down-stop bolt jam nut locking device came loose and allowed the stop bolt to back out.”

The normal elevator travel in this airplane is from 16 degrees trailing edge up to 20 degrees trailing edge down. In this accident, the travel was from 16 degrees trailing edge up to 1.5 degrees trailing edge down, because of the backed our stop bolt.

It was found that the design of the bolt stop consisted of a stop bolt threaded into a casting and secured by a lock nut torqued against the face of the casting. This is a fairly common design in many general aviation aircraft. However, if the nut is not properly torqued, it can vibrate loose which would allow the stop bolt to back out.

4.24.3 Solution

Redesign all control surface stops with positive means of assuring the stops cannot be affected by wear or slackness. They also should be able to withstand the loads imposed upon them by the flight controls in the most unfavorable flight conditions. This is in fact what is implied by CAR 3.340 (to which this airplane was certified) and by its successor regulation, 14 CFR 23.675.

4.24.4 Lessons

1. Design engineers should consider the detail design of control surface stops as flight crucial. Their failure can and has caused major accidents.

2. The use of torque nuts in control stops does not seem to be responsive to either the old CAR 3.340 or the new 14 CFR 23.765.

4.25 Control System Compliance

4.25.1 Problem

The first flight of the SIAI-Marchetti S-211 (See Figure 4.31) occurred in 1981. When the test pilot landed he reported that the roll-rate performance of the airplane was only about a third of what was expected. It seemed that the roll control power was inadequate.

The contract to design the lateral (roll) flight control system of the S-211 had been given to a small company in Italy. That company had never designed a flight control system for a high performance airplane. Before signing that contract, SIAI-Marchetti asked this author to visit them and conduct a one-day short course with do's and don'ts of flight control system design. I did, but some of it must not have sunk in.

Figure 4.31 SIAI Marchetti S-211 (Not accident aircraft, Courtesy SIAI Marchetti)

4.25.2 Cause

I suspected that the ailerons did not deflect fully following a full lateral stick deflection. Therefore, I asked the test pilot to have the ailerons loaded up with a simulated loading apparatus and to deflect the stick in the cockpit to the stop and measure the aileron deflection. The next day he called and said that he got only about 9 degrees of aileron deflection. At that point I knew that the cause of the problem was control system compliance.

Control system compliance can occur if one or more pulley brackets have not been mounted stiffly in the structure. Figure 4.32 shows how control cable forces can exert a force on a pulley which is mounted on flexible structure.

I suspected therefore that one or more of the lateral control pulley brackets had not been mounted stiffly in the airframe and that the ensuing deformation limited the aileron deflection. My Italian friends checked and found that one pulley had been installed right in the middle of the rear pressure bulkhead. In the S-211 that pressure bulkhead is a rather thin membrane made of 0.06 inch aluminum.

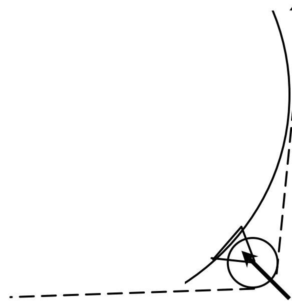


Figure 4.32 *Example of Pulley Bracket Mounted on a Flexible Support*

4.25.3 Solution

By stiffening that bulkhead the problem was solved. The cost is some added weight.

4.25.4 Lesson

Avoid control system compliance. Check all pulley and quadrant attachments for stiffness before passing a design to manufacturing. Even then, it is desirable to simulate control surface loading and check control surface deflections on the ground before first flight. Design engineers should always be aware of elastic deformation in systems where forces occur which can easily deform support structure. Particularly in control systems elastic deformation (called system compliance) is not acceptable.

4.26 One Engine Out Control Problem

4.26.1 Problem

In September of 1985 a Midwest Express Airlines DC-9-14 (Figure 4.33) experienced an uncontained engine compressor failure right after take-off.

Figure 4.33 Douglas DC-9-14 (Not accident aircraft, Courtesy G. Helmer)

Control was lost shortly thereafter and the airplane crashed killing all 31 persons aboard. The accident happened in clear weather.

4.26.2 Cause

In Ref. 4.25 the NTSB determined that the loss of control was precipitated by improper operation of the flight controls by the flight crew, specifically the introduction of incorrect rudder pedal forces about 4 to 5 seconds after the right engine failure, followed by aft control column forces, which allowed the airplane to stall at a high airspeed (accelerated stall) about 10 seconds after the right engine failed.

The DC-9 series of airplanes in fact behave in a very docile manner even if an engine fails right after take-off. However, if the rudder is moved in the wrong direction, the asymmetric yawing moment due the operating engine, the extra drag on the failed engines and the rudder induced yawing moment create a significant amount of sideslip in a fairly short period of time. Because of the high rolling moment due to sideslip, typical for all swept aft wing jet transports, the airplane would then begin a rapid roll. If on top of that the longitudinal controls are moved in a nose-up direction, control will be lost in the ensuing stall. In its many findings published in Ref. 4.25 the following are of particular interest:

- Forward visual cues (outside the cockpit) were not available to the crew at the time the right engine failed. Peripheral visual cues were available.
- The visual flight simulator, which was used by the crew members in training, did not provide onset yaw and longitudinal acceleration cues, peripheral visual cues, or aural cues which were available to the crew in the airplane.
- Both crew members were relatively inexperienced in DC-9 flight operations.

- The FAA Principal Operations Inspector who was responsible for oversight of Midwest Express was inexperienced in FAR 121 turbojet air carrier operations.
- A "silent cockpit" philosophy was suggested by Midwest Express in response to certain emergency situations, although the concept was not approved by the FAA and was in conflict with approved emergency procedures.

4.26.3 Solution

The NTSB made a series of recommendations relating to the engine failure type. The Board also made recommendations to the FAA in regard to crew training procedures.

4.26.4 Lessons

1. Even though many jet transports are not particularly difficult to control following an engine failure right after take-off, designers should assume that the crew controlling the airplane may not be very well skilled. That seems to be a fact of life.

2. Therefore, the flight control system should be designed to automatically move the critical controls in the correct direction. In many modern transports this has now been done. The author believes this feature should be incorporated in all multi-engine airplanes.

4.27 Redundant System is not Redundant

4.27.1 Problem

In July of 1989 a McDonnell-Douglas DC-10-10 (Figure 4.34) experienced an uncontained failure of the fan disk in its center (No. 2) engine.

The fan debris disabled all hydraulic lines servicing the lateral, longitudinal and directional flight controls. The pilots, using differential engine thrust on the remaining No.1 and No.3 engines to control the airplane, made a partially successful landing in Sioux City, Iowa. Of the 11 crew members and 285 passengers on board, one flight attendant and 110 passengers were fatally injured.

Figure 4.34 McDonnell-Douglas DC-10-10 (Not accident aircraft, Courtesy www.al-aliners.be)

4.27.2 Cause

Refs. 4.27–4.29 describe the crash and its investigations in detail. Quoting from Ref. 4.26, the NTSB established that the probable cause was:

“the inadequate consideration given to human factors limitations in the inspection and quality control procedures used by United Airlines’ engine overhaul facility which resulted in the failure to detect a fatigue crack originating from a previously undetected metallurgical defect located in a critical area of the stage 1 fan disk that was manufactured by General Electric Aircraft Engines. The subsequent catastrophic disintegration of the disk resulted in the liberation of debris in a pattern of distribution and with energy levels that exceeded the level of protection provided by design features of the hydraulic systems that operate the DC-10’s flight controls.”

Ref. 4.26 addresses a number of significant safety issues. Three of these are:

1. General Electric Aircraft Engines’ (GEAE) CF6-6 fan rotor assembly design, certification, manufacturing, and inspection.
2. United Airlines’ maintenance and inspection of CF6-6 engine fan rotor assemblies.
3. DC-10 hydraulic flight control system design, certification and protection from uncontained engine debris.

To understand how the loss of control in this accident happened, it is necessary to review some aspects of the DC-10 design. Figure 4.35 shows the general arrangement of the No. 1, No. 2 and No. 3 engines in the airframe.

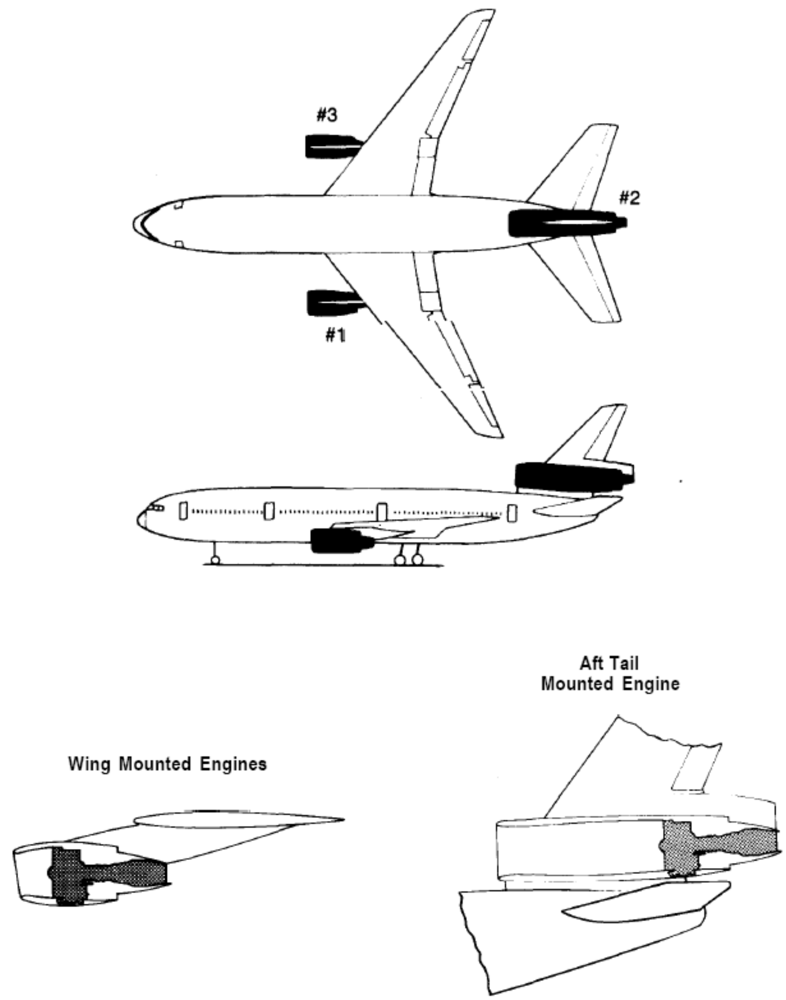


Figure 4.35 Engine Arrangement in the DC-10 (Courtesy NTSB)

Figure 4.36 shows an outline of the area where the engine components which were torn off (precipitated by the stage 1 fan disk failure) were located. Some of the debris from this area put all hydraulic systems out of commission because of hydraulic line failures induced in areas of the horizontal stabilizer shown in Figure 4.37.

In Figure 4.37, note the line indicating the stage 1 fan disk. Predictable areas of debris spreading include a cone with a 15 degree angle forward of the fan disk (this is sometimes referred to as the spread angle). For a more precise definition of debris spread angles, spread patterns and energy levels the reader is referred to Ref. 4.29 which was not yet published when the DC-10 was being certified. Such a cone would encompass the damaged areas of the hydraulic system indicated in Figure 4.37.

Figure 4.36 Area of Engine Forming the Source of Fatal Debris (Courtesy NTSB)

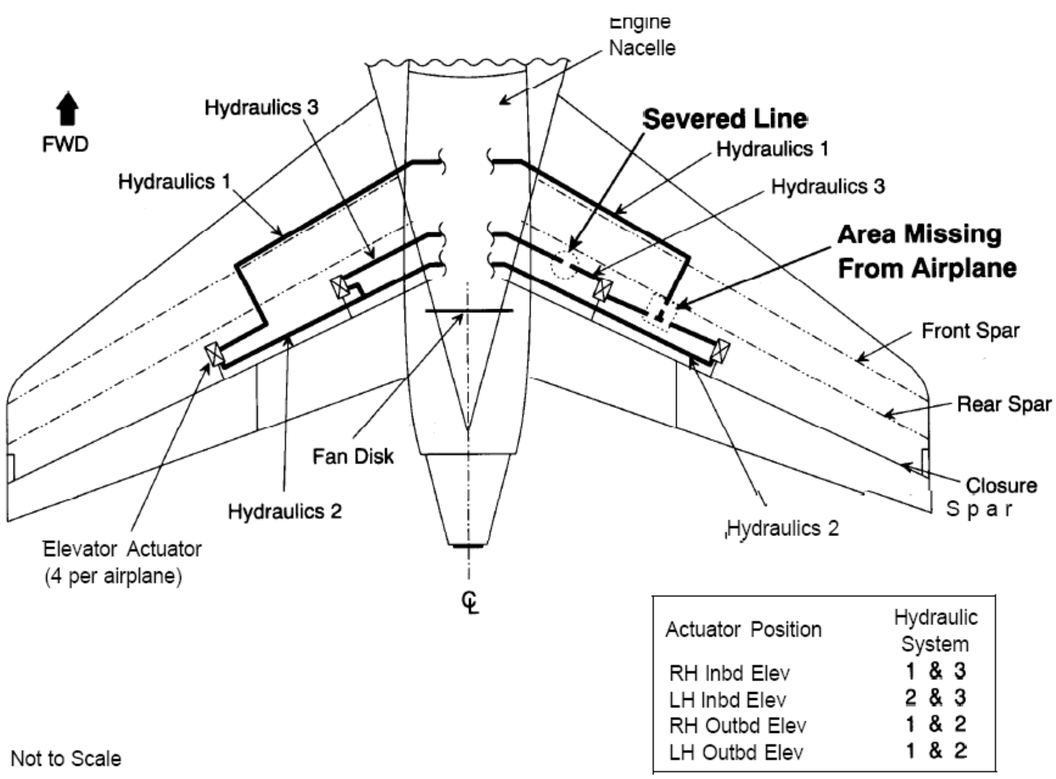


Figure 4.37 View of Damaged Hydraulic System Components (Courtesy NTSB)

In retrospect, the design of the hydraulic system of the DC-10 was not redundant: one failure event rendered the airplane uncontrollable.

The reason the fan disk failed was because of an undetected metallurgical defect in the material from which the disk was machined. Not only was this defect not detected at General Electric, but when a fatigue crack developed emanating from that area of defect, the fatigue crack was not detected. The NTSB rightfully was concerned about both fan disk design, manufacturing, quality control and inspection issues.

4.27.3 Solutions

As a result of this accident Douglas incorporated certain hydraulic system safety enhancements in all DC-10 aircraft and later in all MD-11 aircraft. The enhancement basically consists of a system that will prevent the complete loss of hydraulic fluid in the event of a repeat failure of a center engine fan disk.

Also, the safety of flight crucial systems in view of uncontained engine failures must now be analyzed in detail in terms of a formal fault-tree analysis. Ref. 4.29 contains a summary of the methodology that can be used.

Improved manufacturing and inspection processes have since been introduced at General Electric and at United Airlines.

The NTSB expressed its concern in Ref. 4.26 that: "other aircraft may have been given similar insufficient consideration in the design for redundancy of the motive power source for flight control systems or for protecting the electronic flight and engine controls of new generation aircraft. Therefore, the Safety Board recommends that the FAA conduct system safety reviews of currently certificated aircraft in light of the lessons learned in this accident to give all possible consideration to the redundancy and protection of power sources for flight and engine controls."

4.27.4 Lessons

1. Designers should not need a regulation to conclude that the basic DC-10 hydraulic system design was not redundant and therefore not safe. It is the view of this author that the airplane should not have been certified in this manner.

2. Some other unforeseen event may render the aerodynamic flight controls of a transport unserviceable at some future date. Therefore, NASA-Dryden developed an automatic system that could use the remaining, operable engine(s) as a sole means of safely landing the airplane. Some results of this interesting investigation may be found in Ref. 4.30. Extensive flight simulations of this system on a twin-engine fighter, on 3- and on 4-engine transports showed that such a system can indeed be used to safely land an airplane.

3. The author finds it hard to understand why these systems are not incorporated in any of today's airplanes.

4.28 Uncommanded Elevator Travel

4.28.1 Problem

In October of 1993 a British Airways Boeing 747-436 (Figure 4.38) suddenly pitched down, right after take-off while the landing gear was being retracted.

Figure 4.38 Model of Boeing 747-436 (Courtesy geminijets.com)

The pilot was able to maintain a positive rate of climb by applying almost full aft deflection on the control column. Some seconds later the flight controls responded normally again and the airplane continued to its destination (Bangkok, Thailand) without further incident.

4.28.2 Cause

The investigation in Ref. 4.31 found the following causal factors:

The secondary slide of the servo valve of the inboard elevator Power Control Unit (PCU) was capable of over-traveling to the internal retract stop; with the primary slide moved to the limit imposed by the extend linkage stop, the four chambers of the actuator were all connected to both hydraulic supply and return, the servo valve was in full cross-flow resulting in uncommanded full down travel of the right elevators.

A change to the hydraulic pipe work with the right inboard elevator PCU was implemented on the Boeing 747-400 series aircraft without appreciation of the impact that this could have on the performance of the unit and consequently on the performance of the aircraft elevator system.

4.28.3 Solution

Three safety recommendations were made to preclude this from happening again.

4.28.4 Lesson

The real puzzle here is why any change in the primary flight control system of a transport airplane was made without checking out the consequences on a systems simulator (with the actual hardware) or, at least, on a prototype of the airplane. Changes to primary flight control hydraulics should be made only after carefully checking the consequences.

4.29 Uncommanded Roll at Take-off

4.29.1 Problem

In August of 1993 a British Airways Airbus A-320-212 (Figure 4.39) was on its first flight after a flap change.

The airplane exhibited an un-commanded roll to the right after lift-off. The problem persisted until the airplane landed back at London, Gatwick about 37 minutes later. There were no injuries.

Figure 4.39 Airbus A-320-212 (Not accident aircraft, Courtesy www.al-airliners.be and British Airways)

4.29.2 Cause

Several causal factors were identified in Ref. 4.32, the important ones of which are:

During the flap change compliance with the requirements of the A320 maintenance manual was not achieved in two areas:
During the flap removal the spoilers were placed in maintenance mode and moved using an incomplete procedure. Specifically, the collars and flags were not fitted.
The re-instatement and functional check of the spoilers after flap fitment were not carried out.

The purpose of the collars and the way in which the spoilers functioned was not fully understood by the maintenance engineers. This misunderstanding was due in part to familiarity with other aircraft and contributed to a lack of adequate briefing on the status of the spoilers during a shift handover.

During the independent functional check of the flying controls the failure of spoilers 2 and 5 on the right wing to respond to right roll commands was not noticed by the pilots.

The operator had not specified to its pilots an appropriate procedure for checking the flight controls.

4.29.3 Solution

A number of safety recommendations were made to prevent a re-occurrence.

4.29.4 Lesson

Maintenance manuals should be easy to understand. The way the flight controls operate and what should be done to them during maintenance should be clearly spelled out.

4.30 Elevator Trim Tab Failure

4.30.1 Problem

In December of 1995 a Canadian Airlines International Boeing 737-200 (Figure 4.40) experienced severe airframe vibration during a climb to cruise altitude.

Figure 4.40 Boeing 737-200 (Not accident aircraft, Courtesy G. Helmer)

The crew decided to return to Vancouver and as the airplane was slowed down, the vibration stopped. Moderate vibration was again experienced during the approach, after the flaps were lowered. The airplane landed safely and there were no injuries. Inspection of the airplane revealed that a two-foot section of the right elevator trim tab was missing.

4.30.2 Cause

The 737 has two elevator trim tabs, one on each side. Each tab is 90 inches long and is attached to the elevator at 4 hinge points about 27 inches apart. The section between the first and second hinge points had broken off and was not found. The remaining small inboard section of the tab, with the control horn, broke free from the first hinge and was held to the elevator only by the attachment to the control rod. The outboard section was still attached by the remaining three hinges. Marks on the hinges indicated that this section was probably oscillating rapidly through its full range of movement. The tab oscillations in turn caused the elevator to oscillate which is the vibration felt by the crew.

The investigation of the remaining tab (Ref. 4.33) indicated that the outer layers of the composite surface material had disbonded from the honeycomb core. This weakened the tab enough for the aerodynamic loads to cause it to fail in flight. The remaining tab portion was sent to Boeing for a detailed analysis. Boeing engineers concluded that the tab damage was consistent with damage inflicted by a ground vehicle bumping into the tab.

4.30.3 Lesson

Damage by ground vehicles to airplanes parked on ramps is estimated to be 2 billion US Dollars per year. Until more careful ground vehicle operators appear on the scene, designers should probably include in their "what-if" analysis scenarios the possibility of unreported damage to flight control and stabilizing surfaces.

4.31 The Hard-stop which was not a Hard-stop

4.31.1 Problem

In January of 2000 a McDonnell-Douglas MD-83 (Figure 4.41) of Alaska Airlines lost control at cruise altitude and crashed into the Pacific Ocean. There were no survivors.

Figure 4.41 McDonnell-Douglas MD-83 (Not accident aircraft, Courtesy Tim Perkins)

4.31.2 Cause

The cause was established to be a stripping of the threads in the stabilizer jack-screw acme-nut. This failure allowed the horizontal stabilizer to move leading edge up to a very large angle making the airplane uncontrollable in pitch. The NTSB report (Ref. 4.34) cites lack of lubrication due to lapses in required maintenance of this system as the primary reason for this failure.

Figure 4.42 shows a perspective view of the stabilizer trim actuating mechanism. The mechanical stops in Figure 4.42 are designed to limit the motion of the stabilizer in the up-down direction. In the MD-83 these limits are set so that the stabilizer is limited from -12.5 degrees (leading edge down) to + 2.1 degrees (leading edge up). There are also electrical limit switches set to approximately the same limits.

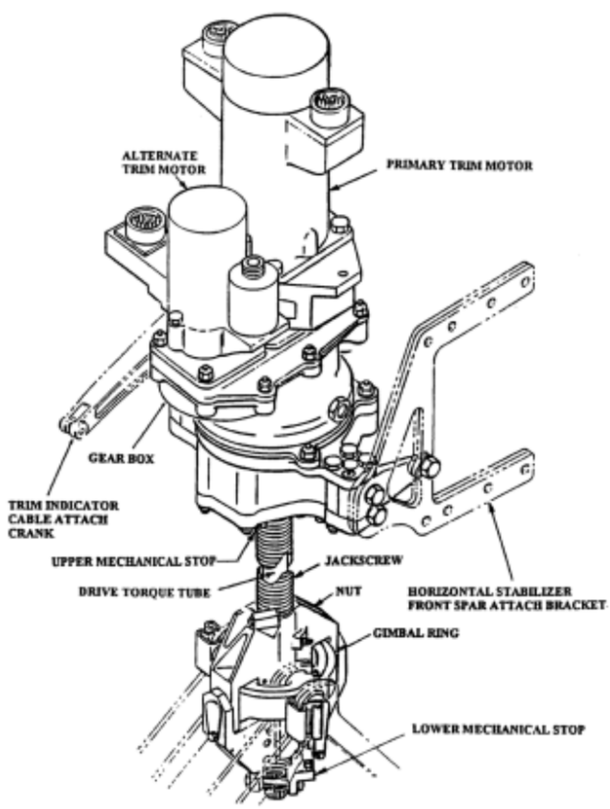


Figure 4.42 Perspective of the Stabilizer Trim Actuating Mechanism (Courtesy NTSB)

Generically, the reason for incorporating mechanical stops in a flight control system is to prevent control surfaces from reaching deflection angles beyond which airplane controllability becomes questionable. Such stops are often referred to as hard-stops. In most airplanes these hard-stops are anchored in primary structure which, by physical contact, prevents further motion.

In the MD-83 (and in fact in the entire DC-9, MD 80/90 and Boeing 717 derivative family) these hard-stops are in fact not hard-stops. As long as the threads in the stabilizer jackscrew are intact the acme nut stops can be considered to be hard-stops. The reason is the very large safety margins used in the design of the jackscrew.

However, the hard-stops cease to be hard-stops if the thread strips due to improper maintenance procedures. With improper maintenance one event (a mechanic not doing the job properly or the maintenance organization failing to install and adhere to maintenance procedures) can cause a catastrophic event.

In commercial transport airplanes the design paradigm for flight crucial structure or systems is that one failure event should not result in a catastrophe. Clearly, this paradigm is violated with this design.

The following quotation from Ref. 4.34, page166 should raise concerns in any design engineer or DER:

"Further, the Safety Board is concerned that the FAA certified a horizontal stabilizer trim system that had a single-point catastrophic failure mode."

4.31.3 Solution

The NTSB report in Ref. 4.34, page 166, also suggests the following action:

"The Safety Board concludes that transport-category airplanes should be modified, if practicable, to ensure that horizontal stabilizer trim system failures do not preclude continued safe flight and landing. Therefore, the Safety Board believes that the FAA should conduct a systematic engineering review to (1) identify means to eliminate the catastrophic effects of total acme nut thread failure in the horizontal stabilizer trim system jackscrew assembly in DC-9, MD-80/90, and 717 series airplanes and require, if practicable, that such fail-safe mechanisms be incorporated in the design of all existing and future DC-9, MD-80/90, and 717 series airplanes and their derivatives; (2) evaluate the horizontal stabilizer trim systems of all other transport-category airplanes to identify any designs that have a catastrophic single-point failure mode and, for any such system; (3) identify means to eliminate the catastrophic effects of that single-point failure mode and, if practicable, require that such fail-safe mechanisms be incorporated in the design of all existing and future airplanes that are equipped with such horizontal stabilizer trim systems."

4.31.4 Lessons

1. Single-point failures, whether caused by a structural failure, systems failure or failure of maintenance procedures should not cause catastrophic consequences.

2. In this case there have been several precursor events (incidents) that pointed toward eventual catastrophic consequences (for specifics, see Ref. 4.34 pp 41-42). The proper inference from these events was never drawn.

3. The author, to his chagrin, observes that the fundamental changes to the hard-stop design (needed to satisfy design lesson 1) have yet to be demanded by the FAA.

4.32 Jammed Servo Tab

4.32.1 Problem

During February of 2000 a McDonnell-Douglas DC-8-71F (Figure 4.43) belonging to Emery Worldwide Airlines experienced control difficulties immediately after take-off.

An attempt by the crew to return to the airport was not successful and the airplane crashed. All three crew members died.

Figure 4.43 McDonnell-Douglas DC-8 (Not accident aircraft, Courtesy www.al-aliners.be)

4.32.2 Cause

The DC-8 has a reversible flight control system in which the pilot directly controls two servo tabs, one on the left elevator and the other on the right elevator (the reader is referred to Figure 4.11). The cause was found to be a disconnected bolt. The bolt in question normally connects a pushrod (operated by a cable system from the cockpit control column) with a servo-tab crank fitting on the right elevator. The bolt is normally secured with a castellated nut and a cotter pin. It could not be determined where in the maintenance procedures either was omitted.

As a result the control tab was jammed in a trailing edge down position. As dynamic pressure increases during the take-off roll this jammed servo tab then drives the elevator trailing edge up. After lift-off the pilots were struggled to overcome the very large control forces needed to prevent the nose from pitching up.

4.32.3 Solution

Figure 4.44 shows the right elevator control tab pushrod contacting the servo-tab crank fitting on a test airplane to demonstrate the effect.

Figure 4.44 Photograph of the Right Elevator Control Tab Pushrod Contacting the Crank Fitting on a Test Airplane (Courtesy NTSB)

To quote from Ref. 4.35, page 62:

"In its submission on this accident, Boeing stated that a failure/disconnection of the control tab crank fitting/pushrod attachment was considered by the manufacturer and the FAA during the DC-8 development and certification. The submission noted, however, that the trailing edge down control tab motion and the subsequent rod end escape from between the crank fitting lugs was a failure mode that had not been anticipated, nor had it been experienced prior to this accident."

The NTSB (in Ref. 4.35 on pages 84-85) recommended a total of 15 of maintenance and crew operating procedures to prevent this from happening again.

4.32.4 Lesson

Designers should anticipate this type of failure mode and alter the design so that this type of jam becomes extremely improbable.

4.33 Unnecessary Loss of Control

4.33.1 Problem

In May of 2001 a British Aerospace Jetstream 3101 (Figure 4.45) operated by East Coast Aviation Services crashed near Wilkes-Barre, PA. The airplane had been conducting a missed approach due to the weather being below minimums.

Figure 4.45 British Aerospace Jetstream 3101 (Not accident aircraft, Courtesy F. Duarte Jr.)

While on the second approach the right engine failed due to fuel starvation, and the crew lost directional control. All seventeen passengers and the crew of two were killed.

4.33.2 Cause

In Ref. 4.36 the NTSB determined as the probable cause: "the failure of the flight crew to ensure adequate fuel supply for the flight, which led to the stoppage of the right engine due to fuel exhaustion and the intermittent stoppage of the left engine for the same reason. The crew failed to maintain directional control after the initial engine stoppage."

It is believed that the intermittent operation of the left engine (after the right engine had stopped and the crew started to maneuver the airplane in a left turn) was caused by intermittent unporting of the left wing fuel tank.

4.33.3 Solution

At the time of publication of this book the NTSB had not yet published its final report with recommendations.

4.33.4 Lessons

1. Obviously, aircrews should ensure that adequate fuel is on board.

2. However, if the fuel system had been designed with a sufficient number of ports the accident might still not have happened. The reader is referred to Section 6.12 for a similar event.

3. It is asking a lot of a crew to keep an airplane under control with an engine out (for whatever reason) and in poor weather. Had the flight control system been designed to automatically control an engine-out situation, the accident also might not have happened.

4.34 Frozen Ailerons

4.34.1 Problem

In December of 2000 a Gates-Learjet 35A (Figure 4.46) was on a medical evacuation flight.

Figure 4.46 Gates-Learjet Model 35A (Not accident aircraft, Courtesy F. Duarte Jr.)

Right after take-off, with the autopilot engaged, the airplane began banking 5 degrees to the right for no apparent reason. The pilot disengaged the autopilot and the ailerons became unmovable. Various control inputs were unsuccessful and the bank angle increased to 20 degrees. After applying considerable force full aileron control returned and the airplane was landed in Vancouver without further incident.

4.34.2 Cause

According to Ref. 4.37 the cause of the problem was an excessive amount of water in the aileron brush seals and a distortion of the drainage channels. The airplane had been standing in heavy rain for several hours before the flight. The drainage channels were distorted because of wear.

4.34.3 Solution

Brush seals were applied to the Model 35A to prevent aileron buzz at high Mach numbers. The seals have to be lubricated with a special silicone grease every 300 hours. The practice was to be generous with the grease although the Learjet maintenance manual cautions against over application, possibly clogging the drainage channels. The latter would result in aileron buzz at high Mach numbers. The solution therefore is to be very careful with the inspection and maintenance of these seals. Furthermore, experienced Learjet pilots will keep the ailerons moving a bit when flying through conditions conducive to freezing of the seals.

4.34.4 Lesson

Ailerons freezing has been a (predictable) operational problem. Designers should search for other solutions. Airfoil shape has a large effect on aileron buzz. Also, carefully tailored vortex generators may solve this buzz problem.

4.35 Misrouting of Control Cables

4.35.1 Problem

According to Ref. 4.38 in February of 2001 a Boeing 737-33A (Figure 4.47) began to roll slightly to the right when the speed brake was selected during a descent into Sydney, Australia.

Figure 4.47 Boeing 737-33A (Not accident aircraft, Courtesy Keith Burton)

The autopilot was disengaged and the speed brake was again selected with the same result. The speed brake was stowed and the flight continued and landed without further incident.

4.35.2 Cause

According to Ref. 4.38 inspection of the aircraft revealed that the left wing number three flight spoiler “up” cable had failed at a pulley in the left wheel well at Wing Buttock Line (WBL) 73.00. The failure was due to corrosion as evidenced by rust deposits at the failure location. During rectification, all other left wing spoiler cables were replaced due to evidence of minor corrosion. Following repair the airplane was released to service.

The problem was traced to misrouted cables that operate the left wing spoilers. The maintenance engineers involved had traveled from Brisbane to Sydney that day and had worked a period in excess of 24 hours with minimal breaks. Excessive hours worked and fatigue of the maintenance engineers was considered to have contributed to the misrouting of the cables and the failure to detect the misrouting during a duplicate inspection of the spoiler control system.

4.35.3 Solution

Tighter regulation of work rules were instigated to prevent a re-occurrence.

4.35.4 Lesson

For design engineers this suggests that locating cables and pulleys for flight controls in wheel wells is probably not a good idea. Wheel wells are highly susceptible to ingress of rain, slush and mud and are bound to be corrosive.

4.36 Water Leaks do it Again

4.36.1 Problem

In August of 2001 a Quantas Airlines Airbus A330-341 (Figure 4.48) was on its way to Melbourne, Australia when, according to Ref. 4.39 the following scenario unfolded.

Figure 4.48 Model of Airbus A330 (Courtesy geminijets.com)

During the initial descent into Melbourne, the crew configured the auto flight system to the approach mode. That action armed the auto flight system localizer and glide-slope modes for the runway 16 instrument landing system (ILS). It also permitted the crew to engage the second autopilot for the approach. As the aircraft descended through 2,500 ft, the crew placed the ground spoiler handle to the armed position. Shortly after, the radio altimeter indications disappeared from both pilots' electronic flight instrument displays. Both autopilots then disengaged. About 20 seconds later, both flight directors disengaged from the localizer and glide-slope modes but re-engaged in the basic modes of current vertical speed and heading.
The pilot in command (PIC) elected to continue the approach and to manually fly the aircraft, because he considered that he would be able to control the aircraft without the auto flight system approach commands or radio altimeter information. The auto-thrust was unaffected by the disengagement of the autopilots, and remained engaged.

At the completion of the landing approach, the PIC flared the aircraft for the landing, and retarded both thrust levers, which disengaged the auto-thrust system. The aircraft landed on the left and right main landing gears, bounced, and became airborne for 4.5 seconds before touching down again on both main landing gears. The aircraft bounced again, became airborne for one second, and then touched down for a third time on both main landing gears. The right main gear then lifted off the runway for about one second. After which the aircraft settled onto both main landing gears. Two seconds later, the thrust levers were advanced to go-around power, and after a further five seconds, the aircraft became airborne again. The nose landing gear remained airborne throughout this sequence. Additionally, the ground spoilers did not deploy, and the thrust reversers did not activate.

The PIC repositioned the airplane for another approach to runway 16. During the second landing, the aircraft again bounced following touchdown, then settled onto the runway. Four seconds later, the ground spoilers deployed; however, the thrust reversers did not activate when selected by the crew. The landing roll-out was completed and the airplane taxied to the gate. There were no injuries.

4.36.2 Cause

To understand the cause it is necessary to review some aspects of the design of the A330 flight control system. The following is taken from Ref. 4.39 with minor editorial changes:

"The aircraft flight control system is of the fly-by-wire type. Three flight control primary computers and two flight control secondary computers control the flight control system. The computers process crew and autopilot inputs to provide appropriate electrical output signals to the hydraulically powered flight control surfaces.
Crew input to the flight control computers is made via electrical signals from either of two side stick controllers, and autopilot input is made via an interface with the aircraft Flight Management and Guidance System (FMGS).
The inputs to the flight control computers are processed in accordance with respective flight control laws. Regardless of the pilot's inputs, the control computers will prevent excessive maneuvers and/or exceedance of the safe flight envelope. The flight control laws are dependent on whether the aircraft is on the ground, in flight or in the flare mode of flight. In the ground mode, there is a direct relationship between the side-stick deflection and the flight control surfaces. In the flight mode, deflection of the flight control surfaces is governed to achieve a

load factor proportional to side-stick deflection, independent of speed. The flight mode provides 3-axis control of the aircraft, and provides flight envelope protection and maneuver load alleviation.

In the flight mode the normal laws are:

- Nz law for pitch control, including load factor protection;
- Lateral normal law for lateral (roll and yaw) control, including bank angle protection;
- Protection against high speed, pitch angle and stall (angle of attack).

In the flare mode, the normal laws are:

- Flare law in place of Nz law for pitch control to allow for conventional flare;
- Lateral normal law for lateral (roll and yaw) control including bank angle protection;
- Protection against stall.

The flare mode permits pilots to use the same landing technique as for non-fly-by-wire aircraft. Transition from the flight mode to the flare mode occurs when the radio altimeters sense that the aircraft is less than 100 ft above the ground.

If faults are detected in both radio altimeters, switching from the flight mode to the flare mode will occur when the landing gear is extended, provided the autopilot is off. If the autopilot is engaged, switching from the flight mode to the flare mode occurs when the autopilot is engaged, provided the landing gear is extended.

Airbus reported that flight tests for the A330 included landing in the flight mode, i.e. without transition to the flare mode. Landing in that condition was not considered difficult, however, it required a different handling technique than would otherwise apply for non-fly-by-wire aircraft. In such circumstances, a pilot would need to apply back pressure on the side-stick to initiate the landing flare, then release that back pressure to maintain the desired pitch attitude until touchdown.

The A330 is equipped with two radio altimeters that provide information about the aircraft height above ground level. Data from the radio altimeters are also used by many of the airplane systems' logic sequences to determine whether certain operating parameters have been met to permit operation of a particular system.

The radio altimeter antennas are located along the keel of the aft fuselage of the airplane. They are connected to the aircraft electronic system by coaxial cables. Inspection of the radio altimeter system antennas subsequent to the occurrence of this incident revealed that they had sustained water ingress at the antenna coaxial cables. The water ingress into the radio altimeter antennas resulted in the radio altimeter signals being interpreted as out of range signals, rather than as a failure of the radio altimeters.

During the period from June 11, 2001 to the date of the occurrence (August 27, 2001), there were 19 entries in the maintenance log of the airplane reporting problems with the radio altimeters. Repairs had been carried out on the radio altimeters, including replacement of a transceiver unit and cleaning of components due to water ingress.

(Note from the author: these maintenance events might have served as precursors to persons who completely understand the workings of the system.)

The airplane is equipped with autoflight and flight director systems. Radio altitude signals from the radio altimeters are used to engage the autoflight system into the LAND mode when the airplane altitude is 400 ft above ground level. The loss of valid radio altimeter signals in the LAND mode results in the loss of both autopilots and the flight directors reverting to the basic modes of vertical speed and heading. The autopilot also uses radio altitude signals to adapt autopilot gains during an ILS approach, with the required gain being dependent on the distance of the airplane from the runway threshold. Any involuntary disconnection of the autopilot triggers an AP OFF INVOLUNTARY warning message to the crew.

The airplane is equipped with wing mounted ground spoilers. The ground spoilers arm when the crew places the speed brake control lever to the armed position, and these ground spoilers will activate after landing provided certain parameters are met. Those parameters include both main landing gears transitioning from flight to ground (weight on wheels), and a radio altitude of less than 6 ft or a wheel speed higher than 72 kts on the front and rear wheels of the main landing gears.

The airplane is equipped with an Allied Signal solid state digital flight data recorder. The recorded data were examined and revealed that each of the flight control primary and secondary computers had operated normally throughout the flight. The recorded data revealed that during both approaches, the autopilots oscillated in the lateral and longitudinal axes.

Both autopilots disconnected simultaneously, but an AP OFF INVOLUNTARILY warning did not accompany the disconnection. The LAND mode engaged at 400 ft radio altitude. One second later, both flight directors disengaged from the localizer and glide-slope modes, then re-engaged in the basic modes of current vertical speed and heading. The recorded data also revealed that the signals from both radio altimeters were invalid throughout most of both approach sequences into Melbourne."

Analysis of the incident according to Ref. 4.39: "During the first landing, the nose-wheel remained airborne throughout the landing sequence, indicating that de-rotation did not occur. Consequently, the front wheels of the main landing gears probably did not contact the ground for a sufficient period to allow them to accelerate to the required wheel speed condition. That resulted in the logic conditions for ground spoiler deployment not being met. Those required compression of both left and right main landing gears (weight on wheels), and a radio altitude of less than 6 ft or a wheel speed greater than 72 kts on the front and aft wheels of the main landing gears. Without a valid radio altitude signal of less than 6 ft, and without the ground spoilers deployed, the logic conditions for reverse thrust were also not met, and it too was unavailable. The absence of an AP OFF INVOLUNTARY warning indicated that the crew had intentionally disconnected the autopilot during the approach.

The loss of valid radio altimeter signals did not result in the automatic switching from the flight mode to the flare mode when the autopilots disengaged. That was due to the water ingress into the radio altimeter antennas, which resulted in the radio altimeter signals being interpreted as out of range signals, rather than as a failure of the radio altimeters.

4.36.3 Solution

The water ingress was cleaned up and the system repaired.

4.36.4 Lessons

1. With so many items that can go wrong in many different ways it is not reasonable to expect pilots to fully comprehend what a flight control system is doing or, what it is not capable of doing.

2. Design engineers have to consider ALL items that can malfunction for ANY reason in complex, automated systems.

3. Water (or in general, liquid) ingress continues to be a problem which can only be prevented by good initial design practices.

4.37 Uncommanded Yaw

4.37.1 Problem

In November of 2001, a Quantas Airways Boeing 747SP-38 (Figure 4.49) flying at 43,000 ft near Moomba, Australia experienced an abrupt yaw to the right followed by a 20 degree bank. The pilots were able to isolate the problem and landed without incident at their destination.

Figure 4.49 Boeing 747SP-38 (Not accident aircraft, Courtesy Frank C. Duarte Jr.)

4.37.2 Cause

The following is taken from Ref. 4.40. When the incident occurred the airplane was flying with autopilot A engaged and the lower and upper rudder yaw dampers engaged. The pilot disengaged the autopilot and stabilized the airplane. Then the uncommanded yaw occurred again. The crew then broadcast a PAN (radio code indicating uncertainty or alert, not yet the level of a Mayday) and received authorization to descend to 38,000 ft.

At that point the rudder position indicators showed an upper rudder displacement of 5 degrees right and a lower rudder displacement of 0 degrees. The flight crew then began activating and de-activating the upper and lower yaw damper switches in an attempt to isolate the problem. During these actions the aircraft began to “Dutch roll”. The crew then isolated the problem to the upper rudder yaw damper which was switched off. Normal operations ensued and the airplane continued its flight without further incident.

4.37.3 Solution

Ground maintenance personnel confirmed an anomaly in the upper rudder yaw damper and the unit was replaced with the airplane returned to service.

4.37.4 Lessons

1. Control surface position indicators in the cockpit should be mandatory: they can be very helpful to pilots in an emergency

2. This occurrence argues for two independent rudder segments instead of the one-piece rudder found on many other transport aircraft.

4.38 Routing Control Cables past Engine Burst Planes

4.38.1 Problem and Solution

During 2001 the Beechcraft (Raytheon) Premier I business jet (Figure 4.50) with the airplane close to its first flight, it was discovered that the longitudinal control system cables were routed in such a manner that a fan- or turbine-disk burst would probably cut all control cables. The FAA refused to certify the airplane. The result was a rather major redesign and remanufacturing effort which caused a lengthy delay in the certification of the airplane.

Figure 4.50 Beechcraft Premier I (Courtesy Wernar Horvath)

4.38.2 Lesson

Design management should conduct frequent critical design reviews with experienced engineers present. Had such a review been conducted before manufacturing had started, it would have saved the company a lot of money and calendar time.

Chapter 5

Lessons Drawn from Engine Installation Design

"If reverse thrust can be commanded in flight, it will be"

Variation of Murphy's Law

5.1 Introduction

In this chapter a series of problems which arose in engine installation design are reviewed. Where applicable, causes and solutions are described and lessons learned are stated. Connections with other areas of design are identified in Appendix A.

Recurring events are propeller blade separation, thrust reverser deployment when not desired and uncontained engine failure.

In the case of propeller blade separation and uncontained engine failure the cause in the background is always some type of metal fatigue that went undetected for various reasons.

In the case of uncommanded thrust reverser deployment the reaction time of the crew becomes critical. The latter can be eliminated as a factor by equipping airplanes with control systems which react to such events automatically.

5.2 Propeller too Large or Landing Gear too Short

5.2.1 Problem

In November of 1946 the prototype of the SAAB Scandia (Figure 5.1) took to the air.

Figure 5.1 SAAB Scandia (Not accident aircraft, Courtesy M. Lawrence)

The Scandia represented the first attempt by SAAB to enter the commercial transport market. It was quickly discovered that the propeller ground clearance with a soft main landing gear tire and a deflated strut was insufficient. Figure 5.2 illustrates the problem which should have been evident during early layout design.

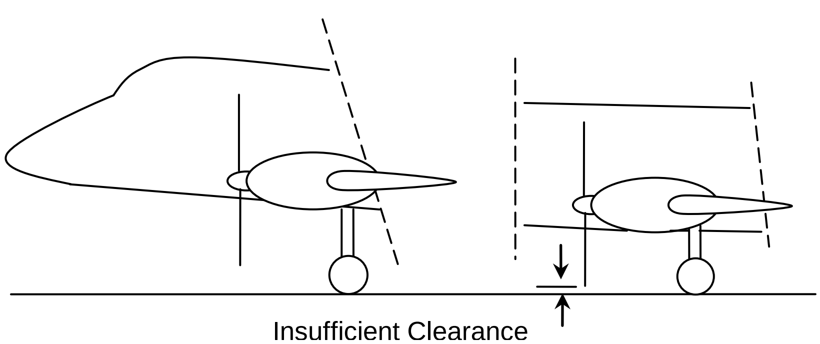


Figure 5.2 Insufficient Propeller Clearance

5.2.2 Solution

A rather extensive redesign effort was launched to raise the engine thrust line by six inches and to raise the nose gear by four inches (Ref. 5.1).

5.2.3 Lesson

Designers should always consider the interactive effect of propeller diameter (for a growth version with a growth engine/propeller combination), landing gear position with deflated tires and struts and minimum required ground clearance. Doing this in the early design phase will save calendar time and money downstream.

5.3 Propeller Reversal in Flight I

5.3.1 Problem

In August of 1949 a Northeast Airlines Convair 240 (Figure 5.3) made an extremely hard landing in Portland Maine following an uncommanded propeller reversal while the throttles were retarded just before touchdown.

Figure 5.3 Convair 240 (Not accident aircraft, Courtesy www.prop-liners.com and Northeast Airlines)

Major structural damage occurred followed by a fire which destroyed the airplane. The crew of three and all 25 passengers were evacuated without injuries.

5.3.2 Cause

According to Ref. 5.2 the probable cause was: “failure of the throttle locking device to function properly thus permitting the movement of the throttles beyond the stop into the reverse propeller position.”

The following material is adapted from Ref. 5.2. “The propeller reversing pitch system includes an electrical switch located on the structure of the left main landing gear. A relative movement of the landing wheel strut of approximately one-half inch, resulting from the airplane weight upon the wheels, closes this switch which energizes a solenoid. This in turn unlocks the throttle reversing mechanism thus permitting rearward movement of the throttles into the reverse propeller pitch position.

The throttle lock on the reversing mechanism can also be manually operated from the cockpit. This is accomplished by withdrawing about one inch a “T” handle manual override control, which is conveniently located on the cockpit control pedestal. This control is interconnected mechanically to the solenoid plunger and its outward movement has the same effect on the throttle lock as does the energizing of the solenoid.

The solenoid plunger and manual override control mechanism is spring loaded to return to its original position when the solenoid is de-energized. In this position the propeller pitch reversing mechanism is locked. This prevents the throttle being unintentionally moved into the pitch reversing range while the airplane is airborne. Investigation revealed that in this instance, due to improper adjustment of the solenoid plunger travel, there resulted a mechanical binding of the plunger in the coil bore which, plus the effect of residual magnetism, left the throttles free to be retarded past the idle detent position into the reverse pitch range. This occurred when the throttles were closed preparatory to landing at Portland.

The manual override handles are conveniently located on the control pedestal, but with the usual cockpit lighting their position is not readily discernible at night.”

5.3.3 Solution

Northeast Airlines added a positive determination of the position of this control to the “before landing” checklist. Shortly after, the CAB issued an AD making such a check mandatory on all Convair 240 airplanes equipped with activated reversing propellers.

5.3.4 Lesson

A mis-adjustment of a safety device almost caused a disaster. Design engineers might keep this in mind when designing the next safety device.

5.4 Propeller Reversal in Flight II

5.4.1 Problem

In October of 1950 A Northwest Airlines Martin 202 (Figure 5.4), while on a training flight, crashed in Almelund, Minnesota.

Figure 5.4 Martin 202 (Not accident aircraft, Courtesy Royal Aeronautical Society)

All six occupants were killed and the airplane was demolished.

5.4.2 Cause

Ref. 5.3 states as the probable cause the uncommanded reversal of the right propeller in flight as a result of which the pilots were unable to control the airplane.

The right propeller blades were found in a 7 to 10 degrees reversed pitch. According to Ref. 5.3 the cause of the propeller reversal could not be definitely established.

During a test flight early in the Martin 202 program, a propeller reversed in flight because of an error made in the installation. The test pilot had great difficulty in controlling the airplane and using full aileron and almost full rudder was able to make an emergency landing in a field short of the Martin airport. This experience clearly showed the control problem that would arise if a reversal occurs in flight.

5.4.3 Lesson

It would not be feasible to assure controllability if propeller reversal should occur at maximum power during take-off. However, in flight situations such as cruise or landing the reversal of one propeller should not result in the loss of control.

5.5 Propeller Reversal in Flight III

5.5.1 Problem

In February of 1952 a National Airlines Douglas DC-6 (Figure 5.5) crashed and burned shortly after take-off from the Newark Airport in New Jersey.

Of the 63 persons on board three crew members and 26 passengers were killed. The remaining passengers and crew members suffered minor to serious injuries.

Figure 5.5 Douglas DC-6 (Not accident aircraft, Courtesy David Schulman)

5.5.2 Cause

All engines were operating when the crash occurred. Propellers No. 1 and 2 were in the 46 to 53 degrees positive pitch range as they should have been. Propeller 3 was in full reverse pitch and propeller No. 4 was found fully feathered.

According to Ref. 5.4 an examination of airplane maintenance records showed that in January and February of 1952 the red flag indicating that the propellers could be reversed came up and stayed up after take-off. This flag should have dropped back out of sight when the wheel left the ground. In both cases the problem was corrected by replacing the micro-switches located on the nose-wheel and right main landing gear.

On January 24, 1952, during a maintenance run-up check, it was found that the No.4 propeller would go into reverse pitch when being taken out of the feathered position. This propeller was removed and sent to the propeller overhaul shop where it was found that moisture between the slip ring assembly and the contact plate was causing the trouble. The slip ring assembly was replaced which corrected the difficulty. This propeller was then installed as the No. 3 propeller on the accident aircraft.

The Board determines the probable cause of this accident to be the reversal in flight of the No. 3 propeller with relatively high power and the feathering of the No. 4 propeller. This resulted in a descent at an altitude too low to effect recovery.

The following is quoted from Ref. 5.4: "There was uncertainty as to what caused the reversal of the No.3 propeller. It was determined however that the propeller governor solenoid valve circuit, which extends from the cockpit to the governor on the nose of the engine and which was not isolated from other circuits, will cause reversal of the propeller if it should become energized. Should this occur, due to some fault in the electrical system, resulting in unwanted voltage to the governor solenoid circuit, reversal of the propeller would result without any action on the part of the crew and as long as the circuit remained energized, the propeller could not be taken out of the reverse pitch position.

On February 14, 1952, the Administrator of Civil Aeronautics sent to all CAA regional offices the following telegram: "……to preclude possibility of inadvertent propeller reversal of Hamilton Standard propellers on the DC-6, DC-6A and DC-6B aircraft the wiring from the engine firewall to the governor solenoid valve is to be isolated from all other circuits to prevent inadvertent application of electric power to the solenoid circuit. This is to be accomplished preferably by removing wire from any bundles in which it may run and placing it in separate isolated conduit. Isolation of this portion of circuit is to be accomplished as soon as possible but not later than midnight February 18. Portion of circuit behind firewall and throughout remainder of aircraft to be inspected immediately. Inspection to include check of all terminal points to assure no hazard of contact with loose wires nearby and check of all points where chaffing or other damage may occur which could permit energized wires to contact solenoid circuit wire or terminals. Further instructions regarding isolation of portion of circuit behind firewall will be transmitted as soon as available. We do not recommend de-activation of reversing propellers on any aircraft while above program being accomplished."

5.5.3 Solution

On February 13 National Airlines nevertheless began a program of rendering the propeller reversing feature inactive on all their DC-6 equipment.

5.5.4 Lesson

Electrical circuits to flight crucial functions must be isolated.

5.6 Exhaust Fairing I

5.6.1 Problem and Solution

In 1954, during early flight testing of the Douglas A4D Skyhawk (Figure 5.6) severe buffeting was noticed by the test pilot (Ref. 5.5).

The problem was traced to massive flow separation between the exhaust and the lower part of the vertical tail. The solution was the addition of a so-called "sugar-scoop" as shown in the "before-after" pictures of Figure 5.7.

Figure 5.6 Douglas A4D Skyhawk (Courtesy Royal Aeronautical Society Library)

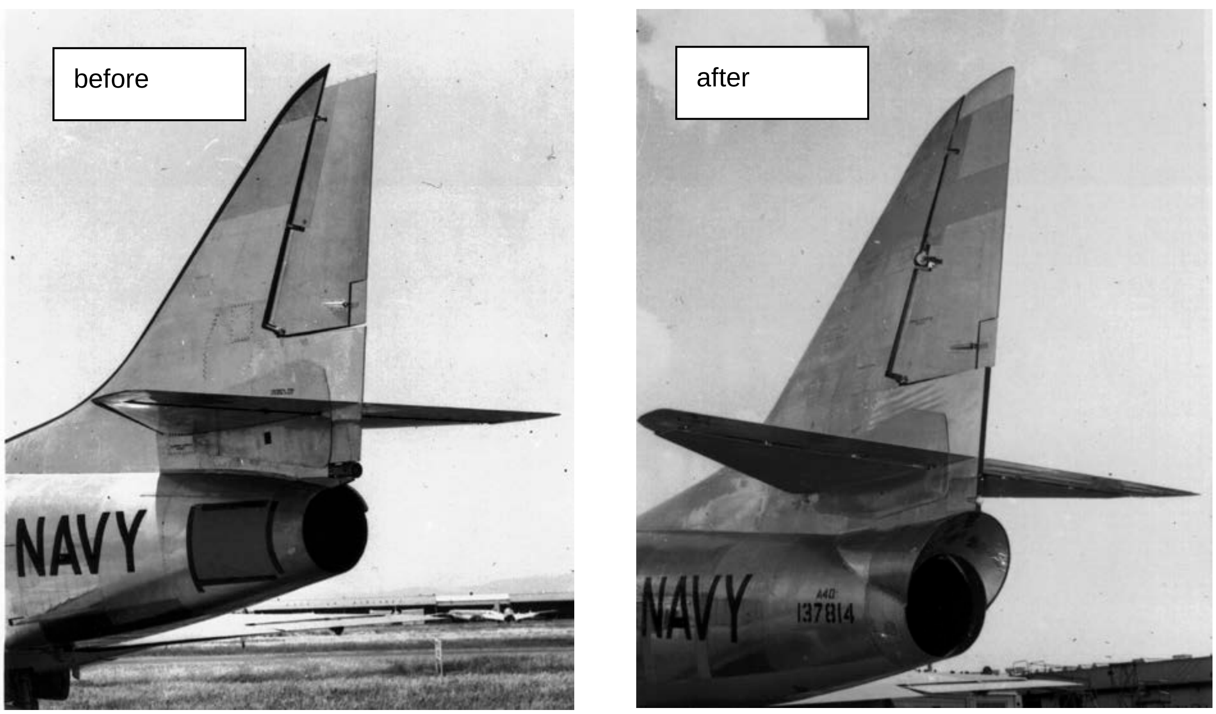


Figure 5.7 Sugar-scoop fairing as installed on the Douglas A4D Skyhawk (Courtesy San Diego Aerospace Museum)

5.6.2 Lesson

Engine exhausts can have a significant impact on local flow stability. Designers should be aware of this.

The problem can usually be discovered during wind tunnel testing. Recently, by using proper CFD techniques areas of potential flow separation can be identified even before going in the tunnel. This can save both money and calendar time.

5.7 Propeller Reversal in Flight IV

5.7.1 Problem

In April of 1955 a United Airlines Douglas DC-6 (Figure 5.8), N37512, on a training flight crashed on MacArthur Field shortly after becoming airborne. The two pilots and a UAL flight manager were killed. The aircraft was destroyed by ground impact and fire.

Figure 5.8 Douglas DC-6 (Not accident aircraft, Courtesy www.prop-liners.com and United Airlines)

5.7.2 Cause

Ref. 5.6 states the probable cause to be: "the unintentional movement of the No. 4 throttle into the reverse range just before breaking ground, with the other three engines operating at high power output, which resulted in the aircraft very quickly becoming uncontrollable once airborne."

A review of the manner by which propeller reversal is commanded in this airplane is presented by quoting from Ref. 5.6:

"The propellers of the DC-6 airplane may be used to provide reverse thrust for braking while the aircraft is on the ground. Propeller reversal is initiated by retarding the throttles aft of the forward idle position at which time an electrical control system is activated causing the blades of the propellers to rotate within their hubs to a position wherein reverse thrust is developed. The extent of engine power and reverse thrust developed is in proportion to the extent of rearward throttle movement. The propellers are unreversed and forward thrust is restored by returning the throttles to the forward idle position or beyond.

While the aircraft is airborne a throttle latch mechanism prevents inadvertent throttle movement aft of the forward idle position and thus prevents unwanted reversal. Operation of the throttle latch is controlled by switches, on the landing gear struts, that close when the aircraft's weight is on the landing gear. This action energizes a solenoid which in turn releases the throttle latch. At the same time the reverse warning flag swings up into view on the control pedestal to show that the latch is out of the way. Mechanically linked to the solenoid, this red metal flag may be raised manually by the crew to operate the latch should the solenoid fail to operate.

When the aircraft becomes airborne the strut switches open and the solenoid becomes de-energized. The latch returns to the locked position and the flag swings out of sight.

Approximately three years ago United Air Lines, concerned over the possibility of an unwanted in-flight propeller reversal due to an electrical malfunction, modified the propeller control circuits of its DC-6 fleet. This modification results in the automatic removal of electrical power from the circuits controlling propeller reversal whenever the aircraft is airborne. Electrical power is restored to these circuits when the aircraft is on the ground. Removal and restoration of electrical power is accomplished automatically through the addition of a relay (known as the H-

relay) controlled by switches which are in turn actuated by the throttle latch solenoid. The propeller control circuit of the subject aircraft had been so modified.
Investigation disclosed that once a propeller starts into reverse position it need not cycle completely but can be unreversed from any negative blade angle. Should the propeller become reversed due to movement of the throttle rearward past the forward idle position, while the aircraft is on the take-off run and, should the aircraft then become airborne in this configuration, the propeller may be unreversed by (1) feathering or (2) lifting the reverse warning flag and advancing the throttle. Raising the flag serves the same function as the landing gear switch when the aircraft is on the ground; i.e. the reverse control system of the propeller is again energized permitting unreversal to take place. If the flag is not lifted when the throttle is moved forward, the blades will remain in reverse pitch and the amount of reverse thrust developed will depend upon the amount of throttle applied.

Over the years during which propeller reversing systems have been in use on air transport aircraft, UAL has conducted numerous tests to determine aircraft flight characteristics with various combinations of forward and reverse propeller thrusts. For the most part, these earlier tests were conducted at cruise airspeeds and with cruise power settings, since the greater interest was associated with the effect of an unwanted propeller reversal on the aircraft performance and controllability while in level flight. These flight tests were extremely beneficial to the industry as a whole, and provided needed information relative to procedures to be followed should an unwanted reversal occur while in level flight. Within a few days following this accident, UAL conducted another series of flight tests to further investigate, among other things, the effects of a reversed outboard propeller upon the handling characteristics of a DC-6 at low airspeeds.

These tests indicated, among other things, that in the take-off configuration with METO power or higher on No. 1, No. 2, and No. 3 engines, the aircraft almost immediately became uncontrollable when full power was applied in reverse on No. 4 engine and the aircraft speed was 100 kts or less. In this test the roll was delayed for a short time by using full opposite aileron. The violent yawing continued, however, with an attendant loss of airspeed, and within a few seconds a violent roll and pitch developed. The resulting aircraft maneuver closely approximated the maneuver which N37512 made.

One of the most significant points developed during these tests related to the positioning of the throttle following unintentional displacement of the throttle into the reverse range. The tests confirmed the fact that if the throttle is moved into the reverse range during a take-off run, moving the throttle back into the forward thrust range after becoming airborne will not bring the propeller out of reverse but will only result in increased thrust power. This follows since, as described earlier, the reversing circuitry is de-energized upon becoming airborne, and the propeller remains in the reverse range, in which position it was placed while on the ground. Unreversing can only be accomplished under this condition by depressing the feathering button or by raising the reverse warning flag and advancing the throttle."

5.7.3 Solution

Following acquisition of DC-7 equipment and favorable operating experience with the sequence gate latch (also called the Martin bar) on those aircraft, UAL decided to equip its DC-6 and DC-6B aircraft with the device. In principle, it consists of a bar placed across the throttle at the idle position. It may be moved out of the way by the pilot when he wishes to pull the throttles back into reverse; when in position, it is impossible to pull the throttles into reverse. Orders were placed for the Martin bar kits several months prior to this accident and the first DC-6 was modified a week before the accident occurred.

5.7.4 Lesson

The scenario of events leading to this unwanted propeller reversal was predictable. It seems that the Martin bar was an obvious solution to a looming problem.

5.8 Propeller to Fine Pitch during Approach

5.8.1 Problem

In February of 1956 a Capital Airlines Vickers Viscount (Figure 5.9) crashed during the final part of a landing approach at Midway Airport, Chicago, IL.

The five crew members and 37 passengers were evacuated, a few with minor injuries.

Figure 5.9 Vickers Viscount (Not accident aircraft, Courtesy www.prop-liners.com and Capital Airlines)

5.8.2 Cause

According to Ref. 5.8 the probable cause was: "a malfunctioning of the propeller control switches which culminated in an abrupt loss of lift."

The following has been adapted from Ref. 5.8.

The pilot flying the airplane testified that when the flaps were moved to the 47 degree position just prior to the flare he saw that three of the four 17-degree propeller pitch lights were lighted. These lights are actuated by a blade switch on each of the four propellers when the blades are at 17 degrees or below and warns the pilot that the blades are below the minimum in-flight angle. The pilot immediately moved the throttles forward but the airplane settled onto the ground upon which the throttles were closed.

The propellers were equipped with pitch lock solenoids which are energized by the closing of switches caused by the telescoping action of the landing gear upon landing.

The switch mounted on the right main landing gear was found to contain water in the switch housing, including the contact cavity, and showed evidence of corrosion. The switch operated freely; however, a considerable amount of corrosion products in granular form were loose inside the switch housing. Significant with respect to the water found in this switch housing is that the aircraft was exposed to a 19 degree F temperature prior to take-off for Chicago.

Checks of all other switches revealed that one was stuck in the closed position.

To simulate what happened and at the request of the CAB the manufacturer conducted a flight test with all propeller stops withdrawn. At an airspeed of 100 kts and about 8 ft above the ground all throttles were quickly opened about half throttle distance. When this was done, there appeared to be a complete loss of lift and the aircraft sank rapidly to the runway.

The circuit of the 21-degree pitch lock solenoid contains four micro-switches, two connected in parallel on the positive side and two connected in parallel on the negative side. This necessitates that one switch on each side of the circuit be electrically conductive before the solenoid is energized, thus completing one of the steps toward withdrawal of the 21-degree pitch stops. This circuit is designed expressly as a safety measure in that malfunctioning of two switches is required to establish an unwanted circuit. However, this double failure feature was compromised in that a failure of one switch could go undetected for an indeterminable period of time. No specific inspection period had been established for these switches.

It is of interest to review the characteristics of the power plants on the Viscount.

Power control consists of four throttles which schedule r.p.m. and fuel flow for each engine. The propeller response to the signal for higher r.p.m. is more rapid than the engine response to increase power to maintain this r.p.m. This is a normal turbine propeller characteristic and the lag of the Rolls Royce Dart engine is considered to be acceptable. A number of variables, such as airspeed and rate and extent of throttle movement would affect the duration of this lag. In this instance, it is believed the lag was about 2.5 seconds. During a major portion of this period, the propeller blades would be at four degrees attempting to maintain the higher r.p.m. called for, through wind-milling action which would greatly increase drag and decrease lift.

5.8.3 Solutions

Immediate corrective actions were taken as follows:

- a dual, 21 degree pitch lock solenoid warning light was installed on all Capital Airlines Viscounts.
- a 300-hour periodic check of all micro-switches was implemented. This requires their removal and installation of newly overhauled switches.
- a hole was drilled in each micro-switch case to allow excess moisture to drain from the switch.

- a test circuit was installed in all Capital Airlines Viscount aircraft consisting of a dual light and single pole double throw switch which provides a means to check, while in flight, the positive and negative sides of the 21-degree pitch solenoid circuits to determine if the micro-switches are malfunctioning.
- the 21-degree pitch lock warning lights were duplicated on the fire control panel of the co-pilot.
- hermetically sealed landing gear actuated micro-switches were ordered and are to be installed upon delivery

5.8.4 Lesson

Moisture ingress should be prevented in all electrical components of an airplane. The scenario in this accident was predictable and the flight tests carried out after the accident should have been carried out as part of the certification of the airplane.

5.9 Design for Engine Removal

5.9.1 Problem

This problem was taken from Ref. 5.8. In the Cessna T-37 (Figure 5.10) the USAF had insisted that it be possible to lower an engine vertically out of each nacelle onto a maintenance cart.

To accomplish this, the lower spar cap in the front spar of each wing had a swing-link designed into it. Figure 5.11 shows the general arrangement.

In the fall of 1956 a demonstration had to be given to USAF personnel to show that engine replacement could be done within a short period of time. Nobody had given any thought to a dress rehearsal. The day of the demonstration came and turned out to be very embarrassing. A crowd of important USAF watchers were right there.

Figure 5.10 Cessna T-37 on the Production Line in 1956 (Courtesy Cessna)

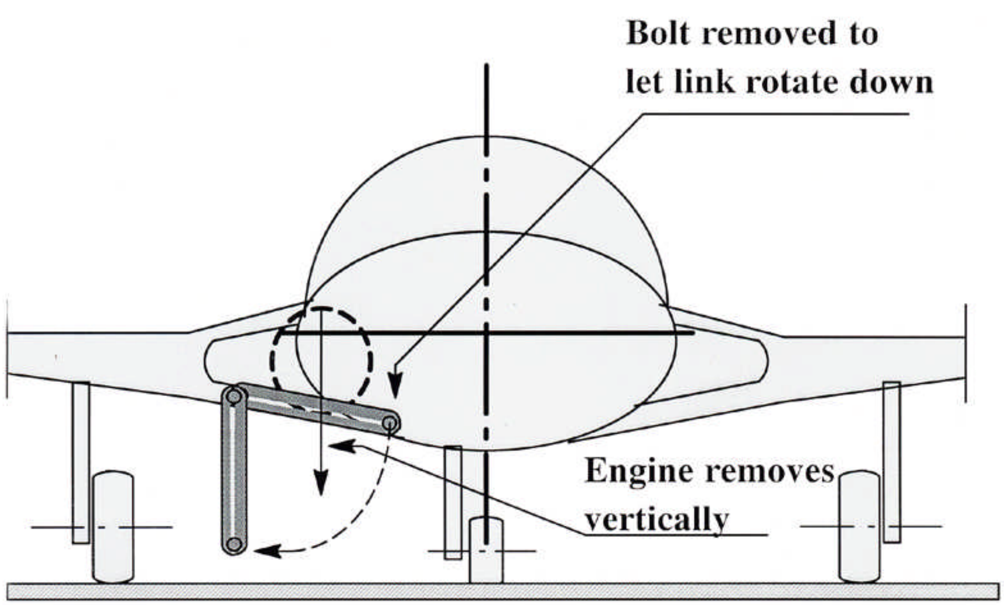


Figure 5.11 Swing Link Installation in the Front Spar of the T-37

The bolt allowing the swing-link to be moved downward was duly removed. The link swung down. The engine fittings were disconnected. The engine was lowered onto its cart and then moved away. Another cart with another engine was moved into position. The engine was raised into the nacelle. All fittings were re-attached. Someone swung the swing link upward and... oops, the bolt holes were no longer aligned.

5.9.2 Cause

Now look at the airplane, sitting on its landing gear in Figure 5.11. It should have been obvious that when the lower spar link is taken away there will be just enough deformation (because of bending) to prevent bolt-hole realignment.

5.9.3 Solution

The spars could be "beefed up" to increase the bending stiffness of the wing. This solution obviously adds weight. The problem should have been recognized in the early design phase and either the engine should have been shifted aft or the front spar should have been moved forward to allow for adequate engine clearance.

5.9.4 Lesson

It is usually not a good idea to have to remove primary structural components to exchange engines. It is also not a good idea to arrange for customer demonstrations without a dress rehearsal.

5.10 Flame-out due to Engine Mount Compliance

5.10.1 Problem

Sometime during 1957, after the Cessna T-37 (Figure 5.12) had been in service for a while, reports from the field indicated that during typical fighter approach maneuvers one or more engines would frequently flame-out (Ref. 5.9). This was clearly unacceptable.

Figure 5.12 Cessna T-37 (Not incident aircraft, Courtesy NASA)

5.10.2 Cause

A detailed investigation revealed that the engine mounts had elastomeric grommets inside to assist in reducing transmission of vibrations to the airframe. Due to the high temperatures in the engine bays the grommet material had softened and allowed the engines to move relative to the engine hard-points. The engine fuel control system consisted of an airframe mounted cable system. As a result of the engine movement the fuel control levers were put in the shutdown position.

5.10.3 Solution

The problem was solved by switching to a different and longer lasting elastomeric material. Perhaps a better solution would have been to change the fuel control system to an engine mounted system rather than an airframe mounted system.

5.10.4 Lesson

Engine fuel control systems should be mounted on the engine and not on the airframe. Also, any vibration or shock control material should be tested for its characteristic under actual operating (in this case elevated temperatures) conditions.

5.11 Engine Bearing Failure Followed by Propeller Separation

5.11.1 Problem

In March of 1957 an American Airlines Douglas DC-7 (Figure 5.13) lost the nose section and propeller of its No. 1 power-plant while in flight near Memphis, TN.

Figure 5.13 Douglas DC-7 (Not accident aircraft, Courtesy www.prop-liners.com and American Airlines)

Major damage was inflicted to the fuselage followed by explosive decompression. There was no fire. The crew made a successful landing in Memphis where several crew members and passengers were treated for the effect of the sudden de-compression. Five passengers also received lacerations as a result of flying debris.

5.11.2 Cause

The following material was adapted from Ref. 5.10. The probable cause of this accident was failure of the propeller thrust bearing assembly, which resulted in separation of the propeller and subsequent penetration of the fuselage causing explosive decompression of the airplane in flight.

A 17x4 ft section of the forward cabin roof was blown off the airplane resulting in significant debris flying around the passenger cabin. Two lavatory doors were also torn off their hinges and hurled through the cabin. Fortunately no flight control runs were affected.

All of the major components of the front crankcase assembly were recovered with the exception of the roller thrust bearing, a section of the hollow propeller shaft, and part of the nose-cage. Examination of the recovered parts revealed excessive operating temperatures of the propeller shaft where the ball and roller thrust bearings are seated. The shaft which had failed aft of the roller bearing journal was blackened by heat and was necked down under high tensile loading on one side. On the opposite side of the shaft the wall thickness had been increased due to high compression loads.

The ball thrust bearing was severely damaged by over-heating. Its inner and outer races were scuffed and flattened, and contained metal deposits from the melted bronze bearing retainer. All balls were in place but were flattened and blackened by over-heating. The operating temperature of this bearing had reached a point high enough to melt the bronze ball bearing retainer as well as weld the inner races together.

The fit of these bearings at the time of engine assembly is very critical. Engine overhaul records reflect that these components were built up according to manufacturer’s recommended procedures. The last engine overhaul was 346 hours prior to the accident. At that time the ball thrust bearing used was a serviceable unit and the roller thrust bearing was new. The engine itself had a total service time of 6,609 hours which embraced ten overhauls.

The reason for the overheating of the bearing assembly could not be positively established although lack of lubrication is likely.

5.11.3 Solution

Prior to the accident the ball bearing was installed ahead of the roller bearing. This was done to reduce the radial load on the roller bearing. As a result of this accident the engine manufacturer issued service bulletins for the improvement of the inspection of the propeller thrust bearings and for the interchanging of the position of these bearings. The CAA has issued Airworthiness Directive 57-6-4 which covers the same items.

5.11.4 Lesson

In power-plant design the relative location of ball and roller bearings may be important. The design should be such that only one sequence during installation is possible.

5.12 Whirl Mode Flutter

5.12.1 Problem

In 1959 a Lockheed Electra (Figure 5.14) operated by Braniff International Airways crashed near Buffalo, Texas after a structural break-up during cruise flight.

Figure 5.14 Lockheed Electra (Not accident aircraft, Courtesy Bob Garrard)

In 1960 a Northwest Orient Airlines Electra crashed near Cannelton, Ohio after a structural break-up in flight. There were no survivors in either crash.

The remaining Electra airplanes were restricted to a normal operating speed of 225 kts with a never exceed speed of 245 kts.

5.12.2 Cause

After painstaking and prolonged analyses and investigations the probable cause was established to be whirl mode flutter. It was established that, as a result of a hard landing experienced by the Braniff Electra and severe buffeting encountered by the Northwest Electra on previous flights, a structural weakening had occurred in the engine mounting structure. This allowed the frequency of the engine whirl mode to coincide with a wing torsional mode at cruise dynamic pressures. As a result the structure failed in cruise flight and the airplanes broke up. The physics of flutter dictates that one vibration mode extracts energy from the air and excites another vibration mode to the point of dynamic instability. Refs. 5.11-5.13 contain detailed discussion of the accidents and the subsequent investigations.

5.12.3 Solution

After the cause had been established, Lockheed embarked on a major modification program to prevent this from happening again. Figures from Ref. 5.11 show the modifications made to the nacelle and wing structure.

These modifications added about 1,200 lbs to the empty weight of the airplane. Lockheed underwrote the entire cost of the modification program.

As a result the Electra was again declared fully airworthy and the airplanes served with distinction.

5.12.4 Lessons

1. A lesson for aircraft designers is to always ask the question: "what if". When there is a connection between the engine attachment structure and the wing structure, a valid question to be raised in early design is: "Can a hard landing so weaken any part of the structure as to make stiffness assumptions made in flutter analyses invalid?"

2. It is probably a good idea to also ask this question in the case of nose gears which are attached to the fuselage via the engine truss mounts. Many single engine, propeller driven airplanes have this design feature.

3. Another important point. In several turbo-propeller installations the gearbox/propeller is mounted far ahead of the actual engine. As a result the propeller whirl mode in a turbo-propeller installation exhibits rather low frequencies when compared to a conventional piston/propeller installation where the propeller is mounted directly forward of the engine. This becomes important in a flutter analysis.

5.13 Adjacent Engine Installations

5.13.1 Problem

Whenever two jet engines are installed adjacent to each other the issue of uncontained failure of one of the rotating engine components should be considered. Examples of this are the British Vickers VC-10 (Figure 5.15) and the Russian Ilyushin Il-62 (Figure 5.16). The prototype of the VC-10 made its first flight in 1961.

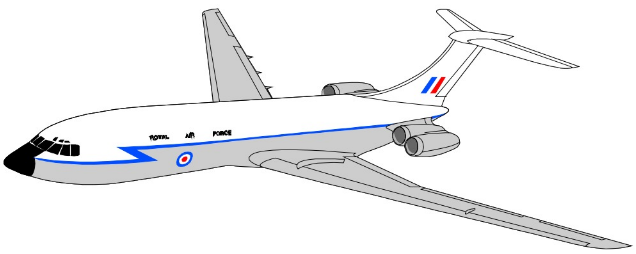


Figure 5.15 Vickers VC-10

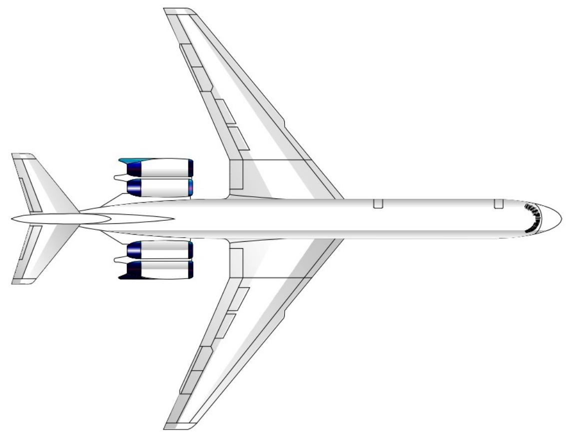


Figure 5.16 Top-view of the Ilyushin Il-62

The VC-10 and the IL-62 both have suffered a number of uncontained engine failures. There were no fatal consequences in the case of the VC-10 because the designers incorporated design features which made catastrophic failures of the fuselage structure or the flight controls extremely improbable events.

In addition, containment features were incorporated in the VC-10 to prevent one exploding engine from doing serious damage to its adjacent counterpart (Ref. 5.14).
Sadly, there have been several instances of uncontained engine failures fatally downing the Il-62 where these provisions were not made.

5.13.2 Lesson

Occurrence of uncontained engine failures should be considered a given. Designers must incorporate features to prevent loss of control or fatal damage to primary structure. However, there are significant weight penalties when this is done. These weight penalties did not help the competitiveness of the VC-10.

5.14 Exhaust Fairing II

5.14.1 Problem and Solution

Designers of jet aircraft are often confronted with unexpected high drag as a result of poor exhaust fairing design.

A typical example of a high drag installation is the original VC-10 exhaust configuration shown in Figure 5.17.

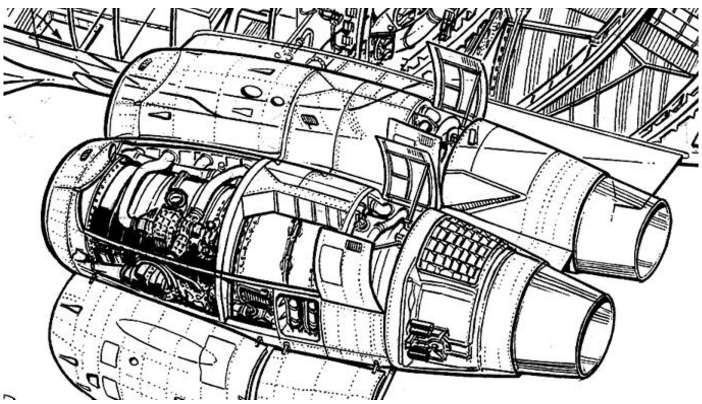


Figure 5.17 *Original Exhaust Configuration (Courtesy The Flight Collection)*

Figure 5.18 Modified Exhaust Configuration (Courtesy J. Hieminga)

Figure 5.18 shows the exhaust fairing, known as the "beavertail", added to the airplane. This "beavertail" added to the rear end in between the exhaust resulted in significantly lower cruise drag (nine drag counts!).

This beavertail was installed in all VC-10 production airplanes since 1962.

5.14.2 Lesson

There is a very significant drag reduction to be had by properly "fairing" the area between engine exhausts. Observe from Figure 5.16 that the Il-62 also incorporated such a beavertail fairing. It will be seen in Section 5.16 that this lesson was not learned in the design of the XB-70.

5.15 Propeller Reversal in Flight V

5.15.1 Problem

In April of 1962 an FAA operated Lockheed Constellation L-749A (Figure 5.19) crashed 220 ft off shore on a coral shelf off the runway at Canton Island (in the Pacific Ocean) during an attempted go-around.

Figure 5.19 Lockheed L-749A Constellation (Not accident aircraft, Courtesy G. Helmer)

The four crew members were fatally injured. There were two passengers on board, one was fatally injured the other seriously. The airplane was seriously damaged, but there was no fire.

The purpose of the flight was to train the co-pilot in various maneuvers and flight configurations prior to being tested for an airline transport rating, and to train a flight maintenance technician as a flight engineer.

5.15.2 Cause

Ref. 5.15 states that the probable cause was loss of control during an attempted go-around following initial touchdown, as the result of an undetected reversal of the No. 4 propeller.

The following material has been adapted from Ref. 5.15. The surviving passenger provided significant information about the flight.

The airplane successfully performed a series of take-off and landing maneuvers as part of the training objectives for this flight. During the last touchdown the airplane rolled 239 ft on the right main landing gear with the right wing continuing to drop. The aircraft then lifted off in a nose-high and right-wing down attitude, and the right wing tip struck the ground at the right edge of the runway. This crushed the right wing tip as well as the outboard portion of the wing and the right aileron. The airplane cart-wheeled and finally came to rest 220 ft off shore in about three feet of water on a coral shelf.

Examination of the four propellers showed that the blade angles at the time of impact were 15 degrees positive, 15 degrees positive, 23 degrees positive, and 20 degrees negative for the Nos. 1,2,3, and 4 propellers respectively. The low pitch stop in this airplane is 15 degrees positive and the reverse pitch stop is 20 degrees negative.

Functional tests and disassembly of the pitch changing mechanism and governors of the four propellers did not reveal any irregularities. There was one exception: the No. 4 governor displayed excessive scoring and pitting of its low pressure relief valve. The cause for this could not be established. However, such a condition can make the low pitch stop ineffective and that is apparently what happened.

With the No.4 propeller in reverse there is added drag and loss of lift on the right side.

5.15.3 Solution

Since the cause for the ineffective low pitch stop could not be established no specific recommendations were forthcoming.

5.15.4 Lesson

Throughout this book reversing of propellers for a variety of reasons is a recurring event. The reader is also referred to Sections 5.21-5.29 in this book. There ought to be a better way to prevent this type of accidents.

5.16 Exhaust Fairing III

5.16.1 Problem

The North-American XB-70A made its first flight in 1964. Figure 5.20 shows a three-view of the airplane. The airplane lacks exhaust fairings between the engines.

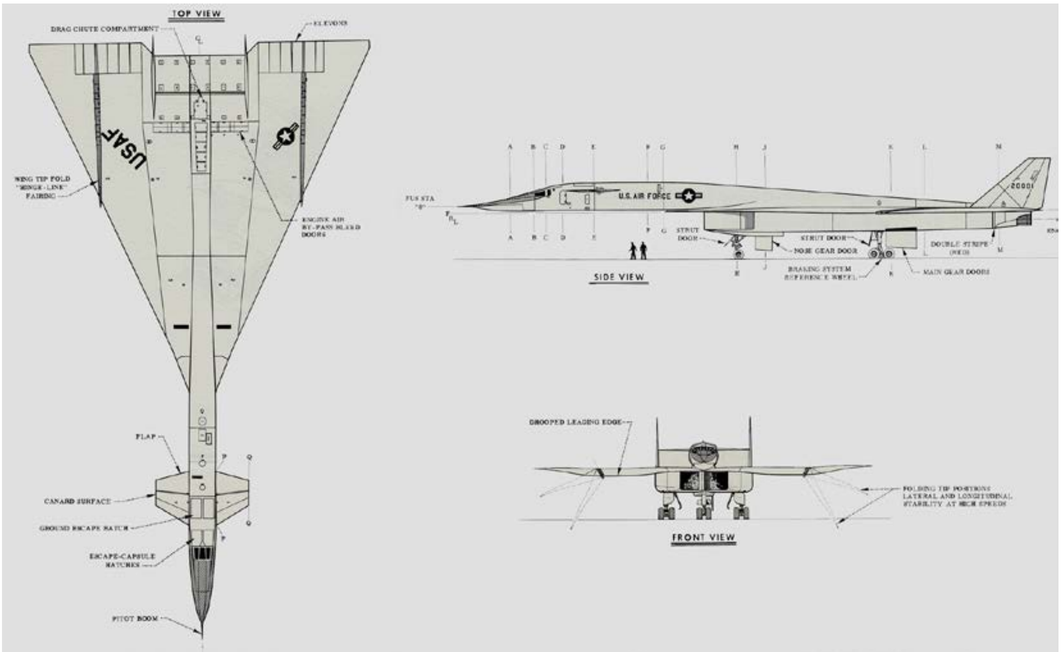


Figure 5.20 North American XB-70A (Courtesy North American)

This lack of exhaust fairings is better evident in Figure 5.20. The drag penalty due to this engine installation in the subsonic regime was very large. A solution was never tested because the entire program was cancelled for other reasons. Figure 5.21 shows a rear end view which gives a good illustration of the large “base areas” associated with this installation. These base areas cause very large pressure drag in subsonic flight.

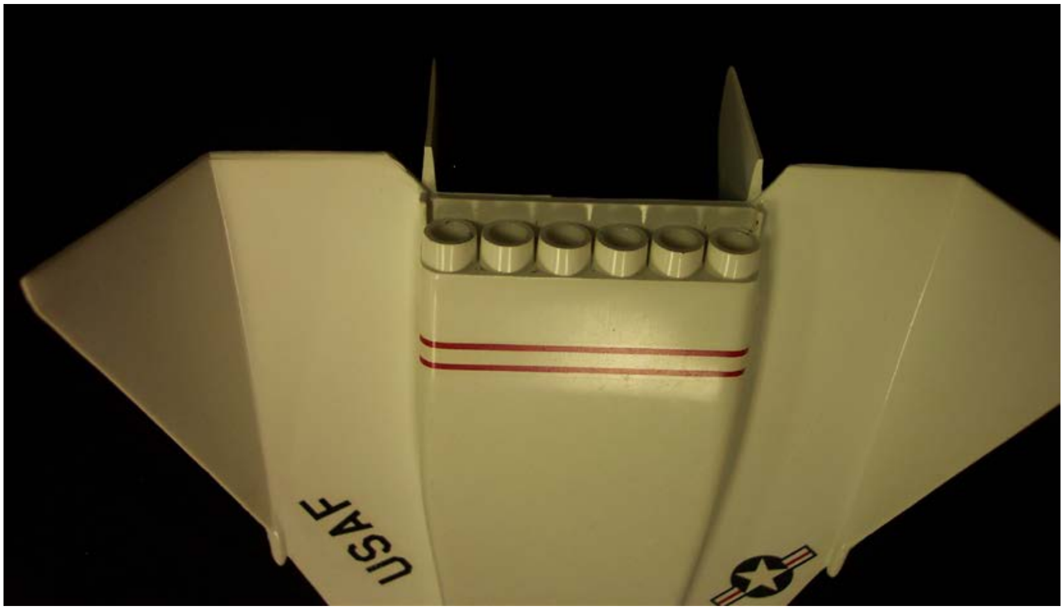


Figure 5.21 Rear Bottom View of Exhaust Installation in an XB-70A Model

5.16.2 Lesson

One more time, in airplanes with engine installations near the back the use of exhaust fairings should be explored early in the design phase.

5.17 Propeller Blade Separation I

5.17.1 Problem

In March of 1967, a Lake Central Airlines Convair 340 (turboprop version) crashed near Marseilles, Ohio. Figure 5.22 shows the piston-propeller version of this airplane. There were 38 people on board and no survivors.

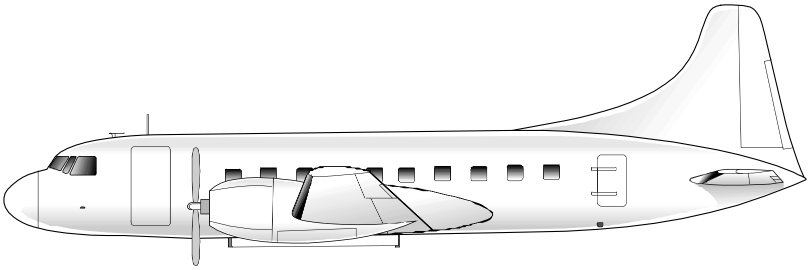


Figure 5.22 Convair 340

5.17.2 Cause

According to Ref. 5.16 the cause was massive propeller blade separation from the No.2 engine which penetrated the fuselage structure to the point were the fuselage failed structurally. The blade failure was initiated when a torque cylinder of the number three blade failed due to fatigue. The resulting separation of the No.3 blade led to a rapid decrease in propeller pitch which could not arrested by the propeller pitch lock. The over-speed condition in turn caused the other blades to separate.

The investigation showed that the torque piston of the No.3 blade had not been nitrided during manufacture and this omission was not detected by inspection.

5.17.3 Solution

The NTSB recommended that inspection procedures at Allison be tightened to prevent a recurrence.

5.17.4 Lesson

Here is an example of one human failure leading to the failure of a blade which in turn led to a rapid blade pitch decrease which could not be arrested by the pitch lock which in turn led to catastrophic structural failure of the airframe. This sequence should have been predicted. Designers must assume that anywhere in the chain humans are involved that someone will fail to do the job. If that means the design then has to be altered to prevent catastrophic failure, that is why one does such a failure analysis.

5.18 Propeller Blade Separation II

5.18.1 Problem

In July of 1975 a propeller blade separated from the No.2 propeller of a DeHavilland DH-114 (Figure 5.23) during the take-off run in San Juan, Puerto Rico.

Figure 5.23 DeHavilland DH-114 Heron (Not accident aircraft, Courtesy Bob Garrard.)

The take-off was discontinued and the airplane brought to a stop. Only one of 11 persons on board was slightly injured.

5.18.2 Cause

According to Ref. 5.17 the blade separated as a result of vibratory stresses which induced fatigue cracks not readily detectable during pre-flight inspections. So far so good.

However, Ref. 5.17 also indicates that the separated propeller blade entered the left side of the fuselage and tore out a portion of the left forward passenger seat, which was not occupied. The airplane flight control system was disabled when the blade exited through the cabin floor. Control cables, electrical wiring, and aircraft plumbing were severed in the lower fuselage area. Fuel and pneumatic lines, which are routed through the lower section of the fuselage, were heavily damaged.

5.18.3 Solution

The NTSB made recommendations regarding improvement of inspection procedures. However, Ref. 5.17 does not address the following design issues:

1. Routing control cables such that one separated propeller blade can render an airplane uncontrollable is not an acceptable design practice.
2. Routing control cables such that one separated propeller blade can seriously damage fuel lines is not an acceptable design practice.

5.18.4 Lesson

When this airplane was originally certified it did not have to meet the more stringent design requirements of today. A thought for aircraft designers: do we really need a requirement to see that this type of design practice is not acceptable? Simply asking the "what if" question with regard to where a separated propeller blade goes should lead designers to the right answer.

5.19 Uncommanded Propeller Blade Pitch Reduction

5.19.1 Problem

In 1991 an Atlantic Southeast Airlines Embraer EMB-120 (Figure 5.24) suddenly rolled to the left until the wings were perpendicular to the ground. It then crashed. There were no survivors.

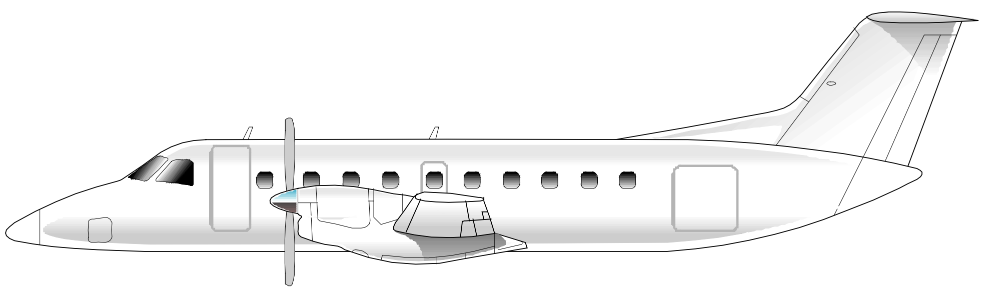


Figure 5.24 Embraer EMB-120

5.19.2 Cause

According to Ref. 5.18 the loss of control was caused by a malfunction of the No.1 propeller control unit, which allowed the propeller blade angles to go below the flight idle position. This increased the drag over the left wing while also lowering the lift over the left wing resulting in an uncontrollable roll to the left.

Detailed analysis of the failed propeller control unit showed extreme wear on the quill spline teeth which normally engaged the titanium-nitrided surface of the propeller transfer tube. It was found that the titanium-nitrided surface was much harder and rougher than the nitrided surfaces of the quill. Therefore, the transfer tube splines acted like a file and caused abnormal wear of the gear teeth on the quill. The investigation found that wear of the quill was not considered during the certification of the propeller system.

In the executive summary of Ref. 5.18 the cause is formulated as follows.

"Examinations of the left propeller components indicated a propeller blade angle of about 3 degrees at impact while the left propeller control unit balls-crew position was consistent with a commanded blade angle of 79.2 degrees. The discrepancy between the actual propeller blade angle and the angle commanded by the screw is a strong indication that there was a discrepancy inside the propeller control unit prior to impact and that the left propeller had achieved an uncommanded low blade angle.

If propeller blade angles suddenly move to a below flight idle position there is a large loss of lift over the wing behind the propeller. There is also an increase in drag over that part of the wing. Simulations and calculations have shown that the resulting rolling and yawing moments cannot be controlled in this airplane.

5.19.3 Solution

The NTSB made several recommendations. The most important ones were:

- That the FAA conduct a certification review of the Hamilton Standard model 14RF propeller system and require appropriate modification to ensure that the propeller system complies with the provisions of 14 CFR Section 35.21. The certification review should include subjecting the system to the vibration spectrum that would be encountered in flight on those aircraft for which it is certificated.
- That the FAA examine the certification basis of other model propeller systems that have the same design characteristics.
- That the FAA establish an inspection time requirement for the transfer tube splines, servo ball-screw and balls-crew quill on Hamilton Standard 14RF propellers and other propeller systems of similar design.

5.19.4 Lessons

1. To quote from page 38 of Ref. 5.18:

   "The Safety Board notes that there have been four reported instances of extreme wear of the PCU servo ball-screw, one of which was discovered in flight. The worn parts were not in contact with a titanium-nitrided surface or a surface that had a finish rougher than allowed in the specification. Therefore, the wear of the servo ball-screw is another case where wearing of components was not considered in the certification. The Safety Board believes that if the engagement between the ball-screw and the quill fails, it would be possible for the propeller blade angle to rotate below the flight idle angle, resulting in loss of control of the airplane. Therefore, the Safety Board concludes that the Hamilton Standard model 14RF propeller system does not comply with the purpose of the certification requirements contained in 14 CFR Section 35."

2. The real lesson for design engineers is that wear and tear must be considered in the certification process if it can be shown to lead to loss of control.

3. The failure mode found in this accident is another illustration of one single failure resulting in fatal consequences. From a certification viewpoint this is not acceptable. It is the author's view that in this case the DER system broke down.

5.20 Uncommanded Thrust Reverser Deployment

5.20.1 Problem

On May 26, 1991 and Air Lauda Boeing 767 (Figure 5.25) crashed over Thailand. There were no survivors among the 213 passengers and 10 crew members.

Figure 5.25 Boeing 767 (Not accident aircraft, Courtesy Thomas Wirtenberger)

5.20.2 Cause

The airplane had broken apart in many pieces and the wreckage was scattered over a large area. Ref. 5.19 states that the hydraulic actuators for the thrust-reverser mechanism on the No.1 engine were found in the fully deployed position. This indicated that the reverser had deployed before the crash.

Detailed investigations showed that in the PW4000 powered 767 airplanes the thrust reverser, when deployed in flight, reduces the lift on that wing by about 25%. The resulting uncommanded roll rate can reach 28 deg/sec in about 4 seconds. This would be very difficult to control unless the crew applies immediate full opposite roll control.

Investigations into the possible failure modes of the thrust reverser control system revealed several failure possibilities which had not been considered during certification.

5.20.3 Solution

A number of system design changes were made by Boeing:

- Replacing the solenoid operated HIV (Hydraulic Isolation Valve) by a motor-operated HIV;
- Adding new electric wiring from the electronics bay and flight deck to the engine struts. Critical wire isolation and protective shielding would now be required;
- Adding a new reverser test/reverser system maintenance indication panel to the flight deck;
- Replacing existing reverser stow proximity targets with improved permeability material to reduce nuisance indications;
- Adding a thrust reverser deploy pressure switch.

5.20.4 Lesson

Thrust reversers and their operating systems should be considered as flight crucial systems unless tests show that controllability can be maintained and safe landings are possible.

5.21 Power levers Moved to Beta Range in Flight I

5.21.1 Problem

In February of 1994 a Simmons Airlines SAAB 340 (Figure 5.26) the captain actively moved the power levers from the flight idle gate into the beta range for undetermined reasons (the flight manual specifically prohibits this).

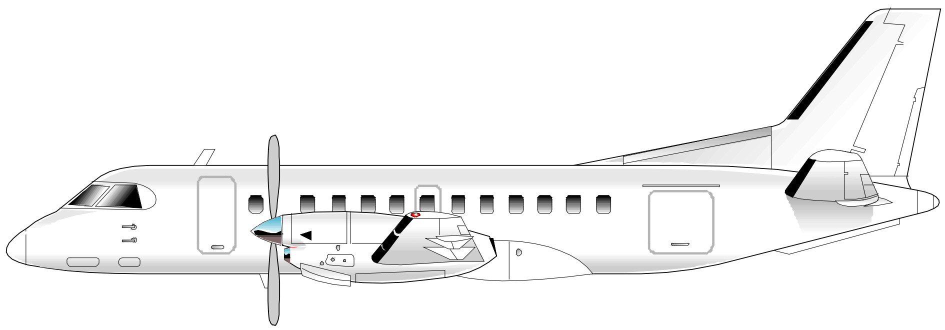


Figure 5.26 SAAB 340

As a result both engines over sped and were substantially damaged. A successful power-off landing was made. There were no injuries.

5.21.2 Cause

The design of the power levers in this airplane met the then existing design and certification requirements. However, the design did not prevent movement of the propellers into the beta range while in flight.

5.21.3 Solution

The NTSB, in Ref. 5.20, found that the airframe and engine manufacturing industry, the FAA, and the certification authorities from other countries were slow in reacting to several previous in-flight beta occurrences that led to serious incidents and accidents

5.21.4 Lesson

The fact that a pilot can move propellers into the beta range in flight, when this is specifically prohibited in the flight manual hints strongly at the need for a design change. The author believes that this type of design should never even have been submitted for certification.

This lesson does not seem to get learned. The reader is asked to read Section 5.29 about a similar situation which occurred in a Fokker 50 transport.

5.22 Uncontained Engine Failure I

5.22.1 Problem

In June of 1995 a Valujet Airlines Douglas DC-9-32 (Figure 5.27) suffered an uncontained engine failure of the No.2 engine during the start of the take-off roll.

Figure 5.27 Model of Douglas DC-9-32 (Not accident airline, Courtesy geminijets.com)

The pilot did abort the take-off and the airplane was evacuated. The flight attendant seated in the aft jump-seat received serious puncture wounds from shrapnel and thermal injuries. Another flight attendant and five passengers received minor injuries. The pilots, the third flight attendant and 52 passengers were not injured.

As a result of the uncontained engine failure, engine fragments penetrated the airplane cabin, severing the right main fuel line and causing the release of pressurized fuel inside the cabin.

Sparks which were probably generated by steel engine fragments contacting galley components ignited a fire that spread through the airplane cabin and destroyed the fuselage of the airplane.

Figure 5.28 shows the airplane after evacuation.

Figure 5.28 DC-9-32 after Uncontained Engine Failure (Courtesy NTSB)

5.22.2 Cause

According to Ref. 5.21 there were several cause-effects in this serious accident:

- The uncontained engine failure was caused by a fatigue crack in a stress distribution hole in the 7th stage compressor disk. This fatigue crack was not detected during a 1991 engine overhaul at Turk Hava Yollari in Turkey.
- Because of an electrical power loss the public address system was not functional. There have been several cases in the past where the NTSB pointed out the need for independent power systems for the public address system in airplanes. The FAA and industry have been slow in reacting to this concern.

- The aircraft did not meet the current regulatory requirements regarding flammability standards used in the interiors of transport category airplanes nor was it required to do so. This accident demonstrates again the importance of the current standards and the need to bring existing airplanes up to these standards.

5.22.3 Solution

As a result of this accident the NTSB made a number of recommendations to the FAA. The most significant of these were:

1. To upgrade aircraft interiors as soon as possible to meet the 1985 FAA standards. In this regard the NTSB observed that the need to do this was already demonstrated during a 1991 runway collision of a Boeing 737 and a Fairchild Metroliner. This collision resulted in a cabin fire in the 737 which killed 20 people due to inhalation of toxic smoke generated by the burning cabin furnishings. At that time the FAA decided that upgrading to the standards of 14 CFR 25.853 was not economically feasible. In 1994 the NTSB classified its recommendation as "Closed-unacceptable action".
2. To require foreign repair stations to adhere to the same record keeping requirements as domestic repair stations. Also, to ensure that the language used in operations specifications for repair stations clearly indicates the extent of the authority of that repair station.
3. Require that all transport category airplanes manufactured before November 27, 1990, be retrofitted with a public address system capable of operating on an independent power source.

5.22.4 Lessons

1. It is not the first time that this type of accident happens. In this 1995 accident nobody suffered toxic smoke inhalation. We were lucky this time. How long do we push our luck? Should manufacturers not themselves insist on their airplanes being modified to upgraded flammability standards?

2. The NTSB report (Ref. 5.21) does not mention the fact that a main fuel line was severed and that such an event should not be accepted even as a result of an uncontained engine failure. The author considers this a design deficiency.

3. In this airplane the rear flight attendant jump-seat and the rear passenger row are in the engine burst plane. The author considers this a design deficiency.

5.23 Propeller Blade Separation III

5.23.1 Problem

In August of 1995 an Atlantic Southeast Airlines Embraer EMB-120RT (Figure 5.29) experienced the loss of a propeller blade from the No.1 engine while climbing through 18,100 ft altitude.

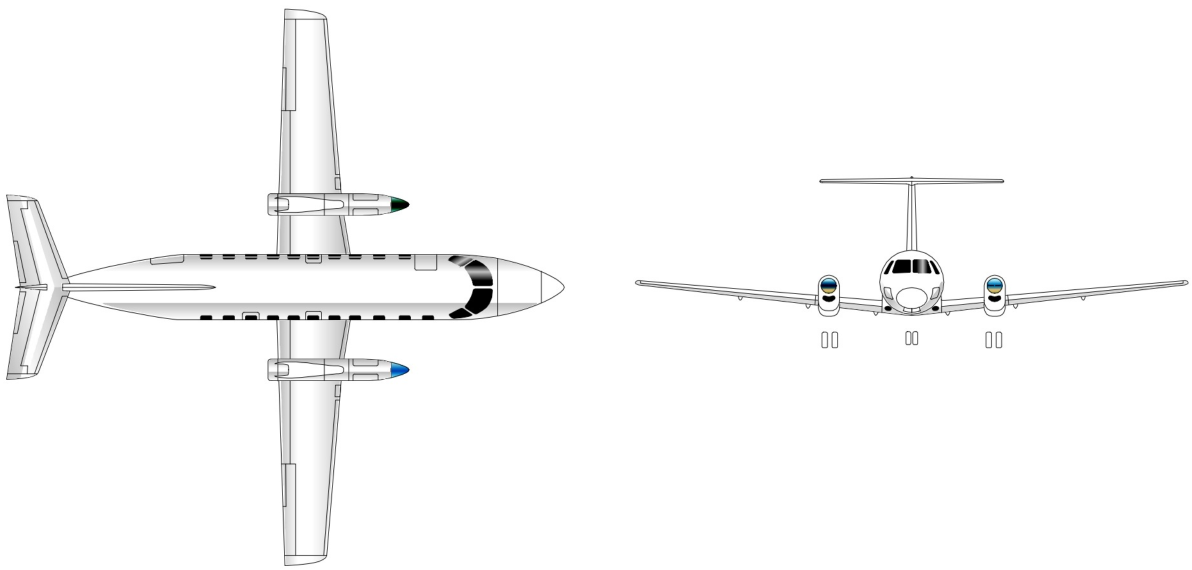


Figure 5.29 Embraer EMB-120RT

The airplane crashed during an emergency landing. The captain and seven passengers were fatally injured. Two other crew members and 11 passengers were seriously injured while 8 passengers sustained minor injuries.

5.23.2 Cause

The NTSB, in Ref. 5.22, determined that the probable cause was the in-flight fatigue fracture and separation of a propeller blade resulting in distortion of the left engine nacelle, causing excessive drag, loss of wing lift, and reduced directional control of the airplane. The fracture was caused by a fatigue crack from multiple corrosion pits that were not discovered by Hamilton Standard because of inadequate and ineffective corporate inspection and repair techniques, training, documentation and communications.

Contributing to the accident was Hamilton Standard's and the FAA's failure to require recurrent on-wing ultrasonic inspections for the affected propellers.

5.23.3 Solution

The NTSB made several following recommendations to the FAA. The most important were:

- To review the need to require inspection after the completion of work that is performed by un-certificated mechanics at part 145 repair stations to ensure the satisfactory completion of the assigned tasks.
- Require Hamilton Standard to review and, if necessary, revise its policies and procedures regarding internal communication and documentation of engineering decisions, involvement of the DER and the FAA.

5.23.4 Lesson

1. An important lesson is to always follow up on the quality of work done by as yet un-certificated mechanics.

2. The author wonders if it should be acceptable to have blade separation cause such significant distortion of a nacelle that the airplane becomes difficult to control.

5.24 Uncontained Engine Failure II

5.24.1 Problem

In July of 1996 a Delta Airlines McDonnell-Douglas MD-88 (Figure 5.30Figure 5.30) suffered an uncontained failure of the left engine fan hub during the take-off roll.

The flight crew was able to abort the take-off. There were two fatal, two serious and three minor injuries among the passengers.

Figure 5.30 Model of McDonnell-Douglas MD-88 (Courtesy geminijets.com)

5.24.2 Cause

According to Ref. 5.23 “the probable cause was the fracture of the left engine’s front compressor fan hub, which resulted from the failure of Delta Air Lines’ fluorescent penetrant inspection process to detect a detectable fatigue crack initiating from an area of altered microstructure that was created during the drilling process by Volvo for Pratt & Whitney and that went undetected at the time of manufacture. Contributing to the accident was the lack of sufficient redundancy in the in-service inspection program.”

5.24.3 Solution

To prevent this type of accident from reoccurring several recommendations are made in Ref. 5.23 to improve the manufacturing and inspection procedures of critical engine components. From an aircraft design point of view the author believes that the following comments are in order. This accident would have been inconsequential in terms of passenger fatalities and injuries if the aft two seat rows had not been in the burst plane of the fan hub.

Figure 5.31 shows the damaged engine, its nacelle and the damage done to the fuselage.

The fact that several passenger rows are in the burst plane of a critical engine component is alluded to by diagrams and pictures in Ref. 5.23. Whether or not this condition should be deemed certifiable is not discussed. Figure 5.32 shows a side-view of the rear fuselage indicating the significant overlap of the nacelle and windows of the passenger cabin.

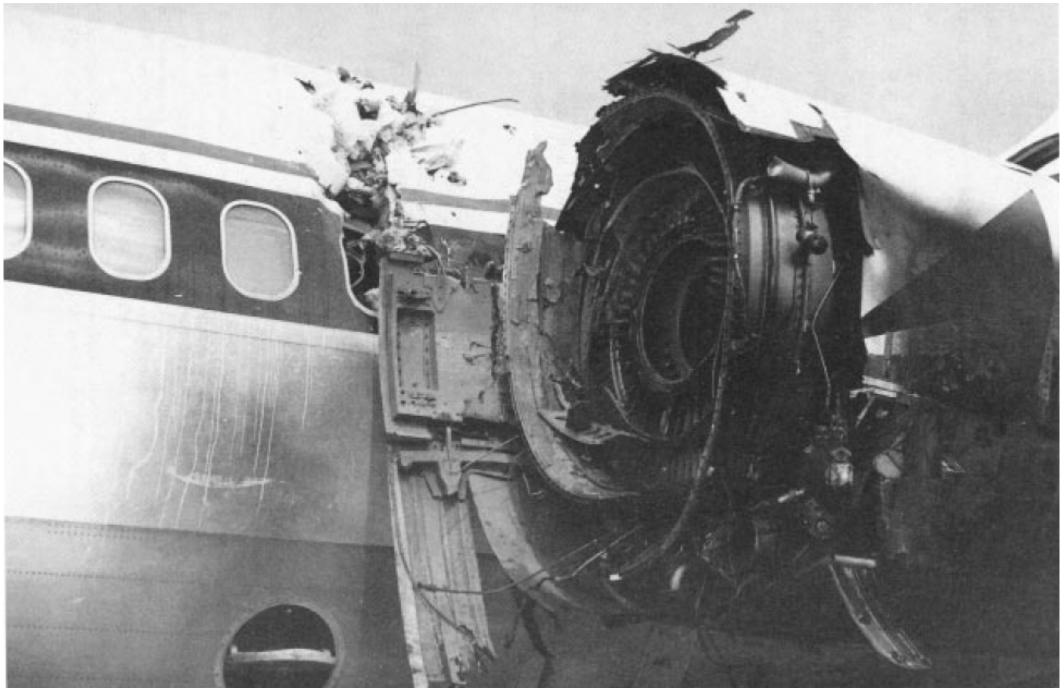


Figure 5.31 Damaged Engine, Nacelle and Fuselage (from Ref. 5.23, Courtesy NTSB)

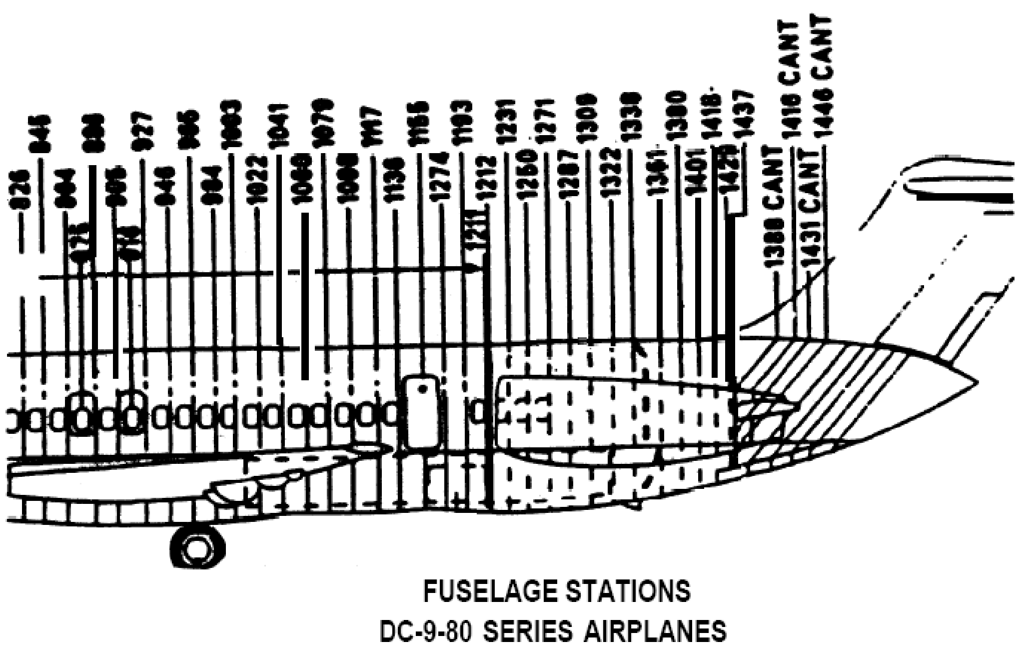


Figure 5.32 Side-view of the Rear Fuselage (from Ref. 5.23, Courtesy NTSB)

A top-view of the rear fuselage indicating where the fatally and seriously injured passengers were seated is shown in Figure 5.33.

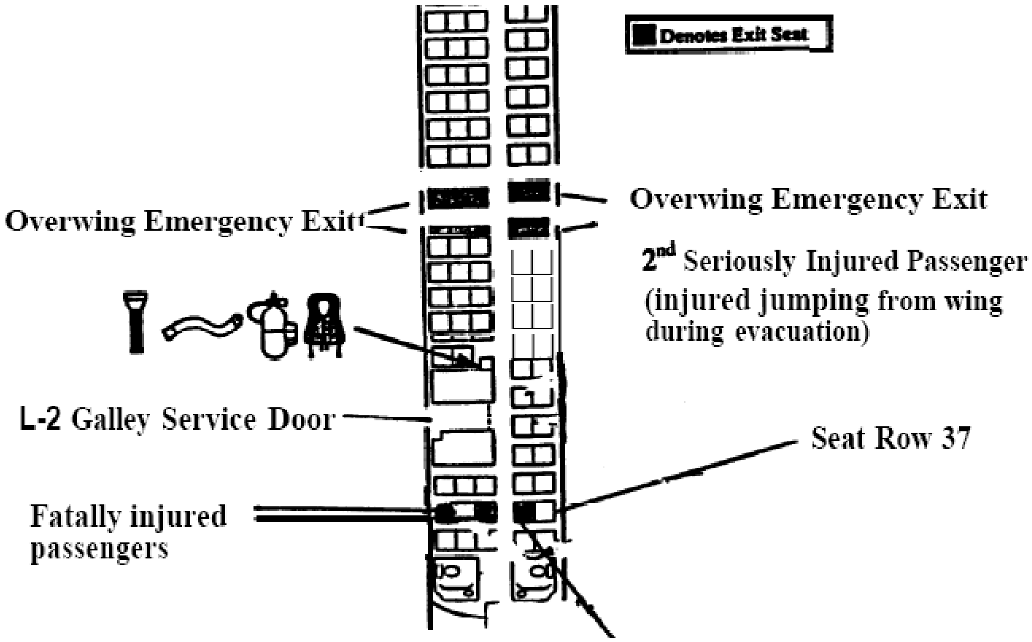


Figure 5.33 Top-view of the Rear Fuselage (from Ref. 5.23, Courtesy NTSB)

Currently there is no requirement in FAR 25 for keeping passenger seats out of burst planes. There is such a requirement for keeping the cockpit crew out of burst planes. The question that should be asked and answered is: should this type of design feature be tolerated? It is not unreasonable to expect that this type of occurrence will be repeated in airplanes with similar engine-to-passenger-seat layouts.

5.24.4 Lessons

1. Uncontained failures of critical engine components should be seen as expected events during the operational lifetime of commercial airplanes. Designers should keep this in mind when defining engine-to-passenger-seat layouts.

2. There is another important observation to be made about this accident. To quote from Ref. 5.23, page 6: "Most of the wires in the wire bundle located along longeron 4 (on the right side of the fuselage) were severed near FS 1250. Of the 154 wires in the bundle, 146 had been severed."

3. What is important to aircraft designers is to note that wire bundles on the side opposite to the side of the failed engine component were almost totally severed. In this particular case these wire bundles did not have a flight crucial function. For airplanes with fly-by-wire or fly-by-light flight control systems this type of design should certainly not be tolerated.

5.25 Uncommanded Thrust Reverser Deployment II

5.25.1 Problem

In October of 1996 a Fokker 100 (Figure 5.34) of TAM crashed on take-off. There were no survivors.

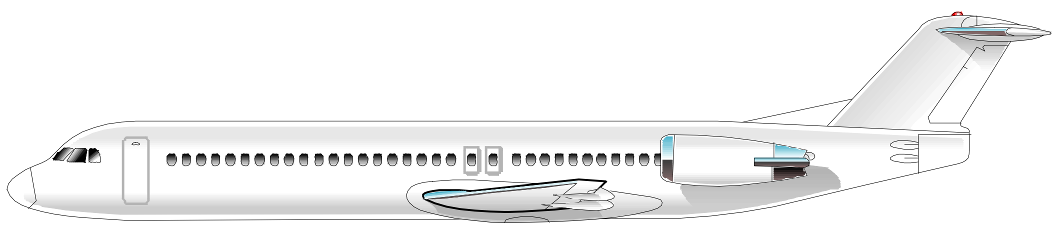


Figure 5.34 Fokker 100

5.25.2 Cause

The cause was found to have been an un-commanded thrust reverser deployment on the No.2 engine early in the take-off roll. As a result, the thrust reverser interlock cable retarded the throttle lever as intended by design of that system. However, the crew assumed that this was caused by an auto-throttle system failure. The captain then forced the throttle forward overcoming the interlock and causing the No.2 engine to go to full thrust with the reverser deployed. As a result directional control was lost and the airplane crashed.

5.25.3 Solution

The author could not find whether or not the cause of the un-commanded thrust reverser deployment was established and therefore no solution to that part of the problem is offered.

5.25.4 Lesson

Designers should contemplate the meaning of this sequence of events. Thrust reverser deployments should always be made transparent to the crew. It certainly was not in this airplane.

Also, this accident may suggest that a device which alerts the pilot to any asymmetry during the take-off run, might have been helpful.

5.26 Tire Tread Ingested Into Engine

5.26.1 Problem

In January of 2001 a BAC-111-500 (Figure 5.35) experienced severe vibration during its take-off roll. The take-off was aborted and the airplane taxied back to an apron. There were no injuries.

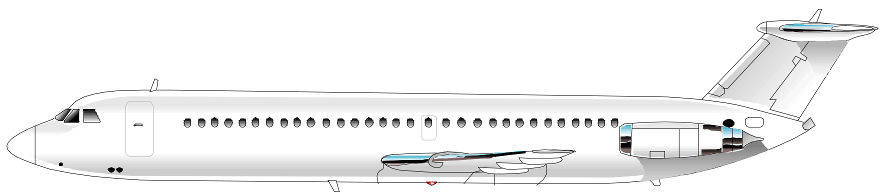


Figure 5.35 BAC 111-500

5.26.2 Cause

According to Ref. 5.24 the cause of the vibration was found to be a complete tire tread separation of the number 2 main wheel. This tread separation caused damage to the following airplane components:

- The left trailing edge flap, spoiler panel and the left main landing gear wiring loom
- Tread ingested into the number 1 engine caused 5-8% damage to the low and high pressure blades and engine inlet guide vanes.

5.26.3 Solution

The tire, flap, spoiler panel and engine were removed and replaced and the airplane returned to service.

5.26.4 Lesson

Tire failures occur fairly frequently. Tire debris should not be allowed to be ingested into engine inlets. This is an issue of early configuration design. The author believes that a configuration where tires fragments can enter engine inlets should not be certified.

5.27 Fuel Line Chafed Through

5.27.1 Problem

In March of 2001 a South African Airlines Boeing 747-400 (Figure 5.36) was on its way from Johannesburg to London.

Figure 5.36 Boeing 747-400 (Not accident aircraft, Courtesy Simon Willson)

It diverted to Barcelona, Spain because of an EICAS message: FUEL IMBALANCE on the No.4 engine. That engine was shut down and the airplane landed at Barcelona without further incident.

5.27.2 Cause

According to Ref. 5.25 the cause was found to be a chafed through fuel line between the fuel filter and the engine driven fuel pump on the No.4 engine. The chafing was caused by a securing clip which allowed relative motion between the clip and the fuel line.

5.27.3 Solution

The fuel line was replaced and a more robust clip was installed. The same type of clip was found on many other 747-400 airplanes and had caused similar problems. All were replaced.

5.27.4 Lesson

This could have been a serious accident. Design engineers should make sure that securing clips in flight crucial lines or wiring do not allow from relative motion. If they do, chafing will occur.

5.28 Involuntary Engine Shutdown

5.28.1 Problem

In October of 2002 a Boeing 717-200 (Figure 5.37) while climbing through 7,000 ft sustained an uncommanded shutdown of the No.2 engine. The R ENG RPM LO alert was observed followed by the RH SYS FAIL advisory. The crew reported that they did not see any caution advisories prior to the shutdown. The crew completed a single engine landing.

Figure 5.37 Model of Boeing 717-200 (Courtesy geminijets.com)

5.28.2 Cause

According to Ref. 5.26, after much testing the cause was found to be fractured resistor solder joints on a printed circuit board. An explanation of the EEC (Electronic Engine Control) system on the engine of this airplane is in order. It is a two-channel unit for redundancy reasons.
The EEC sends signals to the FMU (Fuel Metering Unit) which in turn directs fuel to the engines. Both units were shipped to their component manufacturer's facilities for testing. During environmental stress screening of the EEC from -55 to +74 degrees C, failures of the Channel A EEPROM were recorded when the internal temperature of the EEC was at -2 degrees C or colder. Initial testing of the EEC could not duplicate the dual channel failure (A and B) which had occurred in flight.

The Channel A EEPROM was sent to the manufacturer for detailed examination. Examination indicated that a phenomenon called a 'single bit flip' had occurred within the used memory section area of the input/output micro-processor of the unit. Follow-up vibratory testing of the EEC confirmed a failure of Channel B. Further examination indicated fracturing of solder joints at five resistors on the analog interface module circuit board of Channel B.

A review of previous similar occurrences indicated fracturing of the solder joints of six resistors of Channel A and four resistors of the Channel B analog interface module circuit boards. This happened both in Australia and in the USA.

5.28.3 Solution

It turned out that the fracturing or cracking of the solder joints resulted from thermal cycle stress due to differential thermal expansion between the printed circuit board and the resistor. Repairs were made by installing 'flying leads' to guarantee proper connections.

5.28.4 Lessons

1. Design engineers should be aware of thermally induced stresses in electronic control units whether these control engines or flight controls. Certification testing should be arranged in such a manner as to detect any failures before certification of the airplane.

2. The lesson is even more important in future 'all electric' airplanes.

5.29 Power Levers Moved to Beta Range in Flight II

5.29.1 Problem

In November of 2002 a Luxair Fokker 50 (Figure 5.38) crashed just before landing at the Luxemburg-Findel Airport. Of the 22 people on board, 20 received fatal injuries.

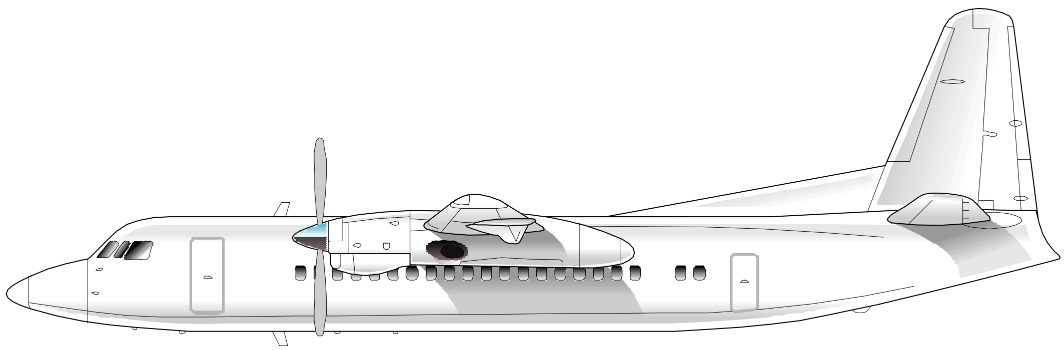


Figure 5.38 Fokker 50

5.29.2 Cause

According to Ref. 5.27 the airplane was on a final approach with poor visibility. At 3,000 ft the captain decided to initiate a go-around by announcing: “yes, well we do a go-around, missed approach” and continues to maintain 3,000 ft altitude. The co-pilot does not react to this and continues with the last item on his final approach checklist: he disengages the ‘ground idle stop’ and properly announces that he carried out that action. At that point the control tower advises the flight that ground visibility has improved above minimums and the captain changes his mind. He decides to continue the approach but realizes that a much larger descent rate is now called for. He also realizes that the airplane is too fast. Figure 5.39 shows the actual airplane trajectory compared to the required ILS glide-slope.

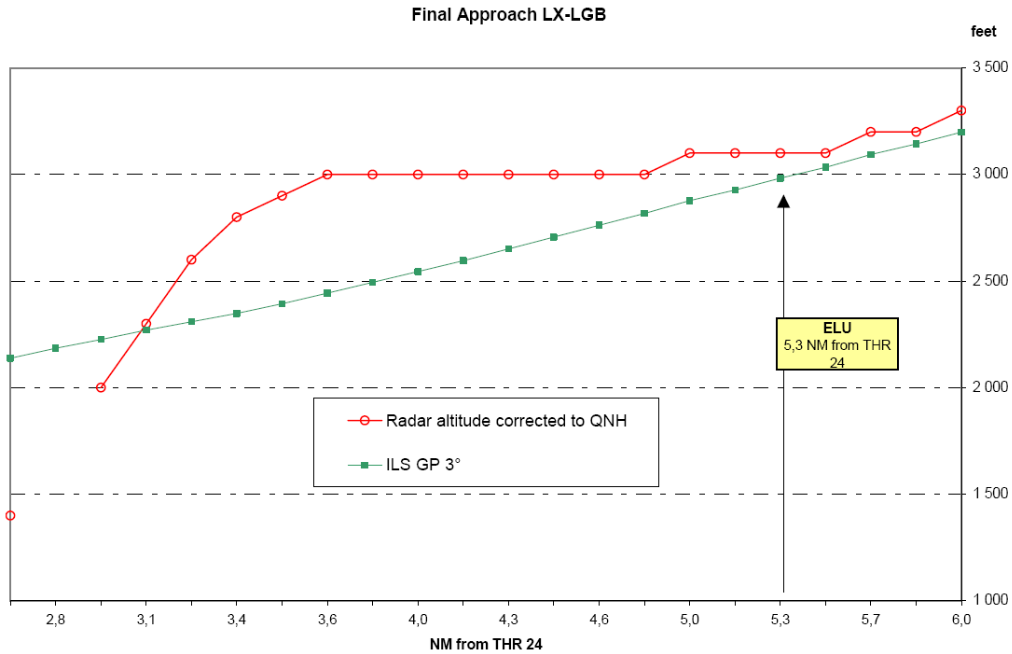


Figure 5.39 Actual and Intended Airplane Trajectory (from Ref. 5.27)

To achieve the higher rate of descent and slow the airplane down, the captain now decides (without announcing his intention to the co-pilot) to move the throttles back to flight idle. At the same time he pulls up on the ground range selectors (Figure 5.40).

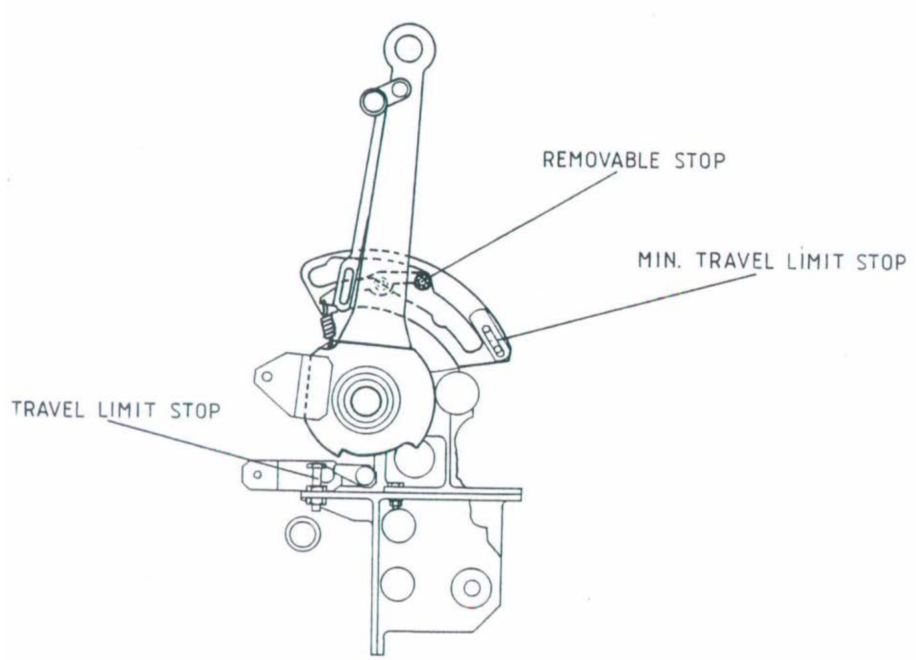


Figure 5.40 Throttles and Ground Range Selector of the Fokker 50 (from Ref. 5.27)

The ground range selectors enable the throttles to be moved a bit further back, close to the ‘beta range’. However, the use of the beta range while in flight is not permitted in this airplane because control can be lost.

The co-pilot now puts the flaps down 10 degrees and requests permission to lower the landing gear. The captain okay’s this. As soon as the gear is selected down, the secondary stop on the throttles (which prevents the throttles from being moved into the beta range) is removed (see Figure 5.40). The captain probably has a firm grip on the throttles, does not know that the secondary stop is gone and unintentionally moves the throttles to ground idle. Then things happen very fast. The propellers are now at reverse, the airplane slows down rapidly and the propellers begin to over-speed. The flight crew then moves the throttles full forward, retracts the flaps and incredibly shuts down the engine fuel supply. The airplane then crashes.

5.29.3 Solution

A mandatory modification aimed at preventing the secondary stop from being removed in flight is introduced.

5.29.4 Lesson

Designers should in fact have foreseen these events and designed the system from the start to prevent this.

Chapter 6

Lessons Drawn from Systems Design

"All systems which carry a liquid will leak, the devil is in the detail design"

Dr. Jan Roskam, 1990

6.1 Introduction

In this chapter a series of problems which arose in systems design, their causes and solutions are described. Finally, lessons learned are stated.

As will be seen, systems design covers a wide area. Systems, when not cleverly designed can cause many problems in maintenance and service, particularly where the training of personnel is an issue. The latter is more and more the case in this era of cost-cutting and low pay of maintenance personnel. Systems designers must keep in mind the fact that the annual turn-over rate of personnel is very high (50% in some cases!). Systems must therefore be designed so that they are simple and fool-proof.

6.2 Electrical System Design I

6.2.1 Problem

In July of 1946 a Transcontinental and Western Air, Inc. Lockheed L-049 Constellation (see Figure 6.1) crashed near the Reading Airport in Pennsylvania. The airplane was on a training flight with six crew members on board. Five were fatally injured and one seriously injured. The aircraft was demolished by impact and by fire.

Figure 6.1 Lockheed L-049 Constellation (Courtesy H. Chaloner)

6.2.2 Cause

According to Ref. 6.1 the probable cause of this accident was: "failure of at least one of the generator lead through-stud installations in the fuselage skin of the forward baggage compartment which resulted in intense local heating due to electrical arcing, ignition of the fuselage insulation, and creation of smoke of such density that sustained control of the aircraft became impossible. A contributing factor was the deficiency in the inspection systems which permitted defects in the aircraft to persist over a long period of time and to reach such proportions as to create a hazardous condition."

To understand what happened it is useful to review several aspects of the electrical and hydraulic system design in this airplane. The following material has been adapted from Ref. 6.1.

Figure 6.2 shows the location of the suspect lead through-studs in the left wing root of the airplane. Figure 6.3 shows evidence of serious electrical arcing on these studs as retrieved from the wreckage.

To facilitate the accident investigation TWA brought in a serviceable Constellation of the same type as the accident aircraft. This made it possible to determine the actual relative position of various components found in the wreckage.

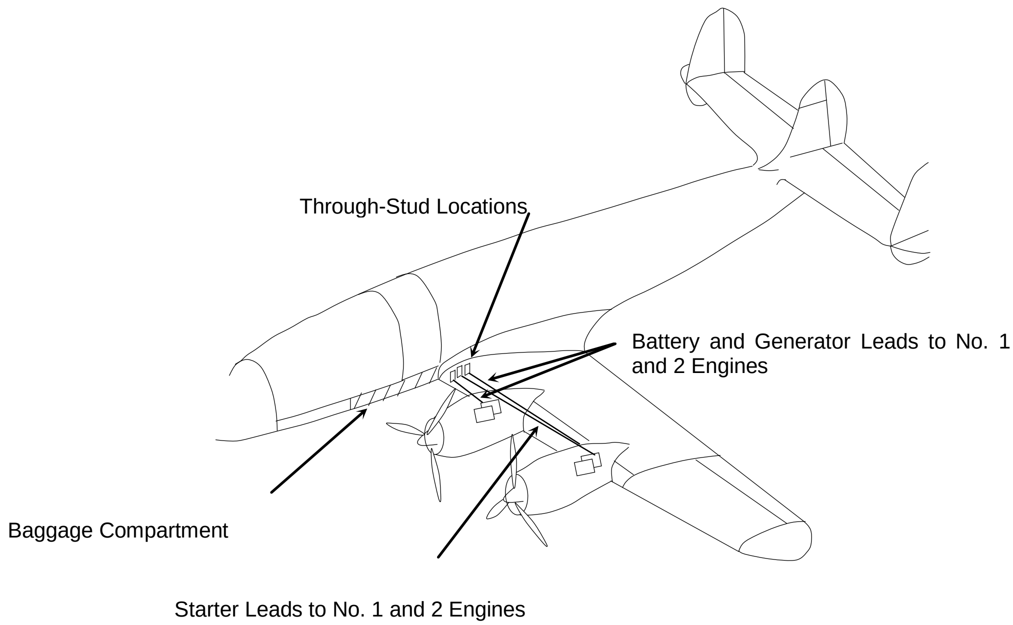


Figure 6.2 Perspective of the Through Studs in the Left Wing Root

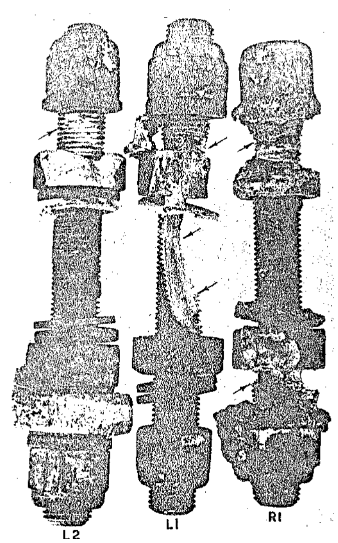


Figure 6.3 Evidence of Arcing on Three Lead Through Studs

Inspection of the generator and starter leads and the fuselage through studs in the forward baggage compartment of the serviceable airplane disclosed that some of the Irvolite coverings of the cable lugs and studs were severely charred. Because high external temperatures could not have been present on this area, it was obvious that this charring had resulted from internal heating. It was further observed that the glass wool lining of the baggage compartment in the proximity of the through-studs was saturated with hydraulic fluid. Because hydraulic lines pass through the forward baggage compartment, it is apparent that the leakage had occurred from hydraulic line fittings.

Investigation of all six (three left and three right) through-studs from the accident airplane showed clear signs of local burning suggestive of electrical arcing. The left three are shown in Figure 6.3 since these were the source of the problem.

The Bureau of Standards was requested to conduct a detailed examination of these studs. Their report indicated that, if in contact with inflammable materials, such an installation could readily produce a fire when operated at high electrical loads for long periods of time. The Bureau also found that none of the studs were assembled according to Lockheed instructions. Both steel and aluminum washers were used; the lugs used were of both copper and aluminum; brass and steel nuts were used, and the bolts themselves were of brass composition. The almost indiscriminate use of materials of different compositions, according to the Bureau of Standards report, aggravated the high contact resistances of the studs and, therefore, increased the likelihood of pitting and the development of high internal temperatures.

Laboratory tests were also conducted by Lockheed to determine the possibility of fire hazard from an improperly installed stud assembly. A mock-up was constructed to simulate a section of the outer panel of the forward baggage compartment and a through-stud installed in this section with the extreme inboard nut loose. As long as the stud was not in actual contact with the fuselage skin, excessive temperatures did not develop. However, the first stud tested in contact with the skin was burned in two within 60 seconds as a result of intense electrical arcing. The same test was repeated to simulate a situation in which the inboard lead contacted a hydraulic line after falling from the skin. Such contact caused sufficient arcing to puncture the line and, under hydraulic pressures of 100 psi or greater, the fluid invariably ignited and burned with intense heat. The findings of the Lockheed tests, therefore, indicate that, in the event of a faulty installation in which contact with the fuselage skin results, sufficient energy may be dissipated at the through-stud position to ignite surrounding inflammable materials. It was further indicated that the resultant arcing may burn the stud in two and that current may continue to be led to the inboard lead should the lead subsequently become grounded before the reverse current relay opens.

The primary deficiency in the electrical system lay in the detail design of the through-stud. Even though insulating material separated the stud from the fuselage skin, this separation was maintained primarily from pressure of the nuts at each end of the stud against micarta insulating blocks on each side of the skin.

It also became evident that because of the absence of drain provisions hydraulic fluid could accumulate and cause a fire hazard.

Finally it became obvious that development of less flammable (or better, non-inflammable) hydraulic fluid was needed and that the insulating materials in the baggage compartment should be replaced with non-inflammable materials.

6.2.3 Solution

As a result of this accident investigation the Safety Board reached the following conclusions:

- The design of the generator lead through-studs rendered those studs susceptible to grounding to the fuselage skin and the development of extremely high local temperatures due to electrical arcing.
- The through-stud design, furthermore, contained excessively high contact resistance, which, at peak loads, may cause arcing even when the stud is properly installed.
- The presence of inflammable fuselage insulation in the vicinity of the through-stud installation created a fire hazard.
- Hydraulic fluid leakage into the forward baggage compartment of L-049 aircraft presented a serious fire hazard.
- The baggage compartments of 049 aircraft were not readily accessible from the cabin or crew compartment for purposes of fire control.
- L-049 aircraft were inadequately provided with fire or smoke detecting systems within the fuselage.
- L-049 aircraft were inadequately provided with fire-extinguishing equipment for possible fuselage fires and require remote extinguisher systems for the baggage compartments.
- Insufficient attention had been provided the subject of air flow control within the aircraft presently employed in air carrier service.
- Inspectional policies which have heretofore been followed did not provide adequate assurance of the elimination of particular categories of deficiencies of design or construction within newer aircraft.

As a result of this accident a large number of changes were incorporated by Lockheed in Constellation type aircraft. Specifically, the following modifications were carried out by Lockheed on all L-049 and subsequent aircraft:

- Electrical System: The insulation of certain wiring was modified to prevent possible damage from chafing and to prevent contact with parts of the aircraft. The fuselage through-stud assemblies were replaced with units of a new design. Undersized aluminum conductors in the generator circuits were replaced with copper cables. A general improvement was accomplished in circuit breakers, fuses, and control switches to prevent shorting.
- Power-plants: Fire extinguisher protection for the accessory section of the engine nacelles was provided and the extinguishing system was modified to provide two 30 pound discharges instead of three 15 pound discharges of the original system. Provisions were made for increased drainage and ventilation of the engine nacelles aft of the firewall in order to prevent the accumulation of combustible fluids or vapors. Fluid carrying lines in the engine nacelles were made more fire resistant. More sturdy attachments were provided for these lines and protection was provided against chafing. The alcohol tanks were replaced by tanks of a heavier gauge steel. Several modifications were accomplished in the exhaust collector ring to prevent failure of this component.
- Miscellaneous: Prior to re-installation and use of cabin supercharger drive shafts, modifications were made to prevent failure of the shafts. Certain hydraulic lines were relocated to reduce possible fire hazards and, in some instances, to prevent contamination of the oxygen system. Drains were provided in the baggage compartments to prevent accumulation of hydraulic fluid.

The CAA required the manufacturer to conduct an accelerated service test of 50 hours following the completion of the above modifications before this aircraft could be used in scheduled operations. The first Constellation was returned to service August 24, 1946.

In addition the CAA airworthiness regulations were amended in several areas to improve the safety of air carrier aircraft in these areas.

6.2.4 Lessons

Every system in an airplane which carries a liquid will leak. Hydraulic lines will most likely leak at their fittings. If hydraulic fluid can be absorbed (i.e. accumulate) in other materials or components and if heat sources are nearby a recipe for disaster is created.

Electrical systems are vulnerable to development of local high temperatures unless great care is taken in the detail design of these systems.

Designers familiar with the modern regulations with regard to systems design will recognize in this accident some of the reasons for these regulations.

Most design engineers will agree that the events leading to this accident could have been predicted. It should not require a regulation to ask simple questions about the possibility of malfunctions and, from the answers, deduce design improvements.

6.3 Fuel System and Electrical System Design

6.3.1 Problem

In April of 1947 a North American Navion (Figure 6.4) crashed near Ada, Oklahoma killing all four occupants.

Figure 6.4 North American Navion
(Not accident aircraft, Courtesy San Diego Aerospace Museum)

As the aircraft was on approach to Walker Field near Ada, OK, witnesses saw the nose wheel extend and, almost simultaneously saw smoke around the fuselage and heard an explosive sound. The right wing appeared to disintegrate in the air, the aircraft crashed and became enveloped in flames.

6.3.2 Cause

Ref. 6.2 states as the probable cause of this accident the disintegration of the right wing resulting from an explosion of a fuel-air mixture in the right wing panel. Ignition may have resulted from the operation of the landing gear position switch.

The following has been adapted from Ref. 6.2.

The most likely explanation for the disintegration of the right wing is that of an internal explosion. The uniform outward distortion of the wing ribs could result only from the existence of an excessively strong internal pressure. The uniform pattern of skin rivet failures, all of which were in an outward direction, can be explained only from the fact that all parts of the wing were forced outward by the presence of an excessively strong pressure in the wing panel. The buckling of the stringer legs between rib stations was also the result of internal pressure in the wing panel. As the stringers were bowed outward, while still attached to the ribs, the legs of the stringers were compressed. Likewise, the tension breaks in the top and bottom attachments of the wing tip, and the ballooning effect indicates a result of excessive internal pressures.

It could not be positively established how a fuel-air mixture was present in the wing. Fuel may have entered the wing panel through leaks in the fuel tank or fuel lines, and then mixed with the air. It is also possible that fuel may have seeped into the panel at the time of servicing. Providing that there is adequate ventilation in the wing panel, free fuel in the wing does not in itself constitute a hazard. There are several lightening holes and control arm access holes in the rear closure web of the wing. These holes apparently did not provide adequate ventilation to prevent a flammable fuel-air mixture from forming inside the wing panel.

The most likely source of ignition in this case was the presence of a toggle switch in the wheel well. This switch serves to indicate the up/down position of the landing gear. The switch was not vapor proof, and hence, not explosive proof. Since the aircraft at the time of the accident was at a position when the landing gear would normally be extended, and since the disintegration of the wing was observed to be nearly simultaneous with the extension of the nose gear, it appears probable that the source of ignition in this case was the operation of this toggle switch.

6.3.3 Solution

Prevent fuel leaks and certainly design electrical systems in the vicinity of fuel tanks and/or fuel lines so that sparks cannot be generated.

6.3.4 Lesson

Fuel systems and fuel lines will leak. Never put spark generating equipment in an area where flammable mixtures might be expected to form.

6.4 Fuel Vent Design I

6.4.1 Problem

In October of 1947 a United Air Lines Douglas DC-6 (Figure 6.5) was observed on fire in flight and crashed while approaching a strip near Bryce Canyon, Utah.

Figure 6.5 Douglas DC-6 (Not accident aircraft, Courtesy www.prop-liners.com and United Airlines)

The airplane was demolished by impact and fire. All 46 passengers and the crew of 6 were killed.

A month later, in November of 1947, an American Air Lines DC-6 caught fire in flight and made an emergency landing at Gallup, New Mexico. None of the 21 passengers or the crew of 4 were hurt. The airplane sustained major damage due to the fire.

6.4.2 Cause

Refs. 6.3 and 6.4 state as the probable cause the combustion of gasoline which had entered the cabin heater air intake scoop from the No. 3 alternate tank vent due to inadvertent overflow during the transfer of fuel from the No.4 alternate tank. Contributing factors were the improper location of the No.3 alternate tank air vent outlet and the lack of instructions provided DC-6 flight crews concerning hazards associated with fuel transfer.

The following material was adapted from Ref. 6.3.

Reconstruction of the fuselage and analysis of the burning of its structural components indicate that the burning in flight took place in an area covering the lower right side of the fuselage beginning at a point in the center section approximately mid-wing and extending rearward approximately 23 feet and upward along the right side of the fuselage to the top of the window line.

Control cables passing through the air-conditioning compartment, commonly referred to as the “boiler room,” were found to have been partially consumed by fire and it was evident that all of these cables had failed in tension in the burned area.

Inspection of the DC-6 fuel system disclosed that the No. 3 alternate tank vent outlet was located on the right side of the fuselage near the leading edge of the wing and close to the bottom wing fillet. Approximately 10 feet of this point and slightly to the left there was an air scoop which served as the source of cabin heater combustion air and cooler air for the cabin supercharger air after-cooler and cabin supercharger oil cooler. Figure 6.6 shows a sketch indicating the approximate relative location of these items.

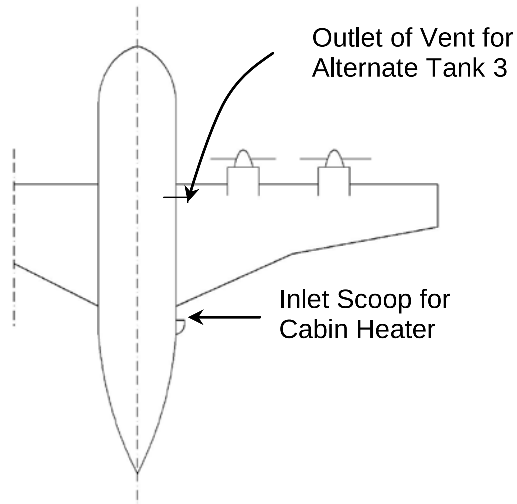


Figure 6.6 Approximate Location of Alternate Tank Vent Outlet and Air Scoop of the Cabin Heater System

Flight tests conducted with other model DC-6 aircraft subsequent to this accident revealed that overflow from the No. 3 alternate tank through the air vent line and out the vent outlet would sweep back in the slip stream toward the cabin heater combustion air intake scoop and that a considerable quantity of fuel would enter the scoop. Ground tests clearly demonstrated that, under conditions simulating the entry of fuel overflow into the scoop in flight while the heater was operating, the cabin heater could be expected to backfire and thereby propagate flame downstream into the air scoop. Incoming fuel would, thereafter, be expected to continue to burn in the air scoop and duct.

According to testimony of the manufacturer’s representatives, the DC-6 fuel system was not designed for fuel transfer between tanks. However, it is apparent that this system is readily adaptable to fuel transfer and was, in fact, extensively employed for this purpose prior to the accident. Testimony of representatives of Douglas Aircraft Company, the Civil Aeronautics Administration, and air carriers operating the DC-6 aircraft disclosed that no tests were conducted prior to certification of this model aircraft to determine whether any hazard existed through possible overflow of fuel from the vent outlet into the cabin heater combustion air intake scoop during flight.

No instructions had been given the air carrier’s pilots concerning possible hazards associated with overflow of gasoline from the No. 3 alternate tank. No instructions were provided in the manufacturer’s DC-6 Operation Manual, or the CAA approved DC-6 Aircraft Operating Manual advising against fuel transfer, nor were any instructions contained in the air carrier's DC-6 Pilot's Operating Manual outlining procedures for fuel transfer.

6.4.3 Solution

Since the industry voluntarily withdrew the DC-6 from scheduled service after the November 11 accident (Ref. 6.3) a list of proposed modifications was drawn up. This list constituted a minimum modification plan before this model was re-entered in service.

The modification plan required:

- Relocation of the Nos. 2 and 3 alternate tank vent outlets to areas from which no hazardous overflow conditions will exist.
- Guards are required for all fuel booster pump switches.
- Extensive modification of the electrical system to increase protection against possible fire hazards from booster pump switches.
- Modifications of fire extinguishing systems for the power-plants.
- Addition of drainage provisions and added protection against fuel leakage.

6.4.4 Lesson

The scenario of events leading to this crash could and should have been predicted. A systems design review would have disclosed this as a possibility and redesign would have followed. The airplane should not have been certified without such actions.

6.5 Fire Extinguishing System Design

6.5.1 Problem

In June of 1948 a United Air Lines DC-6 (Figure 6.7) crashed three miles south of Mt. Carmel in Pennsylvania. All 39 passengers and four crew members were killed.

Figure 6.7 Douglas DC-6 (Not accident aircraft, Courtesy www.prop-liners.com and United Airlines)

6.5.2 Cause

In Ref. 6.5 the Safety Board determines the probable cause to be the incapacitation of the flight crew by a concentration of CO_2 gas in the cockpit.

There were indications that the crew believed there to have been a fire. In that case the crew is supposed to follow the following procedure in the CAA approved Airplane Operating Manual:

- Cabin superchargers-DE-CLUTCH
- Emergency pressure control-Rotate fully open, this will open relief valve also. WARNING-Failure to open valve may result in excessive amounts of CO_2 in cockpit and cabin.
- Compartment CO_2 selector-Pull fully out

- Discharge one CO_2 selector-Pull out (15 seconds after de-clutching)
- Descend immediately to minimum safe altitude.
- If by inspection a second CO_2 discharge is necessary, re-pull compartment selector and then discharge second CO_2 supply.
- If fire is not under control at this point-LAND IMMEDIATELY

From the wreckage it was apparent that the cabin superchargers had not been de-clutched. In such an event it is likely that a dangerous concentration of CO_2 gas will accumulate in the cockpit. This is what happened and the crew was no longer capable of controlling the airplane.

6.5.3 Solution

The Air Line Pilots Association recommended (in March 1948) to the CAB that “smoke masks type oxygen equipment be required available for all members of the crew on transport aircraft.”

The reason for this recommendation (which was eventually accepted) was to “assure that the crew would be able to carry on their work of landing the aircraft safely in spite of possible smoke interference in case of an aircraft fire.”

6.5.4 Lesson

It is very unreasonable for designers to assume that flight crews in the case of a fire emergency (real or perceived) will follow a complicated procedure such as that described in Section 6.5.2. The second instruction in Section 6.5.2 should have raised a warning flag to those responsible for the certification of this system.

6.6 Hydraulic System Design I

6.6.1 Problem

In September of 1950 a Northwest Airlines Martin 202 (Figure 6.8) crashed during take-off from Billings, Montana. None of the 16 passengers or crew of three were injured.

Figure 6.8 Martin 202 (Not accident aircraft, Courtesy Royal Aeronautical Society Library)

During the take-off roll blue smoke was observed between the left pedals. When the aircraft had accelerated to about 80 kts a large puff of smoke suddenly filled the cockpit. Throttles were retarded and brakes applied to abort the take-off. The airplane failed to slow down and since the runway had a down slope the captain applied full reverse thrust and instructed the copilot to steer the airplane with the nose-gear. Both nose-wheel steering and the brakes were not effective and it was noticed that hydraulic pressure in the main and emergency system had dropped to zero. The airplane overran the end of the runway and finally came to a stop. There was no fire and all persons on board were evacuated.

6.6.2 Cause

The following material was adapted from Ref. 6.6.

A considerable amount of hydraulic fluid was found in the nose-wheel compartment. This was traced to a separation of the tubing from a reducer fitting in the hydraulic line from the emergency accumulator to the emergency pressure gauge in the cockpit.

In the Martin 202 the emergency accumulator is charged from the same line which supplies the main accumulator. The two accumulators are separated by a check valve which prevents the fluid from returning from the emergency system. The fitting which failed was located in the line between the emergency accumulator and the emergency power brake valve. When the failure occurred fluid from the main accumulator flowed through the check valve into the emergency accumulator and from there out of the open line where the fitting was located. As a result, all pressure was lost from both accumulators, and neither the brakes nor the hydraulic nose-wheel steering mechanism could be actuated.

It turned out that the problem was with the improper installation of the reducer fitting.

It also turned out that in October of 1949 Northwest Airlines had experienced a similar failure, that time in flight. The result was the same.

These failures demonstrated that one single failure could result in the loss of both the main and the emergency brake systems. Section 4b.337 of the Civil Airworthiness Regulations specifically prohibited this. The conclusion therefore was that the airplane was not properly certified.

6.6.3 Solution

The certification process had to be tightened up to ensure that single failures cannot result in the total loss of brakes and steering.

6.6.4 Lesson

The DER (Designated Engineering Representative) system clearly failed in this instance. At the layout design level of any system in an airplane a critical design review should allow this type of a problem to be spotted before the design is finalized.

6.7 Design Induced Mistake I

6.7.1 Problem

In January of 1951 a Pan American World Airways Boeing 377 Stratocruiser (Figure 6.9) was considerably damaged when the right main landing gear retracted during a landing at Heathrow Airport near London, England.

Figure 6.9 Boeing 377 Stratocruiser (Not accident aircraft, Courtesy Colin Zuppicich)

There were no passengers on board (this was a ferry flight) and the crew of 9 was not injured.

6.7.2 Cause

Ref. 6.7 lists as the probable cause the captain's action in mistakenly placing the landing gear control switch in the "up" position during the landing roll.

There was slush on the runway. After touching down on the main gears the nose-gear also became grounded almost immediately. The captain applied reverse thrust on propellers No. 2 and No. 3. After un-reversing and noting that the slush was getting deeper the captain decided to raise the flaps. However, instead of actuating the flap switch, he mistakenly moved the landing gear switch to the "up" position. Although it was immediately returned to the "down" position, the landing gear horn sounded and shortly thereafter the right wing began to drop.

To understand what happened, the following description of the landing gear system from Ref. 6.7 will be helpful.

Each landing gear oleo strut is equipped with two micro-switches which are actuated when the landing gear wheel is grounded firmly enough to compress the strut approximately one-half inch of its travel. These switches are a part of two entirely separate safety systems, the purpose of one being: to prevent the throttles being moved into the reverse thrust position *before* the aircraft is grounded, and the purpose of the other, to prevent an extended landing gear from being retracted *after* it is firmly grounded even though the landing gear control switch is placed in the gear "up" position.

However, it is not necessary that all three landing gear units be firmly on the ground before the throttles can be manually moved into the reverse thrust position. This can be accomplished as soon as any one of the landing gear units is supporting sufficient weight to actuate the appropriate micro-switch. However, if the landing gear control switch is placed in the gear "up" position during landing roll, any landing gear unit will unlock and retract if there is not sufficient weight maintained to hold the micro-switch in its actuated position.

As no fault could be found in any of the switches in the accident airplane, it was concluded that their was not sufficient weight on the right main landing gear to prevent its retraction when the landing gear switch was placed in the "up" position.

6.7.3 Solution

This is a classical example of a design induced error and Murphy's Law in action. With proper cockpit ergonomics in mind the designers should not have configured the landing gear "up" switch in close proximity to the flap switch.

6.7.4 Lesson

Cockpit ergonomics must be considered in detail during early layout design of the cockpit.

6.8 Service Door Fasteners

6.8.1 Problem

In July of 1951 an Eastern Air Lines Lockheed L-749 Constellation (Figure 6.10) made an emergency landing with the landing gear retracted on Curles Neck Farm near Richmond, Virginia.

There was major damage to the airplane but no injuries to the 48 passengers or crew of 5.

Figure 6.10 Lockheed L-749 Constellation (Not accident aircraft or airline, Courtesy Mel Lawrence)

6.8.2 Cause

In Ref. 6.8 the Safety Board determined the probable cause to be the in-flight opening of the hydraulic access door, which caused extreme buffeting of the aircraft and resulted in the captain's decision to make an emergency landing.

The following has been adapted from Ref. 6.8.

The airplane had encountered violent turbulence and intermittent periods of hail during its flight. It was during this period that the captain first noted severe buffeting of the aircraft. After flying in and out of weather the buffeting became so severe that the crew believed the airplane would disintegrate. Slowing down the airplane did not seem to help and upon seeing Curles Neck Farm the captain decided to set the airplane down in its largest field with the gear and flaps retracted.

The hydraulic reservoir access door located on the top side of the left wing-to-fuselage fillet forward of the front spar, was found fully open. This door is approximately 9 inches wide and 15 inches long. Its four Hartwell-make fasteners were of the type referred to as Messerschmitt fasteners. They furnish a flush closure with the aircraft surface and are actuated simply by a moderate pressure of finger and thumb. The fastener is spring-loaded, and the model used on this airplane had two small coil springs per fastener. The spring keeps the fastener latched, and also serves to force the latch open once pressure has been applied to unlock it. One spring on each of two fasteners on the accident airplane was detached. A slight upward bend was noted on the outer edge of the door frame, corresponding with the position occupied by one of the fasteners with the door closed. This indicated the possibility that the door was sprung open while this one fastener was in the locked position. A considerable amount of mud and debris was found in the drain scupper of the hydraulic filler neck, corresponding closely with the soil and vegetation of the field across which the aircraft skidded.

On the basis of hail damage found in the vicinity of the access door it was conjectured that hail may have forced the change in fastener position.

Investigation disclosed that previous in-flight openings of this door on other Constellations had been reported. In each instance the manner in which the door had opened could not be ascertained.

Eastern Air Lines ran some flight tests following the accident on a similar model airplane. It was found that the very severe buffeting could be eliminated by extension of the flaps. Figure 6.11 shows why this works from an aerodynamic viewpoint.

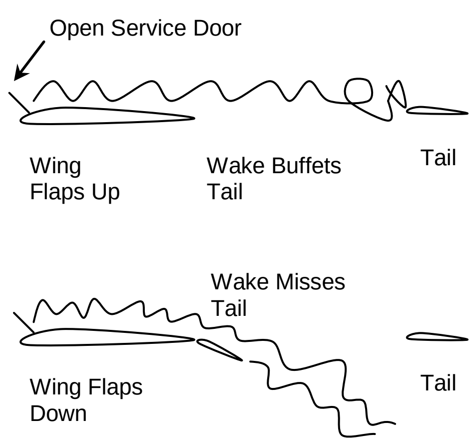


Figure 6.11 Explanation of Buffeting and Elimination thereof by Lowering Flaps

6.8.3 Solution

Following this accident Eastern Airlines modified all these access doors by adding one Dzus fastener as a positive lock should external forces cause the Hartwell fasteners to release. The CAA advised all Constellation operators of the conditions which could be encountered, and changes which might be made in the flight configuration (i.e. putting the flaps down after reaching the appropriate speed) to overcome the buffeting.

6.8.4 Lesson

Service doors sometimes open in flight: Murphy's Law. The consequence of this happening should be considered and, in cases where this is safely possible, flight tested to determine possible action. This should have been done before certification of any airplane.

6.9 Design Induced Mistake II

6.9.1 Problem

In September of 1951 an Eastern Air Lines Douglas DC-4 (Figure 6.12) was seen to land at Miami International Airport, Florida, roll a considerable distance after which the landing gear retracted.

Figure 6.12 Douglas DC-4 (Not accident aircraft, Courtesy M. West)

The airplane was substantially damaged. None of the 23 passengers and 3 crew members were injured.

6.9.2 Cause

Ref. 6.9 gives as the probable cause the inadvertent moving of the landing gear control lever upward during the landing roll, causing the gear to retract.
It turns out that in this airplane as well as in the example of Section 6.7 the flap and gear handles are so close together that a pilot can easily make the mistake of accidentally selecting gear “up” instead of “flaps up”. This is apparently what happened here while there was insufficient weight on the gear to prevent the gear from retracting.

6.9.3 Solution

Again: this is a classical example of a design induced error and Murphy’s Law in action. With proper cockpit ergonomics in mind the designers should not have configured the landing gear “up” switch in close proximity to the flap switch.

6.9.4 Lesson

Again: cockpit ergonomics must be considered in detail during early layout design of the cockpit

6.10 Firewall Fuel Shut-Off Valve Cables in Wheel Well

6.10.1 Problem

In August of 1952 a Curtiss C-46F (Figure 6.13) operated by The Unit Export Company suffered a power loss immediately after take-off which forced a wheels-up landing at Prescott, Arizona. There was no fire and there were no injuries.

Figure 6.13 Curtiss C-46F Commando (Not accident aircraft or airline, Courtesy G. Helmer)

6.10.2 Cause

Ref. 6.10 states as the probable cause a complete loss of power from the right engine shortly after the aircraft became airborne. Under the circumstances the aircraft could not maintain single-engine flight. The loss of power resulted from the closing of the emergency fuel shut-off valves due to the fouling of their actuating cables by the right tire.

The flight crew testified hearing a loud noise, seemingly from the right engine at the time it lost power. Investigators believe that this noise was caused by the wheel well cables to the emergency fuel shut-off valves being yanked violently such that not only the cables broke but their supporting pulley bracket was torn loose from its wall.

It is probable that the incident happened because the wheels were not braked (despite testimony from the pilots that they were) and the cables were not perfectly rigged.

6.10.3 Solution

Do not put cables which actuate fuel shut-off valves in the wheel well so that a rotating or exploding tire can cut off fuel supply to the engines.

6.10.4 Lesson

Landing gear wells should be kept free of all systems that have a flight crucial consequence when failed.

6.11 Hydraulic System Design II

6.11.1 Problem

In September of 1953 a Lockheed L-1049 Constellation (Figure 6.14) operated by Northwest Airlines burned following an emergency landing at McChord Air Force Base near Tacoma, Washington.

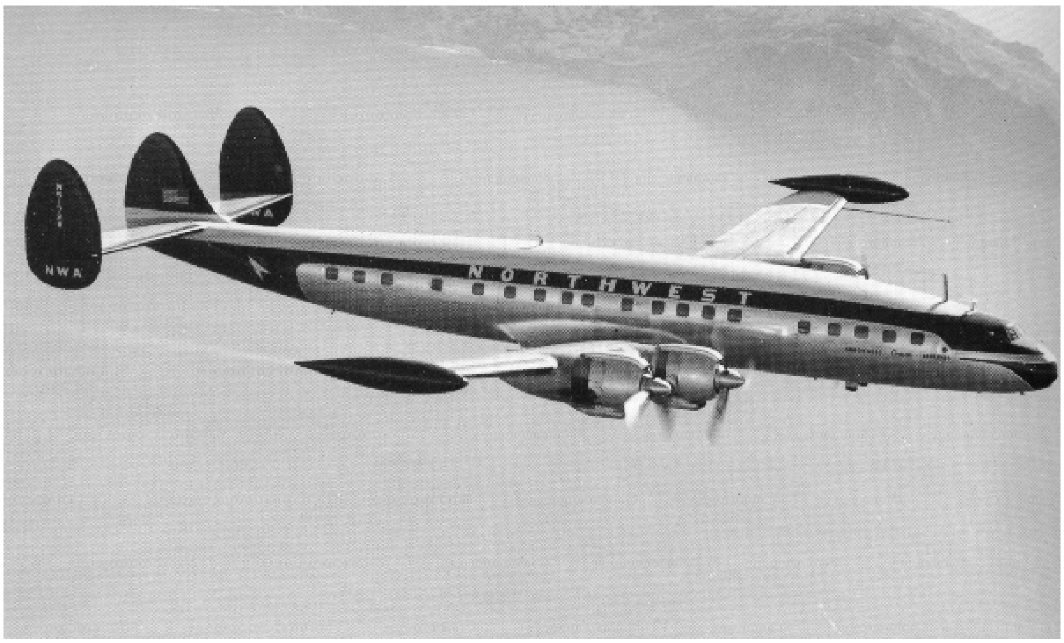


Figure 6.14 Lockheed L-1049 Constellation (Not accident aircraft, Courtesy www.prop-liners.com and United Airlines)

Several of the 26 passengers onboard received injuries from the burning. The crew escaped without injuries. The airplane was destroyed by the fire.

According to Ref. 6.11 the aircraft had taken off from the Seattle-Tacoma International Airport near Seattle under marginal weather conditions with destination Chicago, Il. The ceiling at the time of take-off was 200 ft.

After the airplane became airborne it was immediately in instrument conditions. At that point the No. 3 propeller over-sped. Attempts to correct this were not successful and the propeller was feathered but continued to rotate at 400 r.p.m. which produced extra drag.

The take-off alternate was Yakima, WA a distance of 122 miles but requiring over-flight of high terrain. That is why the captain elected to alternate to Portland, OR. However, when the airplane reached 5,000 ft (now above the overcast) the No. 4 engine showed excessive oil temperatures and its oil quantity diminished rapidly. The No. 4 propeller was also feathered and an emergency was declared with intent to land at McChord Air Force Base, near Tacoma, WA where a GCA (Ground Controlled Approach) approach was conducted.

So, both engines on the right were shut down and the No. 4 propeller continued to rotate, again causing extra drag. A go-around was no longer an option and the GCA landing had to be perfect.

When the captain requested take-off flaps during the initial stages of the approach the flaps would not extend hydraulically. The co-pilot had to go into the passenger cabin to hand-crank the flaps down. He found that it was not possible to turn the crank more than 15-20 turns (normally it takes 100 turns to get the flaps down manually). He then was asked to return to the cockpit to assist the captain in maintaining control of the airplane.

When the airplane broke out of the overcast at 500-800 ft above terrain the gear down command was given but only the right main gear extended fully and locked. However, the airplane was now committed to land and ended up crashing. There was a fire but all aboard were evacuated, some with minor burn injuries.

6.11.2 Cause

To understand the flap and gear problems encountered by the crew it is useful to describe some aspects of the design of the hydraulic system of this airplane.

Each of the four engines drives a hydraulic pump. Those on Nos. 1 and 2 engines furnish jointly or individually (in case 1 or 2 has failed) hydraulic pressure to supply boost for the aircraft flight controls and for certain other purposes not material to this case. This is known as the primary hydraulic system.

Pumps on engines No. 3 and 4 furnish jointly or individually (in case 3 or 4 has failed) hydraulic pressure for wheel braking, nose-wheel steering, wing flap motion, landing gear extension or retraction, and for certain other purposes not material to this case. This is known as the secondary hydraulic system.

A design feature of the system was that change-over from the secondary system to the primary system was possible but the reverse was not.

If the Nos. 3 and 4 engines are inoperative (as was the case here), there is no means of obtaining nose-wheel steering, wing flaps must be cranked down manually, and the landing gear must be lowered with the hydraulic hand-pump. In this case the only source of hydraulic pressure available from the secondary system during its emergency was the hydraulic pump driven by the wind-milling No. 3 engine. The result was an abnormally low volumetric output.

A small internal leak was found in the landing gear selector valve when in the "neutral" position. The leak was caused by an improperly seated poppet valve which permitted flow from "pressure" port to "down" port. Since the "down" port is connected internally to the "return" port when the selector valve is in neutral a leakage path was provided between the pressure and return lines. This leakage at the landing gear selector valve prevented normal flap extension, due to insufficient hydraulic pressure.

With the flap control in the "take-off" position and the flaps still retracted, the existing hydraulic pressure of 1,000 to 1,100 p.s.i., and the reduced output of the No. 3 pump, an abnormally slow extension of the landing gear resulted. It would have required an additional two or more minutes to fully extend and lock all gears but the airplane was too far along its final approach to permit this.

6.11.3 Solution

Lockheed issued a service bulletin recommending that certain changes be made in the hydraulic system of all Constellations to allow for drawing hydraulic pressure from the primary system into the secondary system. Lockheed also incorporated this change in all future Constellations.

6.11.4 Lesson

This scenario, knowing the design features of the hydraulic system, was predictable and, with proper design reviews should have been predicted and corrected before certification.

6.12 Hydraulic System Design III

6.12.1 Problem

In 1954 a KLM Lockheed Super Constellation (see Figure 6.15) crashed into the sea directly off the coast of Shannon, Ireland. There were 28 fatalities among the 56 people on board. The entire flight crew survived the ditching.

Figure 6.15 Lockheed Super Constellation (Not accident aircraft, Courtesy John F. Ciesla)

6.12.2 Cause

The accident history and its investigation are described in Ref. 6.12.

The landing gear was found in a partially down condition, the flaps were completely retracted. Right after liftoff the captain commanded the landing gear to retract. There is a light on the instrument panel which indicates when the gear is up and locked. That light was burned out. As soon as the captain thought the gear had retracted he selected flaps up. In the flight manual of this airplane there are instructions to not retract flaps until the gear is up and locked.

Flight tests on another Super Constellation revealed that if the flaps are selected to retract before the landing gear is up and locked, the hydraulic system can't cope with both. The flaps are then retracted as commanded while the gear retraction process is slowed down or, in some cases, will even extend again. The entire process takes about 38 seconds and during part of this time the drag of the airplane is significantly increased. Since power is typically brought back to climb power, after the gear is up and locked, this can result in a descent.

A significant contributing factor to the crash was the fact that the takeoff was conducted toward a pitch dark background (it was night) and that the altimeter and rate-of-climb indicators were subject to a certain lag giving the captain a false indication that all was well.

6.12.3 Solution

Design engineers should design gear and flap retraction systems so that simultaneous retraction commands do not yield unexpected results such as flaps retracting while the gear will extend again.

6.12.4 Lessons

1. In hydraulic system design make sure that the system has adequate fluid flow to continue raising the landing gear even if simultaneous flap retraction is initiated by the crew. Also, it would be desirable to foresee certain crew actions which are contrary to flight manual instructions and simulate them to check for unexpected consequences.

2. Modern regulations require shorter gear retraction times than what was allowed in this case.

6.13 Fuel System Design I

6.13.1 Problem

Sometime early in 1956 a Cessna 310 (Figure 6.16) was in a prolonged, powered descent when both engines stopped. The airplane was over very inhospitable terrain and crash-landed, killing the pilot.

Figure 6.16 Cessna 310 (Not accident aircraft, Courtesy Mick West)

6.13.2 Cause

The cause was found to be fuel starvation, despite the fact that there was plenty of fuel on board. The flight manual of the airplane reminds pilots to first use the tip-tank fuel and then switch to the main wing tanks. The pilot, by mistake, had reversed this recommended procedure. Figure 6.17 shows the attitude of the tip-tank during a descent.

Note the fuel level in the tip-tank. Since the only fuel pick-off point in the tip-tank is located near the aft end, at some point no fuel can reach the engines.

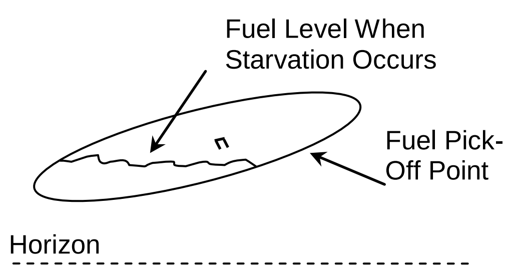


Figure 6.17 Tip-tank Attitude During a Prolonged Descent

6.13.3 Solution

The solution was to add a second fuel pick-off point near the forward end of the tank.

6.13.4 Lessons

1. In designing a fuel system layout, all possible airplane attitudes during which normal engine operation should be expected, must be considered.

2. In FAR 23 airplanes which can be expected to be flown by pilots with fairly low skill levels, reliance on procedures in the flight manual is not sufficient.

6.14 Fuel Vent Design II

6.14.1 Problem

In mid 1956 during a flight test of the Cessna T-37 (Figure 6.18) the test pilot reported that the airplane was becoming increasingly left wing heavy.

Figure 6.18 Cessna T-37 (Not accident aircraft, Courtesy NASA)

The airplane was landed using full right aileron.

6.14.2 Cause

The cause was established to be involuntary asymmetric fuel transfer from the right wing into the left wing. The reason for the fuel transfer was determined to be the unintentional asymmetric installation of the left and right wing tip fuel vents.

The angular difference between the two vents which were installed by hand without the use of a template was enough to result in pressure differences forcing fuel from the right wing into the left wing tanks.

6.14.3 Solution

Specify tolerances for the installation of fuel vents and use templates to assure their proper installation.

6.14.4 Lesson

Murphy's Law was at work. If a component can be installed the wrong way, someone will do it. Flight critical installations should be designed for one-way fits.

6.15 Cabin Door Design I

6.15.1 Problem

In June of 1956 a Piedmont Airlines Douglas DC-3 (Figure 6.19) passenger fell to his death through the main cabin door which was opened in flight.

The air stair door received minor damage but the airplane made a normal landing at its destination.

Figure 6.19 Douglas DC-3 (Not accident aircraft, Courtesy Frank C. Duarte)

6.15.2 Cause

According to Ref. 6.13 the probable cause was the accidental opening of the cabin air-stair door in flight.

Figure 6.20 shows an interior view of the cabin door, facing a closed door.

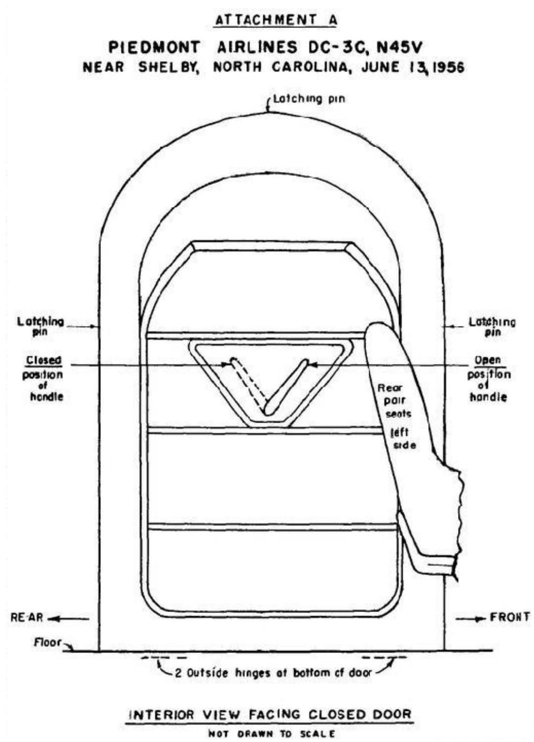


Figure 6.20 Interior View of Closed Cabin Air Stair Door
(Courtesy Civil Aeronautics Board)

The investigation revealed that it was possible to open the door accidentally by simply moving the lever from the closed to the open position. The particular passenger who apparently did so accidentally was said to have been under the influence of alcohol even before boarding the airplane.

It was found that many operators of similar aircraft with similar doors had installed safety devices to prevent accidental openings in flight.

6.15.3 Solution

The CAB recommended to the FAA that an Airworthiness Directive be issued requiring correction of this unsafe condition.

6.15.4 Lesson

The scenario which transpired was predictable. Installing a door without the proper safeguard against accidental opening in flight is unconsciouble. It is, now, also against the airworthiness regulations.

6.16 Fuel System Design II

6.16.1 Problem

Early in 1959 a B-52G (Figure 6.21) returning from a training mission was on final approach to a SAC (Strategic Air Command) base.

There was about 30% fuel left in the wing tanks. Therefore, the airplane was at a rather low weight. Because of a fouled runway the airplane commander decided to initiate a go-around maneuver. The airplane was observed to pitch up, stall, crash and burn, killing all crew members. About a week later the same scenario caused the loss of another B52G airplane and crew. This time the fleet was grounded and an intensive investigation was carried out involving many Boeing engineers.

Figure 6.21 Boeing B-52G (Not accident aircraft, Courtesy NASA)

6.16.2 Cause

The cause was established as follows. Standard procedure in initiating a go-around maneuver is to advance all eight throttles forward and to pitch up the nose a bit. As a result of the large forward acceleration (the airplane was at a low weight) the fuel was forced toward the outboard wing which moved the center of gravity behind the aerodynamic center. This made the airplane statically unstable, so it pitched up sharply and the pilot lost control. Obviously some design change had to be made.

6.16.3 Solution

Retroactively installing more baffles in the wing tanks would have been an obvious, albeit expensive, solution. A simpler and much lower cost solution was developed and adopted. A detent was installed in the throttle quadrant, as shown in Figure 6.22.

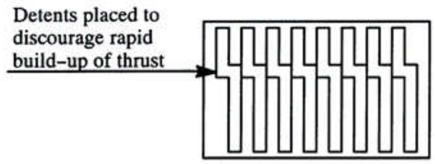


Figure 6.22 Detent in Throttle Quadrant of the Boeing B-52G, Stratofortress

The detent was placed so that it served as a warning to the pilot that suddenly demanding more thrust might result in a problem. The pilot could, if needed, move the throttles further forward beyond the detent, but would hopefully do so slowly. This solution has worked well: there have not been any more problems with the B52G and H models due to fuel motion.

Note: in commercial airplanes the solution depicted in Figure 6.22 would not be acceptable.

6.16.4 Lessons

1. Simplicity is often preferred. Sometimes a $25 solution can be made to work to solve a $25 million problem.

2. Obviously, engineers should have predicted this entire scenario and designed the fuel tanks so that rapid fuel transfer is not possible under foreseeable longitudinal accelerations.

6.17 Cargo Compartment Light Causes Fire

6.17.1 Problem

In March of 1959 a Riddle Airlines, Inc. Curtiss C-46R Commando (Figure 6.23) cargo airplane suffered an intense, uncontrollable fire which destroyed the integrity of the flight control system.

The crew transmitted a Mayday call indicating that they were on fire and had lost control of the airplane. The airplane crashed fatally injuring the two crew members. The airplane was partially consumed by the fire.

Figure 6.23 Curtiss C-46 Commando (Not accident aircraft, Courtesy Bob Garrard)

6.17.2 Cause

Ref. 6.14 lists as the probable cause ignition of cargo in the aft belly compartment caused by contact with an unguarded light bulb.

The following has been adapted from Ref. 6.14.

It was found that extensive in-flight fire damage destroyed the integrity of the flight control systems. The aileron bell-crank assembly and the rudder and elevator cable pulley clusters, all located at and just to the rear of the rear spar, were destroyed by fire. Additional heavy in-flight fire damage was found from the aft cargo bin forward to and including the ventilation louvers. Several pieces of aircraft structure from this area were found back along the flight path.

Samples of the cargo which was loaded in the “G” compartment were tested and, although found to be capable of supporting combustion, exhibited no unusual ignition characteristics. Studies were also conducted to determine whether a light bulb in contact with a mail sack filled with similar materials could develop sufficiently high temperatures to cause ignition. From these tests and studies of histories of baggage compartment fires, it is apparent that ignition could occur in such a manner.

The “G” compartment on Riddle C-46R aircraft was certificated by the FAA as a Class D compartment (CAR 4b 383) on the basis of airflow rate tests and engineering specifications submitted by the company. However, the company, as of the date of the accident, had not installed guards to protect the compartment lights from damage by cargo as required by the approval. In addition, no procedures were in existence requiring that these lights be turned off. Further, no procedure existed which required maintenance personnel to check the condition of the tape used to seal the joints of the compartment. This tape is essential to the installation to limit the airflow through the compartment to a rate not to exceed 1,500 cu.ft./hr. An inspection of the "G" compartment of a sister ship revealed that the tape can be scuffed and pulled away from the joints by cargo movement, thus destroying its effectiveness.

6.17.3 Solution

The company installed guards on all cargo compartment lights in is C-46R aircraft after this accident. It also instituted a program to install fire-detection and fire-extinguishing equipment in the lower cargo compartments.

The Safety Board recommended to the FAA that:

- cargo compartments lights should be off when in flight
- all cargo compartment lights should be guarded
- electrical relays and electric terminals in the hydraulic compartment should be protected.

6.17.4 Lesson

If existing regulations had been followed this accident most likely would not have happened. It would seem that management and maintenance personnel should be required to take a course in ethics and safe operating practices.

6.18 Cabin Door Design II

6.18.1 Problem

In October of 1962 an Allegheny Airlines Convair 340/440 (Figure 6.24) experienced an explosive decompression when the rear cabin service door became disengaged at its lower latch points.

Figure 6.24 Convair 340/440 (Not accident aircraft, Courtesy David Schulman)

The ensuing outward rush of air ejected a flight attendant who was near the door opening. The airplane landed without further incident.

It is noted in Ref. 6.15 that, following departure from Philadelphia, a high frequency whistling noise was heard and inspection revealed an escape of air at the lower aft corner of the rear service door. Pillow cases were placed in this area to reduce the air noise but no further action was taken until the door blow-out occurred.

6.18.2 Cause

Ref. 6.15 states as the probable cause the undetected insecure latching of the rear service door resulting in an in-flight explosive decompression which ejected a hostess from the airplane. Contributing factors were Allegheny Airlines' inadequate emergency pressurization instructions, and the continuation of pressurized flight after discovery of the pressurization leak.

The following has been adapted from Ref. 6.15.

This particular service door employs four latching hooks, two at the top of the door and two on the bottom. When the upper two latches are properly engaged micro-switches energize a cabin door warning light. This light indicated that the door was locked. However it would so indicate even with the lower latches improperly secured. Furthermore, it is easy to overlook by visual inspection whether or not all four latches are properly engaged.

It is of interest to observe that in June of 1954 (eight years before this accident) Convair issued Service Bulletin 126A which recommended improvements in the door latch and warning system. The CAB, in March of 1955, recommended to the CAA (predecessor to the FAA) that an Airworthiness Directive be issued which would have made mandatory the changes noted in this Service Bulletin. The CAA merely issued a recommendation that air carriers do so but failed to make this mandatory.

Convair issued additional Service Bulletins regarding this door in 1955, 1956, 1957 and 1958. Only the first recommendations had been incorporated in the accident aircraft.

6.18.3 Solution

In November of 1962 the FAA issued an Airworthiness Directive (effective on Dec. 18, 1962) which required essentially that all Convair recommendations relative to this door be complied with by all air carriers operating these aircraft.

For design engineers it is of interest to not that this AD contained, among others, the following requirements:

- The Airplane Flight Manual be revised to require inspection of the latching before take-off and each time the rear service door is operated;
- The aircraft be depressurized if there is evidence of a latch disengagement or leakage around the door;
- Inspection holes and lights be installed for inspection of the lower door latches; and
- Door latching electrical warning switches be installed in the upper and lower forward latches.

6.18.4 Lessons

1. Great care has to be exercised in the design of any door. Engineers should remember that, in actual service operation, these doors are not handled by highly trained or highly paid people. Service testing of doors, to make sure they continue to be safe, should be carried out before certification.

2. It seems unreasonable to design a door-latch warning system that can indicate safe when not all latches are engaged.

6.19 Design for Lighting Strikes

6.19.1 Problem

In December of 1963 a Pan American World Airways Boeing 707-121 (Figure 6.25) was struck by lightning which ignited fuel in an outboard wing tank.

The airplane crashed near Elkton, Maryland. All 73 passengers and the crew of 8 perished.

Figure 6.25 Model of Boeing 707-121 (Courtesy geminijets.com)

6.19.2 Cause

In Ref. 6.16 the probable cause is listed as lightning induced ignition of the fuel/air mixture in the No. 1 reserve fuel tank with resultant explosive disintegration of the left outer wing and loss of control.

There were many witnesses who testified that they actually saw lightning strike the airplane. Many also testified that the airplane was on fire and disintegrated shortly after the strike. Wreckage was spread over a wide area.

The accident investigation was hampered a great deal by the level of destruction and scattering of components. Although not agreed to by all involved in this investigation it seems that the problem began with lightning striking the wing tip area near the outboard fuel vent outlet.

Figure 6.26 shows an arrangement drawing of the 707-121 fuel tanks and vent surge tank. Note the proximity of this tank to the wing tip. In modern transport design the most outboard position for a surge tank is taken to be at about the 85% span station unless heavier skin gauges are used to prevent lightning penetration. Also note (in Figure 6.26) the absence of static discharge wicks which are now required on all transports.

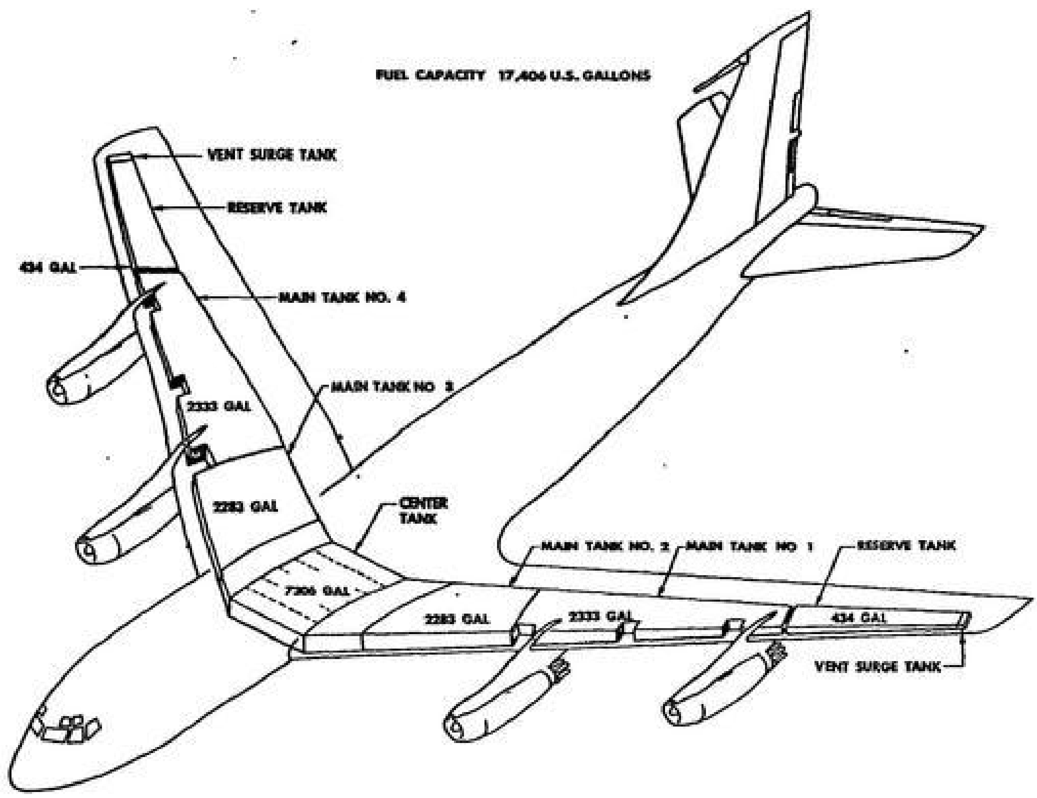


Figure 6.26 General Arrangement of Fuel Tanks in the Boeing 707-121
(Courtesy Civil Aeronautics Board)

6.19.3 Solution

The airplane was flying in an area of significant lightning activity. Today, pilots are urged to avoid flying through such areas. Several recommendations were made to the FAA to improve the survivability of airplanes when struck by lightning. Among these were:

- Installation of static discharge wicks on all turbine powered airplanes.
- Installation of flame arrestors in fuel tank vent outlets.

- Do not install surge tanks close to the wing tip unless extra thick skin gauges are used (this assumes aluminum construction)
- Use only Jet A fuel which is less flammable than Jet B. The accident airplane contained a mixture of Jet A and B.
- Develop a means to inert the space above the fuel by the introduction of an inert gas.

6.19.4 Lessons

1. Regardless of all thunderstorm avoidance procedures airplanes will continue to be struck by lightning. The modern regulations recognize this and fuel systems must be designed accordingly.

2. The introduction of inert gasses has recently received an additional stimulus because of the fuel tank explosion on board a TWA Boeing 747. It is interesting to note the inertia in aeronautics design: this idea was already proposed in the 1960's and 40 years later it still is not incorporated in new designs (exceptions are the Airbus 380 and Boeing 787).

6.20 Fuel Lines Close to Landing Gear Brace

6.20.1 Problem

In November of 1965 a United Air Lines Boeing 727-100 (Figure 6.22) crashed during an attempted landing at the Salt Lake City airport in Utah.

The crash itself was survivable, however, a fire broke out. As a result of the fire and smoke, 85 passengers and 6 crew members on board, 43 passengers died.

Figure 6.27 Model of Boeing 727-100 (Courtesy geminijets.com)

6.20.2 Cause

Ref. 6.17 states as the probable cause of this accident the failure of the captain to take timely action to arrest an excessive descent rate during the landing approach.

Investigations showed that the airplane, when at the outer marker, was flying at 200 kts and 2,000 ft above the glide-slope. In the ensuing seconds the landing configuration (40 degrees of flaps and landing gear down) as well as the reference speed of about 125 kts were established but the rate of descent was 2,300 ft/min (this is three times the recommended descent rate for the airplane at this point of the approach) with the airplane still 1,300 feet above the glide-slope. Although the co-pilot tried to apply more power, the captain indicated to wait with that action. That decision aggravated an already perilous situation. The airplane hit the ground very hard about 335 ft from the runway threshold.

Both main landing gears broke from their main attachment points (the gear is stressed to withstand a vertical impact velocity of 12.5 ft/sec, or 750 ft/min at the design landing weight). The severe upward and rearward impact forces from the right main landing gear assembly produced a large impact hole in the fuselage and ruptured fuel lines and the No. 3 generator leads between fuselage stations 1030 and 1130 on the right side. The fuel was ignited by sparks from the fuselage scraping on the runway and/or the severed generator leads.

The resulting fire made evacuation of many passengers impossible.

To understand the physics of the events some insight into the high lift system and the flight characteristics of the 727 is helpful. The 727 was designed with an extremely effective high lift system to allow the airplane to meet a relatively short field-length requirement which existed (at that time) at the LaGuardia Airport. There are four flap settings available. Before lowering the flaps from setting 3 to setting 4 (40 degree flaps) it is necessary to advance the throttles to help overcome the large increase in drag which accompanies this flap change. One reason is the long spool-up time of the JT8D engines.

This required pilot action is clearly identified in the flight manual and pilots are trained to do this. A consequence of going from flaps 3 to flaps 4 without advancing the throttles first is that a high sink rate develops. If the airplane is already close to the ground, advancing the throttles after lowering the flaps from 3 to 4 will not allow the sink rate to be arrested before touching down at a high sink rate.

The landing itself, although very hard, was survivable. The fire was not.

6.20.3 Solution

It was noted that in this severe impact case the landing gear broke off but the fuselage belly structure did not collapse.

As a result of this crash three recommendations for redesign were proposed:

- Fuel lines through the fuselage should be re-routed so that they pass through the floor beams near the centerline of the aircraft.
- Fuel lines and their shrouds should be made of stainless steel and should have a thickness of sufficient dimensions to withstand survivable impacts.
- Generator leads should be routed so that there exists maximum separation between these leads and fuel lines. Each generator lead should be placed in a separate plastic conduit with suitable strength and flexibility.

In Ref. 6.18, page 51 the following statement is found:

“Following investigation, the Civil Aeronautics Board (predecessor to the NTSB) recommended specific design changes to the FAA. As a result, CFR 25.993(f) was revised to state:

Each fuel line within the fuselage must be designed and installed to allow a reasonable degree of deformation and stretching without leakage.”

All Boeing 727 aircraft had their fuel lines redesigned and relocated so that the likelihood of a reoccurrence is small.

6.20.4 Lesson

The events leading to this crash were caused by a pilot who did not use proper flying procedures. Nevertheless, these events were predictable but were not predicted. Design engineers should keep this clearly in mind when laying out fuel lines and landing gear components which are likely to break in a high descent rate landing.

6.21 Loss of Pitch Control Due to Fire

6.21.1 Problem

In June of 1967 a Mohawk Airlines BAC 1-11 (Figure 6.28) while climbing to 16,000 ft caught fire in the rear of the fuselage and was observed to dive into the ground.

Figure 6.28 BAC-1-11 (Not accident aircraft, Courtesy of www.al-airliners.be)

Witnesses on the ground saw dense smoke emanating from the rear of the airplane. They also saw large pieces falling from the airplane. All 34 persons aboard the airplane were killed.

6.21.2 Cause

According to Ref. 6.19 the probable cause of this accident was the loss of integrity of the empennage pitch control systems due to a destructive in-flight fire which originated in the airframe plenum chamber and, fueled by hydraulic fluid, progressed into the vertical fin. The fire then compromised the pitch control systems of the airplane. In-flight airframe failure also originated at the vertical fin.

The fire resulted from engine bleed air flowing back through a malfunctioning non-return valve and an open air delivery valve, through the auxiliary power unit (APU) in a reverse direction, and exiting into the plenum chamber at temperatures sufficiently high to cause the acoustics linings to ignite.

The problem was compounded by the fact that insufficient drainage had been provided in the basic design to assure that leaking hydraulic fluid would not become absorbed in acoustical linings which then serve as a source of fuel for any fire.

6.21.3 Solution

Following the Civil Aeronautics Board (CAB) investigation the FAA issued an Airworthiness Directive, AD No, 68-1-1. This AD required that, if the use of the APU in flight is to be continued the following must be accomplished (Ref. 6.19, page 54):

1. Replace the non-return valve with a modified valve.

2. Perform the following structural modifications:

    - Install additional fireproof, stainless steel skin over the existing light alloy outer skin on top of the fuselage, between Fuselage Stations 936 and 958, in order to isolate the airframe plenum chamber from the vertical fin.

- Replace the light alloy wall separating the airframe plenum chamber from the hydraulic compensator bay with a stainless steel wall, enlarging the hydraulics bay in the process.
- Modify the hydraulic compensator drain box and the drain outlet.
- Install a revised spring loaded door in the bulkhead at FS 936 (forward of the plenum chamber)

3. Install sealing plates around the control guard, located above the rudder power control units, and over the hole in the vertical fin rear spar, in order to restrict airflow into the vertical fin.

4. Install an additional bimetallic temperature sensor parallel with the existing mercury sensor in the circuitry which controls the electrically actuated primary temperature valve located in the low pressure bleed flow duct to the heat exchanger.

5. Revise the airplane flight manual to assure that at no time is air from either engine and from the APU being delivered simultaneously into a common duct.

6.21.4 Lessons

1. As the AD suggests, there were a lot of design problems with this particular system in this airplane. Also, the events leading to this crash were predictable, should have been predicted but were not.

2. In a complex system it is not sufficient to rely on the assumption that flight crews will always follow or comprehend instructions for the operation of such a system. Design engineers should expect the worst and design the system so that a simple valve failure does cause a catastrophe.

6.22 Confusing Systems Design

6.22.1 Problem

The Piper Apache (Figure 6.29) was a fairly popular, low cost, twin-engined, propeller-driven airplane. The author received his multi-engine instruction and rating on this airplane. Interesting system design features were:

- If the left engine failed, all electrical power was lost. As a result some of the flight instruments would no longer work.
- If the right engine failed, all hydraulic power was lost and the landing gear had to be operated with a hand-driven pump.

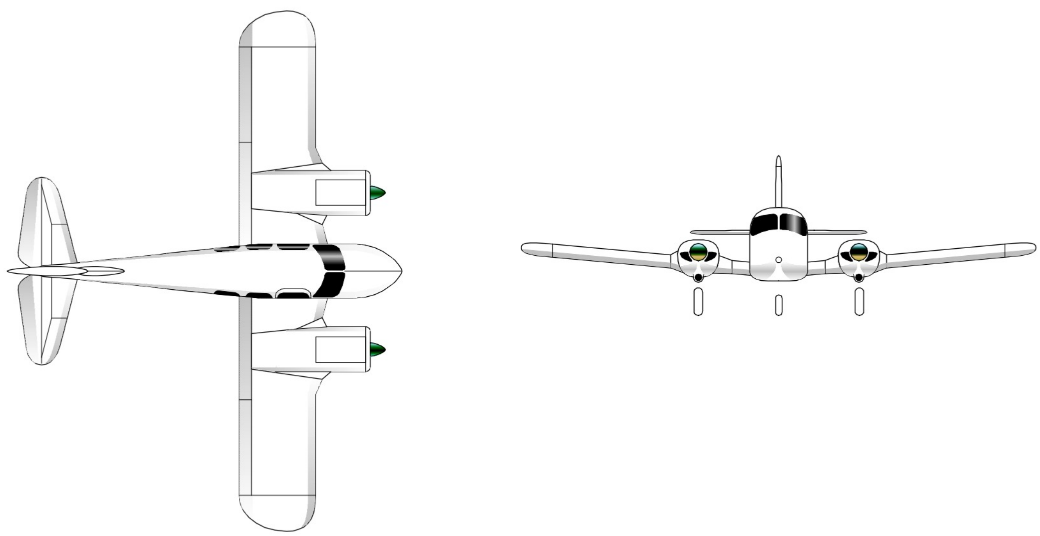


Figure 6.29 Piper Apache

A problem with this type of design is that the pilot is supposed to know the airplane and its systems well enough to be able to cope with engine failures. It is asking a lot of the human mind to assume that most pilots can cope with such complexities in an emergency.

6.22.2 Lessons

1. Engine failures should be non-events as far as airplane controllability is concerned.

2. Systems should not be designed so that a lot of intellectual activity is required on the part of a not very experienced pilot.

6.23 System Redundancy Saves the Day I

6.23.1 Problem

In July of 1971 a Pan American World Airways Boeing 747-100 (Figure 6.30) struck the approach light structure (ALS) for runway 19L while taking off from Runway 01R at the San Francisco International Airport. Two passengers were injured during the impact with the ALS. The crew continued the take-off and, after an in-flight inspection for damage, dumped fuel and returned for a landing at San Francisco. Twenty-one passengers received minor injuries and eight others sustained serious back injuries during the evacuation after the landing.

Figure 6.30 Model of Boeing 747-100 (Courtesy geminijets.com)

6.23.2 Cause

In Ref. 6.20 the NTSB determines that the probable cause of this accident was the pilot's use of incorrect take-off reference speeds for the runway they were using. This resulted from a series of irregularities involving: 1) the collection and dissemination of airport information; 2) aircraft dispatching; and 3) crew management and discipline; which collectively rendered ineffective the air carrier's operational control system.

When the airplane struck the ALS three of the four hydraulic systems (1, 2 and 4) operating the flight controls were disabled. System 3 kept functioning. The right body landing gear truck separated from the airplane, the left body gear truck was hanging down with two wheels missing and there was significant structural damage to the fuselage structure.

6.23.3 Solution

The NTSB made a series of recommendations aimed at improving crew discipline and the proper transmission of critical airport information. The NTSB also expressed concern about passenger cabin ceiling panel design. Some of these panels fell into the cabin in such a way that they could have blocked passenger attempts to escape the cabin.

6.23.4 Lessons

Although only three hydraulic systems are required for certification of transport aircraft, the decision by Boeing to put four systems in the 747 saved lives.

Ceiling panels in passenger airplanes should be designed so they can withstand hard landings. These panels should stay in place under survivable vertical accelerations.

6.24 Engine Failure Precipitates Brake Failure

6.24.1 Problem

In November of 1975 an Overseas National Airways Douglas DC-10-30 (Figure 6.31) crashed during an attempted take-off from Jamaica, New York (JFK Airport).

During the take-off roll the airplane struck sea gulls and the take-off was rejected. Engine Number 3 disintegrated and caught fire. Several tires and wheels disintegrated; and the aircraft did not decelerate as expected. Near the end of the runway the captain steered the airplane onto a taxiway; the landing gear collapsed and most of the aircraft was destroyed by fire. Of the 139 persons on board, two were seriously injured and 30 were slightly injured.

Figure 6.31 Douglas DC-10-30 (Not accident aircraft, Courtesy Michel Gilliand)

6.24.2 Cause

In Ref. 6.21 the NTSB determines that the probable cause of this accident was the disintegration and subsequent fire in the No.3 engine when it ingested a large number of seagulls. Following the disintegration of the engine the aircraft failed to decelerate effectively because:

1. the No.3 hydraulic system was inoperative, which caused the loss of the No.2 brake system and braking torque was reduced 50 percent;
2. the No.3 engine thrust reversers were inoperative;
3. at least three tires disintegrated;
4. the No.3 system spoiler panels in each wing could not deploy;
5. the runway surface was wet

6.24.3 Solution

The NTSB recommended that the FAA re-test the General Electric CF6 engines with regard to compliance with the bird ingestion criteria of AC 33-1A. The NTSB also made a large number of recommendations which had to do with engine certification and bird control at JFK. None of the recommendations were related to causes 1, 2 and 3. It is hard to understand why a brake system can be certified when it looses so much of its deceleration potential with the loss of one engine.

6.24.4 Lesson

This was a predictable scenario and, evidently was not. Design engineers who would have predicted this scenario should have come to the conclusion that redesign of the hydraulic brake system was needed to prevent this from happening. One should not need a regulation to do that.

6.25 Systems Design, Flight Crew Training and Improper Maintenance Procedures

6.25.1 Problem

In May of 1979 the No.1 engine and pylon as well as about 3 ft of the leading edge of the left wing separated from an American Airlines Douglas DC-10-10 (Figure 6.32) during take-off rotation at Chicago, O'Hare. The airplane climbed to about 325 ft and then began to roll to the left and pitch down after which it crashed. All 258 passengers and 13 crew members on board were killed. In addition, two persons on the ground were killed and two others were injured.

Figure 6.32 Model of Douglas DC-10-10 (Courtesy geminijets.com)

6.25.2 Cause

Refs. 6.22 and 6.23 cover the investigation into the cause(s) of this accident. The NTSB, in Ref. 6.22, lists the probable cause as:

"the asymmetrical stall and the ensuing roll of the aircraft because of the un-commanded retraction of the left wing outboard leading edge slats and the loss of stall warning and slat disagreement indication systems resulting from maintenance-induced damage leading to the separation of the No.1 engine and pylon assembly at a critical point during take-off. The separation resulted from damage by improper maintenance procedures which led to failure of the pylon's structure."

Obviously, a chain of events led to this disaster. It is instructive to list all (twenty-three) conclusions reached by the NTSB in Ref. 6.22:

1. The engine and pylon assembly separated at or immediately after lift-off. The flight crew was committed to continue the take-off.

2. The aft end of the pylon assembly began to separate in the forward flange of the aft bulkhead.

3. The structural separation of the pylon was caused by a complete failure of the forward flange of the aft bulkhead after its residual strength had been critically reduced by the fracture and subsequent service life.

4. The overload fracture and fatigue cracking of the pylon aft bulkhead's upper flange were the only pre-existing damage on the bulkhead. The length of the overload fracture and fatigue cracking was about 13 inches. The fracture was caused by an upward movement of the aft end of the pylon which brought the upper flange and its fasteners into contact with the wing clevis.

5. The pylon to wing attach hardware was properly installed at all attachments.

6. All electrical power to the No.1 a.c. generator bus and No.1 d.c. bus was lost after the pylon separated. The captain's flight director instrument, the stall warning system, and

the slat disagreement warning light systems were rendered inoperative. Power to these buses was never restored.

7. The No.1 hydraulic system was lost when the pylon separated. Hydraulic systems No.2 and No.3 operated at their full capability throughout the flight. Except for spoiler panels No.2 and No.4 on each wing, all flight controls were operating.

8. The hydraulic lines and follow-up cables of the drive actuator for the left wing's outboard leading edge slat were severed by the separation of the pylon and the left wing's outboard slats retracted during climb-out. The retraction of the slats caused an asymmetric stall and subsequent loss of control of the aircraft.

9. The flight crew could not see the wings and engines from the cockpit. Because of the loss of the slat disagreement light and the stall warning system, the flight crew would not have received an electronic warning of either the slat asymmetry or the stall. The loss of the warning systems created a situation which afforded the flight crew an inadequate opportunity to recognize and prevent the ensuing stall of the aircraft.

10. The flight crew flew the aircraft in accordance with the prescribed emergency procedure which called for the climb-out to be flown at V_2 speed. V_2 speed was 6 kias below the stall speed for the left wing. The deceleration to V_2 speed caused the aircraft to stall. The start of the left roll was the only warning the pilot had of the onset of stall.

11. The pylon was damaged during maintenance performed on the accident aircraft at the American Airlines Maintenance facility in Tulsa, Oklahoma, on March 29 and 30, 1979.

12. The design of the aft bulkhead made the flange vulnerable to damage when the pylon was being separated or attached during maintenance.

13. American Airlines engineering personnel developed an Engineering Change Order (ECO) to remove and re-install the pylon and engine as a single unit. The ECO directed that the combined engine and pylon assembly be supported, lowered, and raised by a fork-lift. American Airlines engineering personnel did not perform an adequate evaluation of either the capability of the fork-lift to provide the required precision for the task, or the degree of difficulty involved in placing the lift properly, or the consequences

of placing the lift improperly. The ECO did not emphasize the precision required to place the fork-lift properly.

14. The FAA does not approve the carrier's maintenance procedures, and a carrier has the right to change its maintenance procedures without FAA approval.

15. American Airlines personnel removed the bolts of the aft bulkhead and bushing before removing the forward bulkhead attach fittings. This permitted the forward bulkhead to act as a pivot. Any advertent or inadvertent loss of fork-lift support to the engine and pylon assembly would produce an upward movement at the upper flange of the aft bulkhead and bring it into contact with the wing clevis.

16. American Airlines maintenance personnel did not report formally to their maintenance engineering staff either their deviation from the removal sequence contained in the ECO or the difficulties they had encountered in accomplishing the ECO procedures.

17. American Airlines engineering personnel did not perform a thorough evaluation of all aspects of the maintenance procedures before they formulated the ECO. The engineering and supervisory personnel did not monitor the performance of the ECO to insure that it was being accomplished properly or if their maintenance personnel were encountering unforeseen difficulties in performing the assigned tasks.
18. The nine situations in which damage was sustained and cracks were found on the upper flange were limited to those operations wherein the engine and pylon assembly was supported by a fork-lift.

19. On December 19, 1978 and February 22, 1979, Continental Airlines maintenance personnel damaged aft bulkhead upper flanges in a manner similar to the damage noted on the accident aircraft. The carrier classified the cause of the damage as a maintenance error. Neither the air carrier nor the manufacturer interpreted the regulation to require that it further investigate or report damages to the FAA.

20. The original certification's fatigue damage assessment was in conformance with the existing requirements.

21. The design of the stall warning system lacked sufficient redundancy; there was only one stick-shaker motor; and further, the design of the system did not provide for cross-over

information to the left and right stall warning computers from the applicable leading edge slat sensors on the opposite side of the aircraft.

22. The design of the leading edge slat system did not include positive mechanical locking devices to prevent movement of the slats by external loads following a failure of the primary controls. Certification was based on acceptable flight characteristics with an asymmetrical leading edge slat condition.

23. At the time of DC-10 certification, the structural separation of an engine pylon was not considered. Thus, multiple failures of other systems resulting from this single event was not considered.

Figure 6.33 and Figure 6.34 show the general arrangement of the nacelle-pylon-wing attachments.

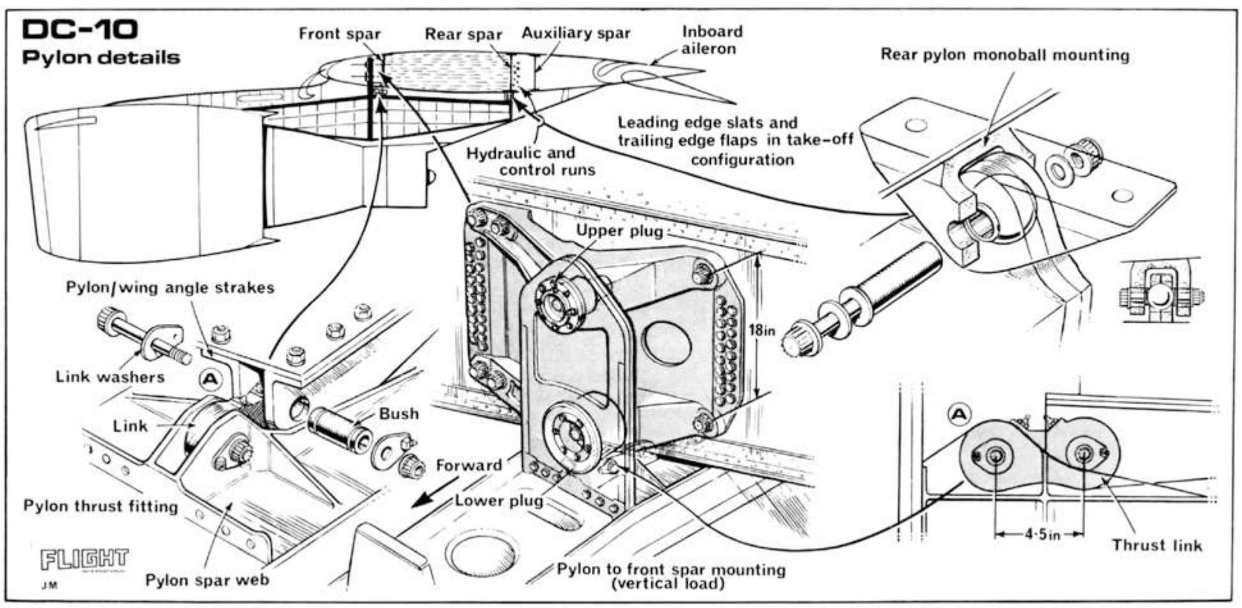


Figure 6.33 General Arrangement of the DC-10 Nacelle-Pylon-Wing Attachment Structure (Courtesy The Flight Collection)

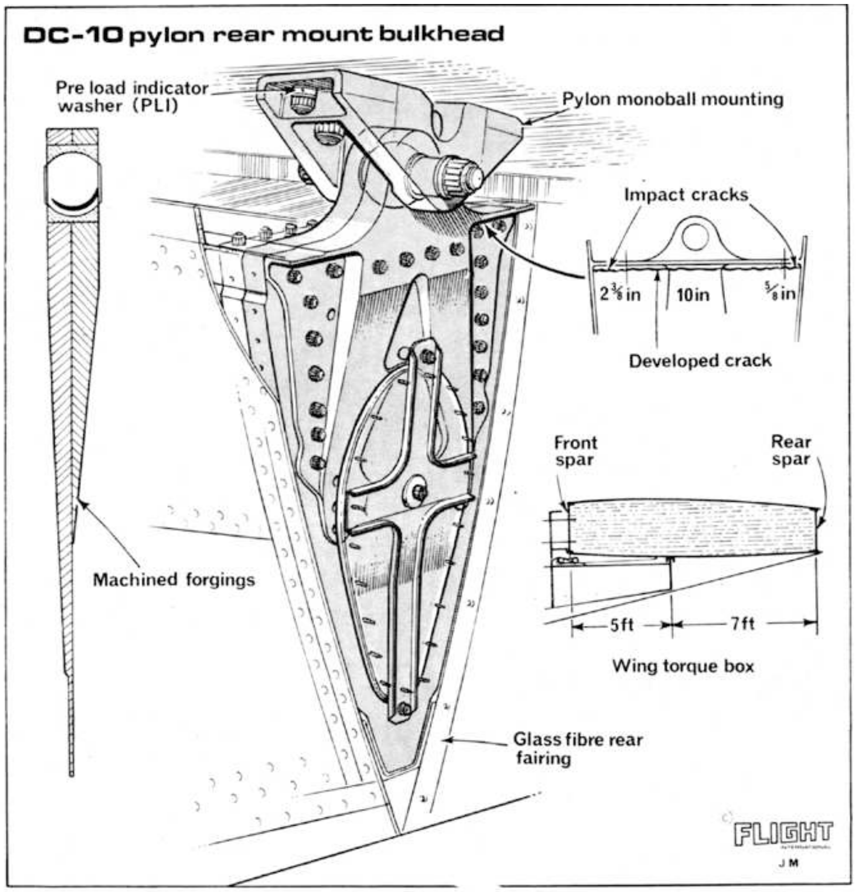


Figure 6.34 General Arrangement of the DC-10 Nacelle-Pylon-Wing Attachment Structure (Courtesy The Flight Collection)

It does not take a rocket scientist to predict that supporting the assembly with a commercial fork-lift (which, because of the large engine weight is located forward of the forward pylon bulkhead and which cannot possibly be positioned accurately) will impose large loads on the attachment fitting at the rear pylon bulkhead with the potential to inflict cracks. In fact this is what happened at the Continental Airlines Maintenance facility on two occasions. Continental made Douglas and other DC-10 operators aware of the problem via an Operational Occurrence circular. The maintenance engineers at American apparently did not see that circular.

It should be stated that Douglas never recommended nor condoned this type of maintenance procedure. The recommended Douglas procedure involved taking the engine off the pylon first, before dismounting the pylon.

6.25.3 Solution

The FAA issued an emergency order of suspension of the airworthiness certificate of the DC-10 series aircraft after it was discovered that similar cracks existed in other airplanes which had undergone similar maintenance procedures. The maintenance procedures at those carriers which had used fork-lifts in a similar manner were altered. Also, more redundancy was installed in the stall-warning systems and slat disagreement systems. The airworthiness certificate was reinstated and the airplanes returned to service.

6.25.4 Lessons

1. It is questionable to consider a nacelle-pylon-wing attachment structure in the same light as a wing structure. If the wing primary structure fails in flight, the result is a crash. That is a well known and accepted fact. However, if a nacelle-pylon-wing attachment fails that should not result in a crash. The flight controls and cockpit indicating and warning systems should therefore be designed to cope with such a failure. They were not.

2. Transport airplanes should not be certified with only one stick-shaker motor.

3. Stall warning and slat disagreement warning systems should not be powered from one source.

4. In a transport airplane any slat asymmetry which cannot be controlled at the stall speed of an airplane should not be allowed to develop: automatic slat brakes should be installed.

5. Designers should not need a regulation defining how to cope with items 1-4. The author believes this to be a simple matter of design ethics.

The author would like to offer the following comments:

- The FAA should have been more pro-active in basic design reviews early in the DC-10 program. Items such as lessons 1-4 should not have been allowed to slip by un-noticed.

- The FAA should also have been more pro-active in assuring that critical maintenance procedures are being carried out in line with the recommendations of the manufacturer.

- Finally, it must be observed that the flight crew had been trained to slow the airplane down to V_2 following an engine failure right after lift-off. That procedure has its origin in the propeller transport era when it was important to fly an airplane with one engine out at the best angle of climb speed.

The rationale at that time was the avoidance of an obstacle at the end of the shorter runways then in use. Modern jets take off from much longer runways and usually from airports without significant obstacles. With one engine out a jet transport will still be able to accelerate. It is a tragic fact that the DC-10 in fact could have been accelerated away from the left wing stall speed and, as simulator studies have shown been safely landed.

Food for thought from an operational viewpoint!

6.26 Service Lift Design

6.26.1 Problem

In September of 1981 a flight attendant on board a World Airways McDonnell-Douglas DC-10-30CF (Figure 6.35) became trapped between a service cart and the service lift door ceiling while the lift was being commanded up by another flight attendant in the passenger cabin galley. The flight attendant lost her life.

Figure 6.35 Model of McDonnell-Douglas DC-10-30CF (Courtesy geminijets.com)

6.26.2 Cause

In Ref. 6.24 the NTSB determines the probable cause to be:
"the malfunction of the galley personnel lift system door electrical interlock switches which permitted the galley personnel lift to rise with the door in the lower galley in the open position. Contributing to the accident was the design of the galley lift service cart retention and release system, and the inadequate pre-flight inspection program for the galley lift system.

Sadly, several flight attendants had observed the movement of the galley lift with the lower door open while the aircraft was still on the ground before the accident flight. They did not report this discrepancy to maintenance or flight personnel and there were no written procedures which required them to do so (author's note: what about common sense?).

The galley lift service cart had a retention and release system which had been troublesome on many occasions before. Because of limited space in the lifts releasing a stuck retention system required a flight attendant to bend over into the lift. If at that time someone else commands the lift to move up and the door interlock switches have failed, serious injuries (or worse) can be the result.

Sadly, Ref. 6.24 lists numerous cases where flight attendants were injured as a result of the same scenario.

Hindsight shows again that there were plenty of precursors but nothing had been done to either improve the design of the cart retention system or that of the lift command system.

6.26.3 Solution

The NTSB received suggestions for improvement from the director of safety of the Association of Flight Attendants. They made a number of suggestions to prevent this from happening again (see Ref. 6.24).

6.26.4 Lesson

Even simple systems like service cart retention devices and electric lifts can pose significant dangers and should therefore be subjected to a design review with persons who will operate such systems involved. That is part of what is now called: total engineering.

6.27 Leading Edge Slat Asymmetry

6.27.1 Problem

In September of 1981 an Air Florida McDonnell-Douglas DC-10-30CF (Figure 6.36) sustained an uncontained failure of its No.3 engine during the take-off roll.

Figure 6.36 Model of McDonnell-Douglas DC-10-30CF (Courtesy geminijets.com)

The engine failure occurred at 90 kts and the pilot aborted the take-off and brought the airplane to a safe stop. There were no injuries. The airplane sustained damage to the Nos. 1 and 3 hydraulic system (which lost all fluid and became inoperative), the electrical system, the No.3 engine control system and fire protection system sustained significant damage. One disturbing result was the un-commanded retraction of the right wing outboard leading edge slat (See also the 1979 DC-10 accident in Section 6.25.1).

6.27.2 Cause

In Ref. 6.25 the NTSB determines the probable cause to be: "the failure of quality control inspections to detect the presence of foreign material in the low pressure turbine cavity during re-assembly of the low pressure turbine module after installation of the stage 1 low pressure turbine rotor disk. The foreign material in the low pressure turbine cavity damaged the bolts holding together the stage 1 low pressure turbine rotor disk and the stage 2 low pressure rotor disk. The bolts failed at high engine thrust and the stage 1 low pressure disk separated from the low pressure turbine rotor assembly, over sped and burst."

6.27.3 Solution

The NTSB made a number of recommendations in regard to inspection procedures during engine maintenance.

With regard to the asymmetric slat situation the following is quoted from page 23 of Ref. 6.25:

"Since the design of the leading edge wing slat system was such that a malfunction could occur during slat operation, which could permit an outboard group of slats to either extend or retract asymmetrically, certification of this system was also based on flight data showing that the aircraft could be flown safely with one outboard group of wing slats retracted and the other in the take-off position within an airspeed range bounded by the stall warning speed and 260 KIAS, the limiting speed for the take-off slat position.

During the re-certification tests conducted after the O'Hare DC-10 accident (See Ref. 6.22) the data showed that the aircraft could take-off safely with all engines operating and the outboard wing slat group retracted on one wing. Although analysis of this indicated that the DC-10, in the event of an engine failure in addition to slat retraction, might not be controllable under certain conditions, it also showed that this particular combination of failures was extremely improbable. As a result, the aircraft was re-certified. The analysis conducted by McDonnell-Douglas after this accident further verified the data that this combination of failures was extremely improbable.

However, despite this, the decision was made to modify the wing leading edge slat system as provided for in AD 82-03-03. The Safety Board supports this decision and believes that where possible and economically feasible, designs should incorporate maximum safeguards regardless of the probability of occurrence." (Italics are by the author)

6.27.4 Lesson

Design engineers should keep the italicized text of Section 6.27.3 in mind with regard to all design aspects that deal with flight crucial systems.

6.28 System Redundancy Saves the Day II

6.28.1 Problem

In September of 1981 an Eastern Airlines Lockheed 1011-384 experienced an in-flight break-up and fire of the No.2 engine which severely damaged the aircraft structure and flight controls (Figure 6.37).

Figure 6.37 Model of Lockheed L-1011-384 (Courtesy geminijets.com)

The flight crew was able to land the airplane. There were no injuries.

6.28.2 Cause

In Ref. 6.26 the NTSB determines the probable cause to be “thermally induced degradation and subsequent failure of the No.2 engine low pressure location bearing (LPLB) because of inadequate lubrication. Oil leaks between the abutment faces of the intermediate pressure compressor (IPC) rear stub-shaft and the low pressure location bearing oil weir and between the intermediate pressure location bearing (IPLB) inner front flange and the intermediate pressure compressor rear stub-shaft reduced the lubricating oil flow to the low pressure location bearing which increased operational temperatures, reduced bearing assembly clearance, and allowed heat to build up in the bearing’s balls and cage. The bearing failure allowed lubricating oil to spray forward into the low pressure compressor/fan (LPC) shaft area where it ignited a steady fire; the fire overheated the fan shaft and the fan fail-safe shaft both of which failed, allowing the fan module to move forward and break through the No.2 engine duct. This caused severe damage to the aircraft structure and flight control systems. The oil leaks were most likely caused by poor mating of the abutment surfaces.

Figure 6.38 shows the general layout of the RB-211 engine and the location of the No.2 engine at the end of the S-duct.

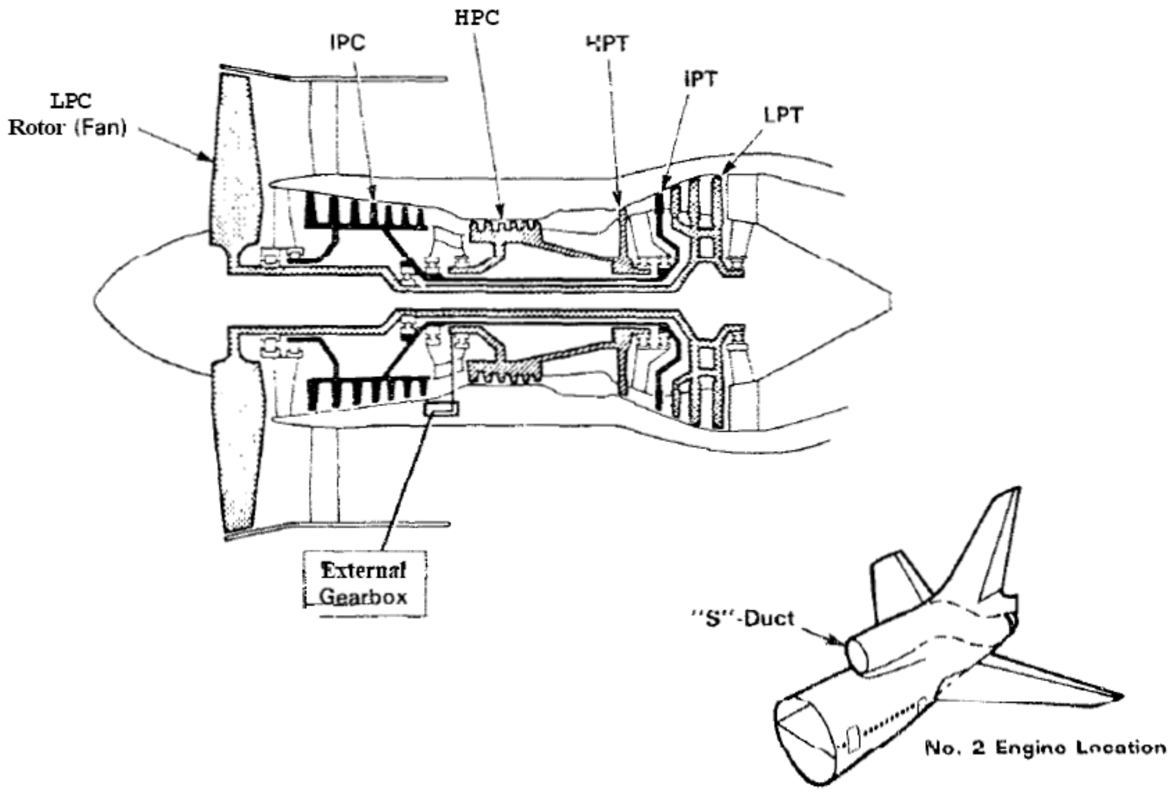


Figure 6.38 Layout of the RB-211 Engine and its Location in the S-duct

When the fan rotor came loose it broke through the duct and inflicted the following damage (see Ref. 6.26):

"The most significant damage which affected the captain's ability to control the aircraft was the disabling of three of the aircraft four hydraulic systems and the damage to the rudder control cables. The L-1011 like all modern wide-bodied aircraft depends upon the integrity of some hydraulic services for flight control. Redundancy is built into the system such that each of the four hydraulic systems is independent and can provide partial power to maintain flight control about each of the aircraft control axes. The systems are physically separated so that normally damage inflicted to a small area of the aircraft will not affect all of the hydraulic systems. A separation of the entire fan module, however, was not considered as a possible occurrence during the design of the airframe and, thus, was not an influencing factor in the placement of redundant systems. The extensive spread of debris from the fan module severed the fluid lines of three of the hydraulic systems. The fourth system sustained a damaged line; however, it was not severed and fluid pressure and capacity was retained. The system which remained provided control to the horizontal stabilizer, the inboard ailerons, the rudder, nose-wheel steering, and the alternate wheel brake system.

However, since the rudder control cables were jammed, preventing the movement of the rudder pedals, both rudder control and nose-wheel steering (the controls for which are interconnected to the rudder control cables) were rendered inoperable. Nevertheless, sufficient flight control was available for the captain to land and stop the aircraft without further incident. The fan module debris also damaged the electrical wire bundles and generator feeder cables to the No.2 engine-driven generator. However, all essential electrical services remained operable through a bus which interconnects the aircraft three electrical power sources.

Thus, the Safety Board believes that, while this accident clearly demonstrates the potential for a catastrophic accident as a result of separation of a major engine component which could cause major structural damage or render multiple redundant systems inoperable, the accident nevertheless demonstrates the value of system redundancy in the design philosophy of modern transport aircraft."

6.28.3 Solution

The NTSB made a number of recommendations with regard to engine maintenance procedures both to airlines and to engine manufacturers. The NTSB also urged further research and study into engine component containment technology.

Finally, the Board re-emphasized the need for detailed Failure Mode and Effect Analyses (FMEA) with regard to the effect of uncontained failures on structures and flight crucial systems.

6.28.4 Lessons

Uncontained engine failures keep occurring. In the case of the RB211 during the period from 8-12-78 through 9-22-81 there were ten uncontained failures. Fleet wide statistics in 1981 indicated 0.024 such failures per 1,000 operating hours for Rolls Royce and 0.001 per 1,000 operating hours for Pratt & Whitney JT9D and General Electric CF-6 engines.

From a design viewpoint these are events to be considered in the design layout of all flight crucial systems and structure. This includes separation of fan modules.

Designers should observe that here is another case (Section 6.23) where the presence of four hydraulic systems (only three would have been required to gain certification) saved the day.

6.29 Flap Asymmetry

6.29.1 Problem

In January of 1982 an Empire Airlines Piper PA-31 (Figure 6.39) crashed after the pilot reported that only one flap was down. The pilot and co-pilot, the only persons aboard, were killed.

Figure 6.39 Piper PA-31 (Not accident aircraft, Courtesy Howard Chaloner)

6.29.2 Cause

In Ref. 6.27 the NTSB determines that the probable cause was excessive wear of the left flap motor flexible drive spline and certification of the airplane that did not meet the requirements of Civil Air Regulation 3.339. The worn spline caused a split flap condition of 34 degrees that resulted in marginal flight control authority. Moderate low altitude turbulence and transient low level wind shear may have contributed to the upset and loss of control.
CAR 3.339 states that a mechanical interconnection is required unless the airplane is demonstrated to have safe flight characteristics while the flaps are retracted on one side and extended on the other.

Figure 6.40 shows a schematic of the PA-31 flap drive system. Two poor design features are evident:

- Flap symmetry relies on the integrity of the two flexible drives and their splines.
- Flap position indication is misleading to the pilot with any of the spline drives failed.

Flight test results to demonstrate that the airplane meets the intent of CAR 3.339 were not available and therefore the FAA required that Piper carry out such tests. This was done and the tests (performed in 1982) showed (Ref. 6.27, page 14) that "an asymmetrical flap condition of 30 degrees was not controllable at 9,000 ft and an airspeed of 130 KIAS. However, the tests did not include the use of asymmetric power or the use of rudder to control the airplane laterally. Therefore, the Board believes that the flight crew of the accident airplane was able to maintain lateral control with an asymmetric flap condition of 34 degrees because they used rudder and asymmetric power as specified in the airplane emergency procedure, that is, full or near full power on the left engine and reduced power on the right engine. Consequently, although they were able to maintain lateral control, there was not sufficient power to either maintain altitude or to provide directional control."

6.29.3 Solution

Quoting from Ref. 6.27, page 15: "The flight tests conducted by the manufacturer in response to the FAA's inquiry of February 1, 1982, about the compliance with CAR 3.339 clearly demonstrated that the flap system in certain models of the PA-31 airplane, including the accident airplane, did not meet the requirements of the regulation. The tests established that with less than full extension on one flap (30 degrees) and the other flap in the retracted position, the airplane was not controllable laterally. According to the regulation, under these conditions, the flap should have had a mechanical interconnection to synchronize movement of the flaps. However, the flap system did not include such a mechanical interconnection. Consequently, PA-31 airplanes equipped with this flap system were not properly certificated. After the manufacturer's flight tests of the flap system, appropriate measures were taken by the manufacturer and the FAA to remedy the improper certification."

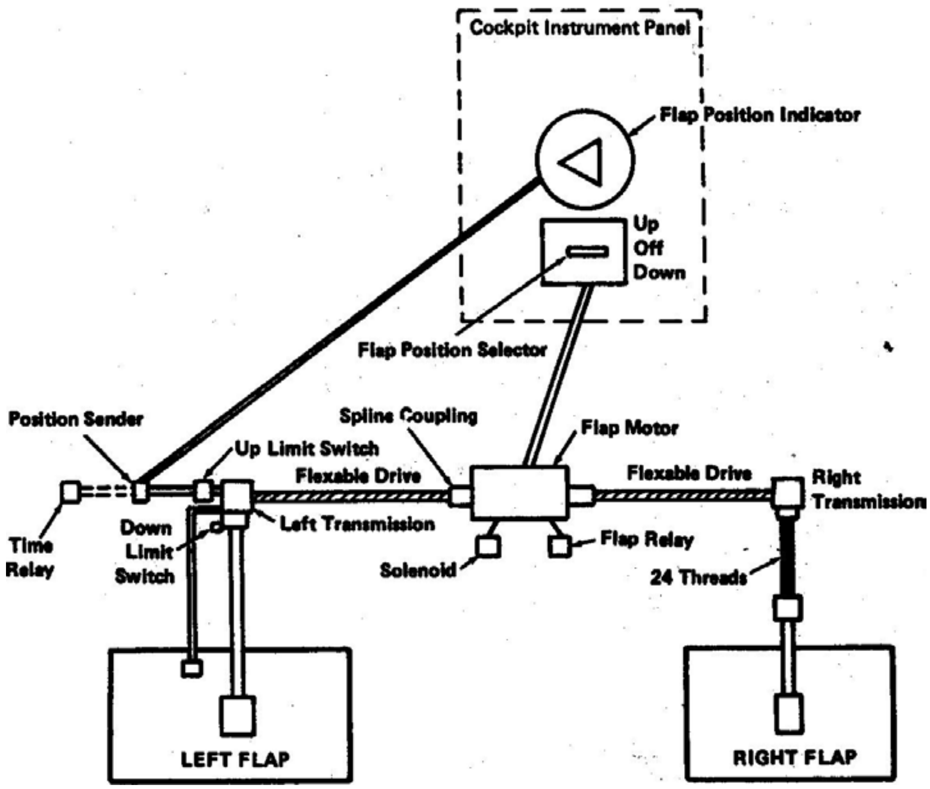


Figure 6.40 Schematic of the PA-31 Flap Drive System (Courtesy NTSB)

6.29.4 Lesson

This is another example of the DER system breaking down. Quite apart of this fact is the following question: does a design engineer really need a regulation to tell that a system such as sketched in Figure 6.40 should not be certified?

6.30 Design of Windshield Washer System

6.30.1 Problem

In February of 1982 a DeHavilland DHC-6-100 (Figure 6.41) of Pilgrim Airlines made an emergency landing after a fire erupted in the cockpit while the aircraft was flying in IMC from Groton, CT to Boston, MA.

Figure 6.41 DeHavilland DHC-6-100 (Not accident aircraft, Courtesy Pierre Langlois)

The captain and first officer were seriously injured, one passenger was killed, eight passengers had serious injuries and one passenger sustained minor injuries.

6.30.2 Cause

In Ref. 6.28 the NTSB determines that the probable cause of the accident was the deficient design of the isopropyl alcohol windshield washer/de-icer system and the inadequate maintenance of the system which resulted in an in-flight fire. The ignition source of the fire was not determined.

Figure 6.42 which shows how the system is located in the airplane. The following is edited material from Ref. 6.28.

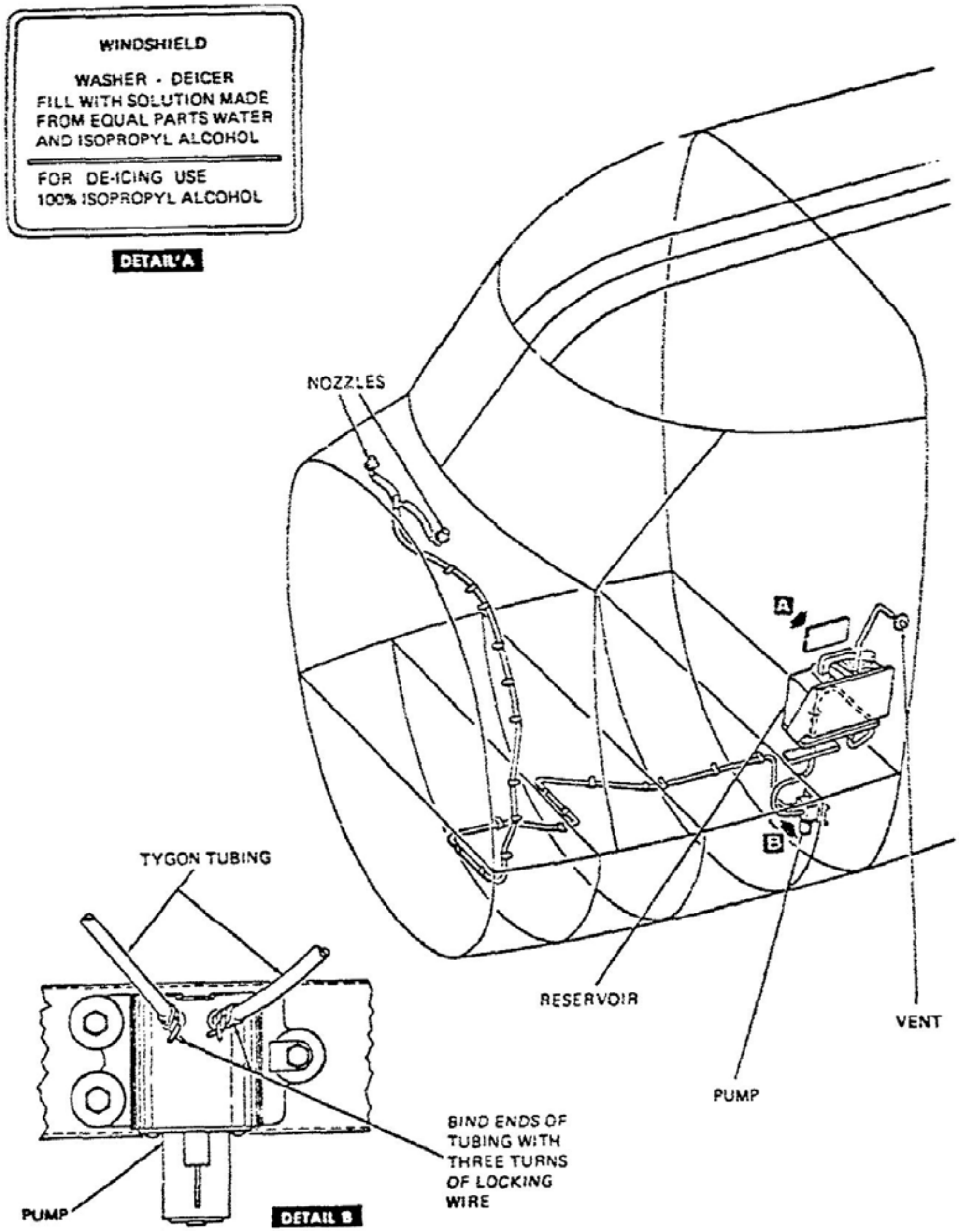


Figure 6.42 Windshield Washer/De-icer System Layout in the DHC-6-100 (Courtesy NTSB)

The 1.5 gallon reservoir containing isopropyl alcohol is located to the left side of the captain's seat-pan. The reservoir is held in place by a tray mounted to the cockpit structure just aft of the captain's entry door. This entry door can be seen in Figure 6.41.

The reservoir is constructed of polyethylene material and is vented outboard through the cockpit structure. The plumbing is Tygon clear plastic tubing (vinyl base). The reservoir supply line is routed from the bottom of the reservoir through the cockpit floor to a 28-volt D.C., electrically driven, low-pressure pump forward, beneath the cockpit floor, and upward to the windshield spray nozzles.

A Pilgrim Airlines pilot reported that the following incident occurred to the accident aircraft on the afternoon of February 18, 1982. During a stop-over the captain saw a clear liquid leaking from the exterior hydraulic access panel which is located below the captain's entry door. After opening the panel, he found that the tubing from the outlet fitting (pressure side) of the pump had come off and alcohol was leaking from the fitting. The first officer tried to re-attach the tubing but it was too short. The flight crew then removed the reservoir from the aircraft, wrote up the discrepancy, and since the weather was fair, continued on their scheduled route. The same discrepancy (tubing separated from the pump outlet fitting) was also reported several months earlier while the aircraft was on the ground at LaGuardia and again it was reported that the tubing was too short.

NTSB investigators determined that the Tygon tubing hardens when it comes in contact with alcohol and becomes mis-shaped at its connection points, often resulting in leaks. To correct the leaks, the hardened, mis-shaped ends of the tubing were cut off and the tubing re-attached. After successive cuts, a splice is required to make the tubing long enough for an unstrained connection. Maintenance procedures approved by DeHavilland allows the connection to be secured with three wraps of safety (locking) wire.

The mechanic who re-installed the reservoir on the evening of February 18, 1982, was interviewed. He stated that the re-installation was carried out in accordance with DeHavilland maintenance procedures.

Tests conducted by the NTSB showed that with the tubing not properly fastened on the outlet side of the pump, iso-propyl alcohol could have leaked through the pump and accumulated in the compartment beneath the cockpit floor without the washer/de-icer system having been activated. Once the system was activated, the tests showed that alcohol would have been sprayed from the pump outlet forward into the compartment area which contained numerous ignition sources.

It is of interest to note the presence of the following potential ignition sources in the compartment below the cockpit floor in this airplane:

- Engine bleed-air ducting with bleed-air temperatures as high as 150 degrees F.
- Ejector, mixing box, and silencer
- Hydraulic power package powered by an electric motor pump
- Windshield washer/de-icer electric motor driven pump

Although it could not be established from the wreckage which of these electrical components was the actual ignition source, the system design clearly enables this.

6.30.3 Solution

The NTSB made a number of recommendations as a result of this crash. Pertinent to the root cause of the crash was a recommendation to the FAA to:

"Issue an Airworthiness Directive to required a redesign and modification of iso-propyl alcohol windshield de-icing systems installed on DHC-6 aircraft to eliminate the potential for alcohol leakage or, if practicable, to require replacement of these systems with the electrically heated windshields offered by the manufacturer as an alternative installation."

6.30.4 Lessons

1. Remember a fundamental law affecting system layout design in airplanes: ANY SYSTEM IN AN AIRPLANE WHICH CARRIES A LIQUID WILL LEAK. It is the responsibility of the designer to trace possible sources of leaks and predict where the liquid will go. From that the consequences of a leak are easy to predict. Redesign should be undertaken if those consequences are not benign.

2. Does a system designer need a regulation to prohibit ignition sources in areas where flammable liquids may leak?

3. There were several precursors to this accident that should have alerted flight crews, mechanics and other maintenance personnel about this potential hazard. Precursors should be taken seriously.

6.31 Three Hydraulics System Lines in the Leading Edge

6.31.1 Problem

In September of 1987 a B1-B bomber (Figure 6.43) while on a low altitude training flight hit a large bird and lost control. The airplane crashed and burned. Only three of the six crewmembers on board managed to successfully bail out.

Figure 6.43 B1-B bomber (Not accident aircraft, Courtesy S. Petch)

6.31.2 Cause

An investigation showed the cause of the crash to be a pelican, which was hit by the inboard leading edge of the left wing. Hydraulic system lines serving the primary flight control system were located behind this leading edge. The collision with the pelican so crushed the wing leading edge that all three hydraulic lines were severed. This rendered the airplane uncontrollable which caused the crash.

6.31.3 Solution

Design engineers should always assume the worst. An airplane intended for high-speed flight at low altitude is going to hit large birds. Therefore, primary flight control system hydraulic lines should never be located together behind a vulnerable leading edge.

6.31.4 Lesson

It is important for design engineers to remember that no single event should put the primary flight controls of any airplane at risk. A redundant system looses all its redundancy if this is forgotten.

6.32 Leaks into the Avionics Bay I

6.32.1 Problem

In September of 1987 a Swissair Fokker 100 (Figure 6.44) was on final approach to the Geneva Airport in Switzerland, in very poor visibility conditions.

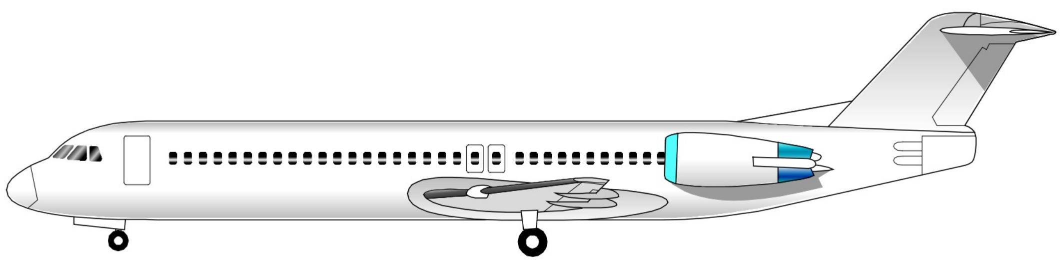


Figure 6.44 Fokker 100

During the approach, without warning, all flat panel displays in the cockpit went dark. The pilots had a flashlight and three mechanical flight instruments that kept on working. With that help they were able to make a safe landing.

6.32.2 Cause

In the Fokker 100, a potable waterline to the front toilet had been installed close to the upper fuselage skin. Earlier in the flight this line had frozen and, of course, cracked. During the slow descent, the ice thawed and water poured into the main avionics bay, which is located in the belly below that toilet (Figure 6.45).

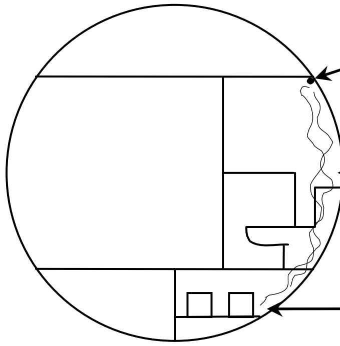


Figure 6.45 Sketch of the Waterline Location, the Lavatory and the Main Avionics Bay

The result was a complete shorting of electrical power to the flat panel displays and all cockpit displays went dark.

6.32.3 Solution

This is an entirely predictable scenario. It should have been caught during an early design-safety review, but was not. Either the main avionics bay must be relocated or the waterline must be relocated. Preferably both.

6.32.4 Lesson

Designers should really remember the following law:

Every system in an airplane which carries a liquid WILL LEAK.

It is the responsibility of the designer to keep this in mind and ask the “what if” question, trace the likely path of the liquid, and take appropriate design action.

6.33 Nacelle Cowl Design and Fuel Filter Cover Design

6.33.1 Problem

In April of 1988 a Horizon Air DeHavilland DHC-8, N819PH, (Figure 6.46) experienced a power loss in engine No.2 shortly after take-off.

Figure 6.46 DeHavilland DHC-8 (Not accident aircraft, Courtesy www.al-airliners.be.)

The captain decided to return to Seattle for a precautionary landing. After lowering the landing gear a massive fire broke out in the No.2 nacelle. After touchdown the crew realized that all directional control and braking capability had been lost. The airplane ran into several objects on the ground and was destroyed by fire. During the emergency evacuation 4 passengers were seriously injured, 24 passengers, the flight attendant and both pilots received minor injuries and 9 passengers received no injuries.

6.33.2 Cause

In Ref. 6.29 the NTSB determines as the probable cause:

"the improper installation of the high-pressure fuel filter cover that allowed a massive fuel leak and subsequent fire to occur in the right engine nacelle. The improper installation probably occurred at the engine manufacturer; however, the failure of airline maintenance personnel to detect and correct the improper installation contributed to the accident. Also contributing to the accident was the loss of the right engine center access panels from a fuel explosion that negated the fire suppression system and allowed hydraulic line burn-through that in turn caused a total loss of airplane control on the ground."

To help in understanding the cause of this accident references will be made to Figure 6.47 for the general nacelle layout and to Figure 6.48 for the proper and improper fitting of the high-pressure fuel filter cover.

Figure 6.47 General Arrangement of the DHC-8 Nacelle and Center Covers (Courtesy NTSB)

Quoting from Ref. 6.29, page 6: "Both engines on the airplane were equipped with a hydro-mechanical metering unit (HMU). An HMU assembly consists of the hydro-mechanical fuel control, a high-pressure fuel pump with an integral fuel filter housing that contains the high-pressure fuel filter. The HMU assembly was replaced on the right engine of N819PH on April 8 and 9, 1988. The replacement HMU assembly was removed as a complete unit from a spare serviceable engine in Horizon stores that had been received from the Pratt and Whitney Canada factory."

Despite the fact that several persons were involved in the process of removing and installing the HMU, no one noticed that the filter cover was not seated properly. Figure 6.48 shows that there was a 0.116 inch gap.

Despite this gap the airplane had completed several flights without problems before the accident flight. Quoting again from Ref. 6.29 (p.25-26):

"The Board believes that repeated high pressure fuel pressurizations of the unsecured fuel filter cover allowed the neoprene o-ring (Figure 6.48) to distort and extrude into a position so that it allowed high pressure fuel to be channeled to a vent and drain hole on the filter housing and thereafter overboard into the nacelle. The distorted o-ring and its position in relation to the vent and drain hole appeared on radiographs before the filter cover was removed. The manufacturer stated that the purpose of the vent and drain holes in the filter housing was to prevent the possible spill of less than 1 pint of fuel during periodic filter changes and that it was mainly a minor environmental safeguard.

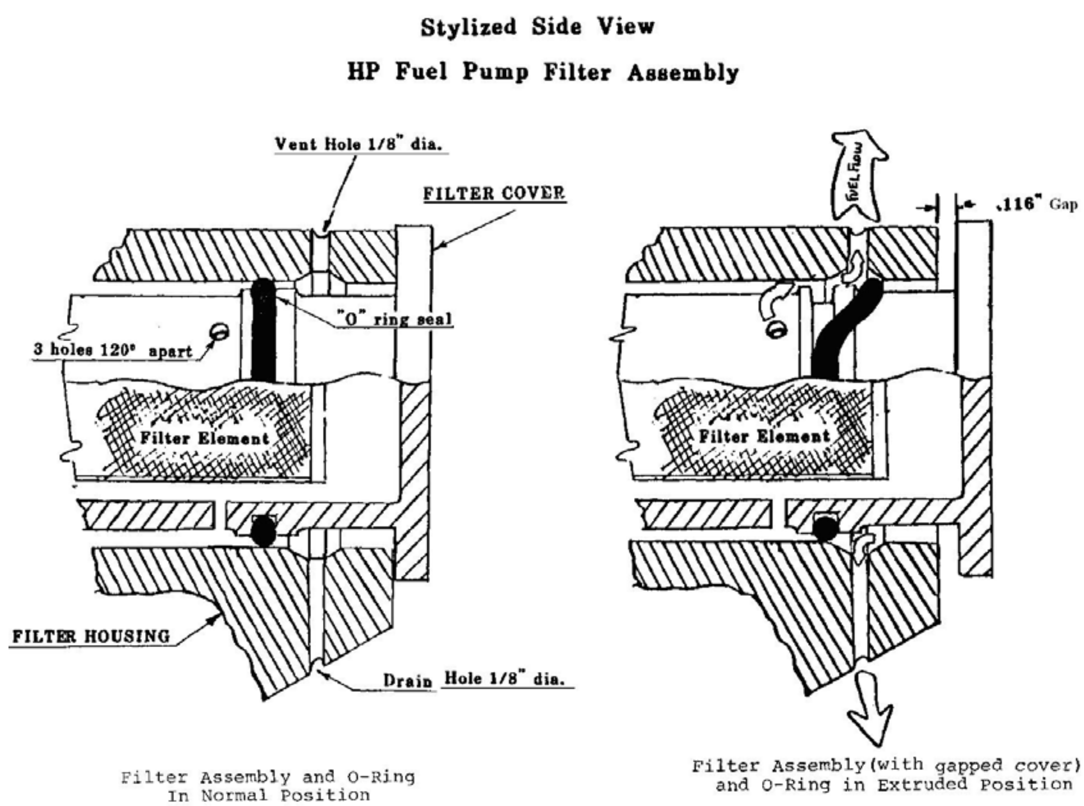


Figure 6.48 High Pressure Fuel Filter Cover. Left is Proper, Right is Improper (Courtesy NTSB)

The Safety Board believes that the fuel leak that was the source of the in-flight fire began shortly after take-off. Fuel began to collect in the engine nacelle, and shortly thereafter, the fuel also flowed rearward to collect in the right wheel well. Fuel also leaked overboard and was observed by a passenger seated on the right side of the airplane. The passenger, following the observation of the fuel leak, could not have been expected to raise alarm because he was unfamiliar with airplanes.

Before the outbreak of the fire, the Safety Board believes that the fuel/air mixture within the nacelle and wheel well was too rich to ignite. As the landing gear doors opened on final approach, this fuel/air mixture was leaned by ambient air, became combustible, and ignited rapidly. The exact source of ignition could not be determined positively. A mis-placed starter generator brush access cover on the right generator conceivably could have been a factor in the ignition because it may have allowed a combustible fuel/air mixture to accumulate in the area of the generator brushes.

There is also another, unshielded path to the brush/armature area. Near the top of the starter/generator, electrical leads progress into the generator armature and brush area. There is an open gap at this location which is about 1 foot closer to the fuel leak than the brush access cover. Therefore, in spite of the mis-positioning of the access cover, there was another open path to an ignition source.

Another possible ignition source could have been the engine exhaust pipe. Atomized fuel could have been drawn into the cooling air shroud surrounding the exhaust pipe. The area where this cooling air originated contained a large amount of accumulated fuel."

The NTSB also observed that a fire and subsequent shut-down of one engine in a twin engine transport should not have caused the deterioration and loss of control of the airplane even though it was then on the ground. In accordance with accepted systems design practice, this should not have happened.

To understand how the loss of control occurred it is necessary to describe some aspects of the DHC-8 systems design. Quoting from Ref. 6.13, pages 27-28:

Following the outbreak of fire, the pilots immediately shut down the right engine in accordance with their emergency training. During a simple right engine shut-down (with no other problems), the following components, which could only receive hydraulic pressure from the right engine-driven hydraulic pump or the No.2 electrical-standby pump, would be disabled:

- The inboard and outboard ground spoilers. These wing-mounted automatically activated panels normally activate on touchdown and aid in airplane control by destroying lift on the wings and by acting as airbrakes.
- The outboard roll spoilers. Also mounted on the wings, these spoilers enhance the roll rate while airborne and automatically activate and act as the ground spoilers when the airplane is on the ground.
- The emergency/parking brakes. This wheel brake system, hydraulically separate from the pilot's main wheel brakes, mechanically slows the airplane down via a hand lever in the cockpit. The pilot attempted to use this system to no avail.
- Nose-wheel steering. This system casters the nose-wheel via the captain's hand control or by either the captain or first officer rudder input. Both the captain and the first officer attempted to use the nose-wheel steering system to no avail.
- The upper rudder actuator. This hydraulic actuator along with the lower rudder actuator powers the rudder, which yaws the airplane and provides directional control at moderate to high speeds during the landing roll-out. The system consists of two actuators, one on each hydraulic system. Both crew members attempted to steer the airplane with the rudder, but to no avail.
- The landing gear extension and retraction system. The nomenclature is self-explanatory.

The No.2 electrical-standby hydraulic pump (located in the right engine nacelle) automatically should have provided hydraulic pressure to these systems when the right engine-driven hydraulic pump was de-activated. This did not occur, however, because the electrical wiring and control unit that furnishes power to the pump was destroyed by the fire. The No.2 electrical-standby hydraulic pump circuit breaker, in fact, was tripped because of short circuiting in the control unit due to the fire.

The Safety Board believes the following components of the left hydraulic system were disabled because the in-flight fire breached a No.1 (left) lift dump hydraulic pressure line, a No.1 hydraulic pressure return line, and a No.1 system hydraulic line servicing the right wing inboard roll spoiler system, all located in the right wheel well:

- The wing flaps. Trailing edge flaps that would have shortened the landing roll to some degree in their fully extended position. The pilots attempted to position the flaps to the 15" landing position, but the flaps stopped at about 6" down as the left hydraulic pressure was lost.

- The main-wheel brakes. These brakes are the primary ground braking devices on the airplane. Both pilots depressed their brake pedals to no avail. In fact, the first officer's pedals are linked mechanically to the pilot's pedals, so the failure of the left hydraulic system disabled both sets of brake pedals.
- The inboard roll spoilers. These spoilers function like the outboard roll spoilers. (See item number 2 under the right hydraulic system discussion.)
- The hydraulic motor half of the PTU. This device is a hydraulically powered motor designed to power automatically an auxiliary right system hydraulic pump to assist only in landing gear retraction in the event of a right engine failure. There was no indication that this device was operating at any time during the flight, nor would it have aided the crew under the circumstances of this accident.
- The lower rudder actuator. This unit is the identical counterpart to the upper rudder actuator, but powered from the left hydraulic system.
- The anti-skid control valves. There are two hydraulic valves that regulate hydraulic fluid flow to the wheel brakes. These valves operate an anti-skid control unit. Since the main-wheel brakes were inoperative during the accident sequence, the failure of these valves did not affect the outcome of events.

6.33.3 Solution

The NTSB made several recommendations in areas not discussed in this section but pertinent to the accident events. From a design viewpoint three items appear of most importance:

- The nacelle covers blew out during the fuel explosion. From the wreckage it was clear that the panel was buckled such that the fasteners popped loose. The fact that the panel blew off made it impossible for the fire suppression system to operate.
- The design and maintenance procedures involving the fuel-filter cover needed to be addressed and were.
- The design of the hydraulic, brake, rudder and spoiler system appears to be not in accordance with the philosophy that one failure should not put all out of commission. The NTSB choose not to address this in its recommendations.

6.33.4 Lessons

1. The design of the fuel filter cover should have been such that installation such as occurred would have been impossible.

2. The spoiler, brake and rudder system design was clearly deficient in that one cause apparently rendered these systems inoperative.

6.34 Ground Spoilers Deploy in Flight

6.34.1 Problem

In January of 1996 a Valujet Airlines DC-9-32 (Figure 6.49) on a flight from Atlanta, GA to Nashville, KY, made a hard landing at Nashville following unintentional ground spoiler deployment while on final approach.

Figure 6.49 Douglas DC-9-32 (Not accident aircraft, Courtesy Frank C. Duarte Jr.)

The airplane touched down hard which damaged the nose-gear (both wheels broke off) and a go-around was initiated. The airplane then returned for a second touchdown after which it was brought to a full stop. Of the 93 persons on board, one flight attendant and four passengers suffered minor injuries during the evacuation. The airplane was substantially damaged.

6.34.2 Cause

To understand what happened, some aspects of the DC-9 systems design need to be explained. The following material was taken from Ref. 6.30.

The DC-9 tricycle landing gear is controlled by a lever on the left side of the first officer's instrument panel and is hydraulically actuated by pressure from the right (No.2) hydraulic system. The main and nose landing gear consist of dual wheels mounted on shock struts, with dual brakes mounted on the main landing gear.

There is a ground shift mechanism, which is actuated by nose-gear shock strut extension/compression. This mechanism controls whether certain aircraft systems operate in the ground mode or in the flight mode. When the nose-gear shock strut is compressed by the weight of the aircraft, the ground shift mechanism causes those aircraft systems to be operated in the ground mode. When the nose-gear shock strut is extended after take-off (this happens under the influence of gravity which pulls the nose-gear wheels downward), it triggers the ground shift mechanism, electronically shifting the aircraft systems to the flight mode. Figure 6.50 shows the location of the ground shift mechanism between the shock strut and the nose-wheel bogey.

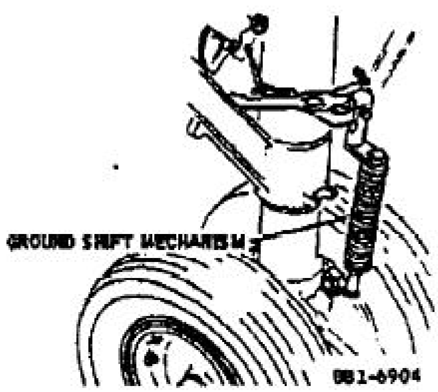


Figure 6.50 Location of Ground Shift Mechanism (Courtesy NTSB)

Figure 6.51 shows a schematic indicating the ground shift mechanism functions and the circuit breakers.

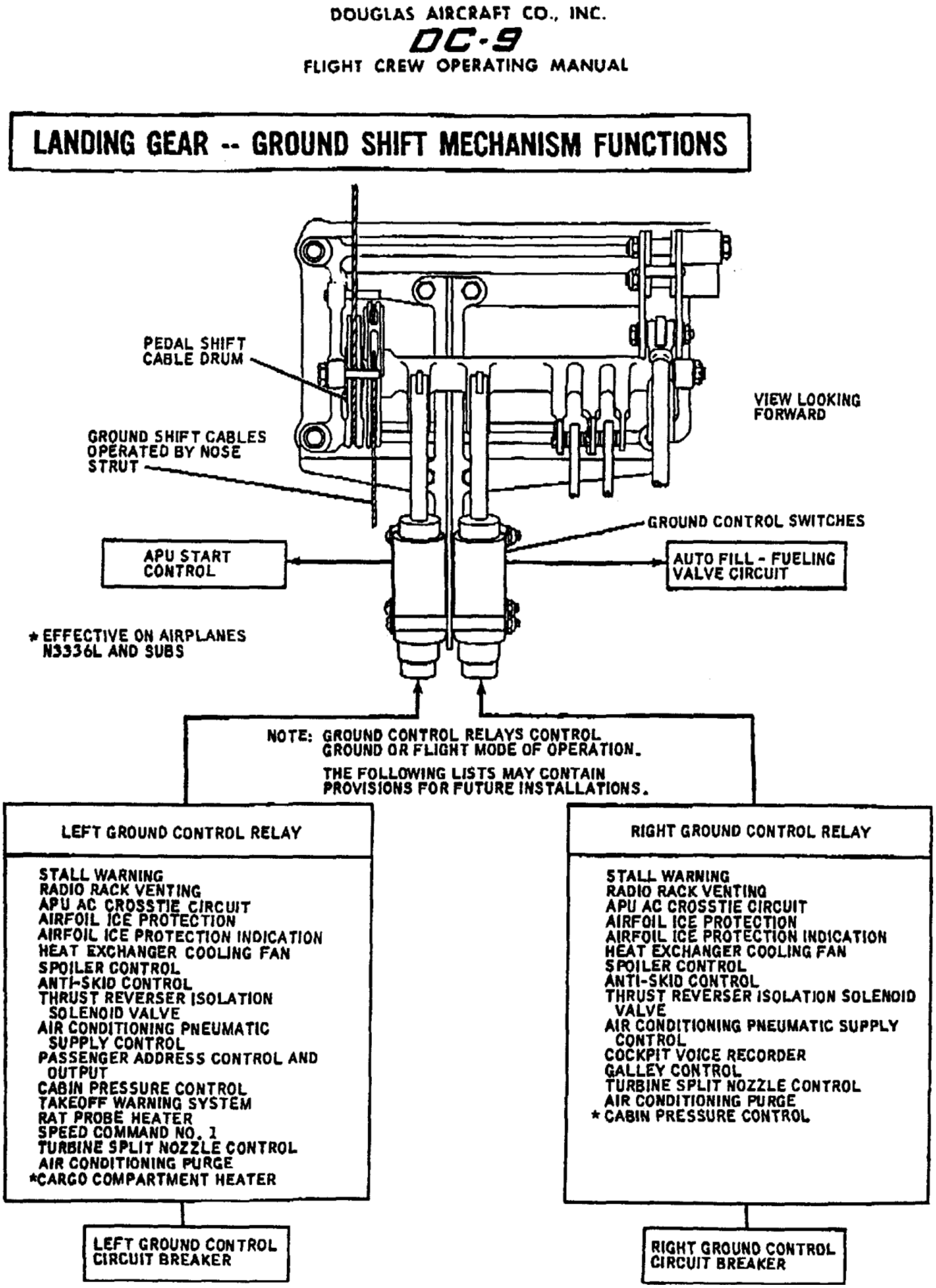


Figure 6.51 Schematic of Ground Shift Mechanism (Courtesy NTSB)

It turns out that when the nose-gear strut is under-serviced/under-inflated, strut extension after lift-off may not be sufficient to activate the ground shift mechanism. This is a fairly common problem during cold weather operations and Douglas had issued numerous Service Bulletins (SBs) and All Operator Letters (AOLs) describing the anomaly and recommending maintenance procedures to avoid under-serviced/under-inflated nose-gear shock struts during cold weather operations.

Note in Figure 6.51 the words “spoiler control” in both boxes.

The DC-9-32 has four spoiler panels located on the upper surfaces of the wings, forward of the trailing edge flaps. During airborne operations, the spoiler panels work with the ailerons automatically, through an aileron/flight-spoiler mixer assembly, to help lower the up-aileron wing. Additionally, when the speed-brake/ground-spoiler control lever is pulled aft during flight, the four spoiler panels extend to function as speed brakes. Maximum spoiler deployment in flight is approximately 30 degrees.

During ground operation, the four spoiler panels can be extended to 60 degrees to perform the ground spoiler function. Ground spoiler actuation can be accomplished automatically or manually. Automatic ground spoiler extension requires main wheel spin-up or the ground shift mechanism to be in the ground mode. According to Douglas publications, the flight-crew’s action of arming the spoilers for landing, per the normal “Before Landing” checklist, was an acceptable technique, PROVIDED that the ground control relay circuit breakers were not reset until AFTER landing.

Now back to what happened during the flight.

During the departure from Atlanta the pilots experienced difficulty in raising the landing gear and had to manually bypass the landing gear anti-retraction system before they could successfully retract the landing gear. As they continued to climb, the pilots realized that although the airplane was airborne, the cabin pressurization and take-off warning systems were still operating in the ground mode. In accordance with the guidance contained in the QRH, the pilots pulled the ground control relay circuit breakers and observed that the airplane’s pressurization and take-off warning systems began to operate in the flight mode. Because of the irregularities encountered by the pilots, and because post-accident examination and testing of the nose-gear and its systems revealed no evidence of pre-impact mechanical anomaly, the Safety Board concludes that the nose-gear shock strut extension during the initial climb-out was insufficient to actuate the ground shift mechanism, release the landing gear lever anti-retraction mechanism, and shift the airplane systems to the flight mode.

It is likely that the nose-gear shock strut did not extend far enough to actuate the ground shift mechanism because it was under-serviced/under-inflated for the cold/winter weather conditions. The en route portion of the flight proceeded uneventfully.. When the airplane was about 100 feet above the ground during the approach to Nashville, the pilots reset the ground control relay circuit breakers, thereby unintentionally shifting the airplane systems from the flight mode to the ground mode. The ground spoilers subsequently extended in flight, and the airplane descended suddenly, impacting the ground in the runway approach light area.

In Ref. 6.30 the NTSB determines that the probable cause of the accident was “the flight-crew’s improper procedures and actions (failing to contact system operations/dispatch, failing to use all available aircraft and company manuals, and prematurely resetting the ground control relay circuit breakers) in response to an in-flight abnormality, which resulted in the inadvertent in-flight activation of the ground spoilers during the final approach to landing and the airplane’s subsequent increased descent rate and excessively hard ground impact in the runway approach light area.

Contributing factors in the accident were Valujet’s failure to incorporate cold weather nose-gear servicing procedures in its operations and maintenance manuals, the incomplete procedural guidance contained in the Valujet quick reference handbook, and the flight-crew’s inadequate knowledge and understanding of the aircraft systems.”

6.34.3 Solution

The NTSB mentioned a number of safety issues in its recommendations to the FAA and these were addressed. However, none of these recommendations dealt with what the author views as a fundamental systems design issue. For the system to operate properly requires:

- detailed knowledge of the operating characteristics of the system by the flight crew.
- inspection and maintenance procedures having been performed properly, particularly during winter operations.

It is probably too much to ask of the flight-crew/inspection/maintenance personnel to do everything right. It is the author's view that the system should not have been certificated with these features.

6.34.4 Lesson

This is a fairly complicated system. A single failure in flight crew knowledge or action and/or proper actions of the inspection/maintenance system should not result in a potentially hazardous flight situation.

6.35 Hydraulic System Design Problem

6.35.1 Problem

In February of 1996 a Continental Airlines Douglas DC-9-32 (Figure 6.52) landed with the gear up at the Houston Intercontinental Airport.

Figure 6.52 Model of Douglas DC-9-32 (Courtesy geminijets.com)

The airplane slid about 7,000 ft on the runway before coming to rest in the grass. The cabin began to fill with smoke and an evacuation of the airplane was ordered. Of the 82 passengers and five crew members on board, twelve passengers reported minor injuries.

6.35.2 Cause

In Ref. 6.31 the NTSB determines that the probable cause of this accident was the captain's decision to continue the approach contrary to Continental Airlines (COA) standard operating procedures that mandate a go-around when an approach is un-stabilized below 500 ft or a ground proximity warning system alert continues below 200 feet above field elevation. The following factors contributed to the accident:

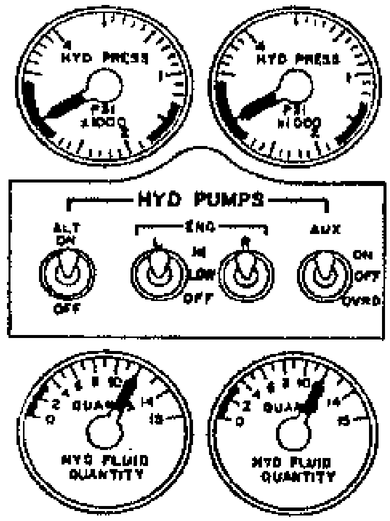
- the flight crew's failure to properly complete the in-range checklist, which resulted in a lack of hydraulic pressure to lower the landing gear and deploy the flaps;
- the flight crew's failure to perform the landing checklist and confirm that the landing gear was extended;
- the inadequate remedial actions by COA to ensure adherence to standard operating procedures; and
- the FAA's inadequate oversight of COA to ensure adherence to standard operating procedures.

To understand how this accident happened a description of the hydraulic system is useful. The following is taken from Ref. 6.31.

6.35.3 Hydraulic System

Hydraulic power on the DC-9 is provided by two independent hydraulic systems. Each system is normally pressurized by its respective engine-driven hydraulic pump. An auxiliary (AUX) electrically operated pump and an alternate (ALT) motor pump provide backup pressure sources. The output pressure of each engine-driven hydraulic pump is controlled by a 3-position switch, which is located on the first officer's instrument panel, but is accessible to both pilots.

Figure 6.53 shows a sketch of the hydraulic switch panel.



Hydraulic switch panel in high pressure configuration.

Figure 6.53 Hydraulic Switch Panel on the Douglas DC-9
(Courtesy NTSB)

With the engine-driven pump switches in the “HI” position, pump output pressure is 3,000 psi. The “LOW” position reduces the pressure to 1,500 psi. The “OFF” position depressurizes the system. Ground, take-off and landing operations are conducted with the engine-driven pump switches in the “HI” position and the AUX and ALT switches “ON”. During in-flight operations, system pressures are reduced to 1,500 psi by positioning the engine-driven pump switches to “LOW” and turning the AUX and ALT switches to “OFF”. Continental Airlines procedures require changeover to the low pressure configuration during completion of the in-range checklist.

Hydraulic components are classified as being either priority or non-priority based on their operating pressure requirements and/or their function. Priority components are mainly associated with normal flight operations and require lower pressures to function. These components include spoilers, slats, rudder, flap/rudder stop, engine reversers, the elevator augmentation system, and the ventral stair system. Non-priority components require a system pressure of at least 2,000 psi to function normally and are required for all ground operations, including take-off and landing. Non-priority components include landing gear, brakes, flaps, nose-wheel steering, and the alternate gear pump. A priority valve in each system gates hydraulic pressure between the priority and non-priority components. When the engine-driven pumps are placed in “HI” mode, the priority valves open as the system pressure exceeds 2,000 psi and permit operation of non-priority components. Placing the engine-driven pumps in the

"LOW" mode reduces system pressure, closes the priority valves, AND RENDERS THE NON-PRIORITY COMPONENTS, INCLUDING THE FLAPS AND LANDING GEAR, INOPERATIVE.

According to Douglas, as of December 31, 1996, 874 DC-9 (Models -10 through -50) and 1,009 MD-80 series airplanes were in service worldwide with the "HI, LOW, OFF" hydraulic switch configuration.

A landing gear warning horn will sound (and did sound) when the throttles are retarded to idle if the landing gear is not down and locked. The pilots can silence the horn by depressing the horn cutoff button located on the instrument panel. The landing gear horn will also sound, regardless of throttle position, if the landing gear is not down and locked and the flap handle is moved beyond the approach (15 degrees) setting. In this condition the horn cannot be disabled and will continue to sound until the gear is down and locked or the flap handle is retracted to a setting of 15 degrees or less.

Examination of the cockpit revealed that the landing gear handle was in the down position and the flap handle was set to 50 degrees. The left and right engine driven hydraulic pump switches were in the "LOW" position, and the ALT and AUX hydraulic pump switches were in the "OFF" position. The left hydraulic system gauge indicated 1,600 psi, and the right gauge indicated 0 psi (Refer to Figure 6.53). The safety wire on the GPWS flap override switch was found broken; however, the switch was not in the "OVRD" position.

The first officer, self-described as being "new to the aircraft", reported being unaware that high pressure in the hydraulic system was necessary to get the gear down……The captain "commented that he had made this mistake before".

Reports by other DC-9 pilots indicate that failure to configure the hydraulic system for landing is not an uncommon occurrence. A review of the checklists from several DC-9 and MD-80 operators revealed that none of the checklists, including the Douglas Aircraft Company's checklist, emphasize the importance of the "Hydraulics" item by placing it first on the in-range checklist or requiring mandatory cross-check of the item by both pilots.

In addition, the Safety Board's review of the information provided by COA to its pilots concerning the DC-9 hydraulic system revealed that the flight manual and training materials do not explicitly state that if the pumps are not switched to "HI", the landing gear will not extend and the flaps will not deploy.

6.35.4 Solution

The NTSB made a large number of recommendations to the FAA aimed at improving procedures and training. The NTSB did not question the design of this system.

6.35.5 Lesson

This is another example of a system design that invites mismanagement. The author believes that professional pilots should be expected to follow procedures and that airline management should not let improperly trained pilots fly their airplanes. Having said that, aircraft system designers could have predicted this type of occurrence and should have designed the system so that this cannot happen.

6.36 Cabin Door Design III

6.36.1 Problem

In November of 1996 A United Express Beechcraft 1900C (Figure 6.54) collided with a Beechcraft King Air A90 (Figure 6.55) at the Quincy Municipal Airport in Illinois.

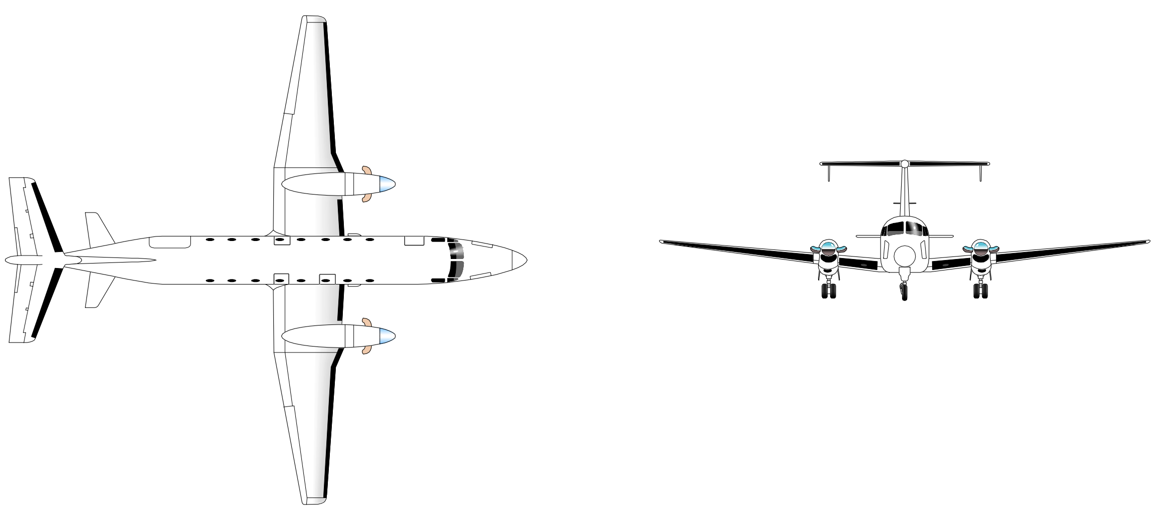


Figure 6.54 Beechcraft 1900C

Figure 6.55 Beechcraft King Air A90 (Not accident aircraft, Courtesy Frank C. Duarte)

The collision occurred while the 1900C was completing its landing roll on runway 13 while the King Air was on its take-off roll on runway 04 at the point of intersection between those two runways. All 10 passengers and two crew members on board the 1900C and the two occupants aboard the King Air A90 were killed.

6.36.2 Cause

In Ref. 6.32 the NTSB finds that the probable cause of the accident was the failure of the pilots in the King Air A90 to effectively monitor the common traffic advisory frequency or to properly scan for traffic, resulting in their commencing a take-off roll when the Beech 1900C was landing on an intersecting runway.

Contributing to the severity of the accident and the loss of life were the lack of adequate aircraft rescue and fire fighting services, and the failure of the air stair door on the Beech Model 1900C to be opened by the surviving persons on board. The reason why the door could not be opened could not be determined.

In Ref. 6.32 the Safety Board first implied that the door frame design was such that in this relatively mild crash (both airframes came to rest on their own landing gears) the doorframe had deformed to preclude the door from being opened. Later, in a revision to Ref. 6.32, the wording was changed because there was no compelling evidence for this.

Two pilots at the airport were first on the scene and attempted to open the door but could not. They did observe severe smoke in the passenger cabin and a fire on the right side of the airplane. Their attempts to rescue those inside were in vain.

6.36.3 Solution

There is a problem with the type of door design used on this airplane. Figure 6.56 shows a view of this door.

Note that the operation of the door latching mechanism relies on cable tension in the actuating cable. It is known that a slightly slack cable will not allow the removal of all locking pins. A preferred alternate design is to use a system of pushrods.

6.36.4 Lesson

In passenger aircraft there should be doors and or emergency exits which are easy to open from the inside as well as from the outside. Anything in the door latching mechanism that makes opening difficult should be avoided.

6.37 Landing Gear Actuator Corrosion

6.37.1 Problem

The following incident is described in Ref. 6.33. In March of 1999 an Ansett Australian Airlines Boeing 737-377 (Figure 6.57) experienced a landing gear anomaly while on final approach to Melbourne, Australia.

Figure 6.57 Boeing 737-377 (Not accident aircraft, Courtesy Raymond Rowe)

When the landing gear lever was placed in the “down” position during final approach a loud thump was heard and the “gear safe” green light for the right main gear illuminated immediately. This was followed by the illumination of the left main and nose landing gear lights, consistent with a normal extension sequence. The airplane rolled about 4 degrees to the right while the gear was extending before the left gear did. This was countered by a left roll control input. The landing was made without further incident.

The airplane was placed on jacks for a retraction test. When the landing gear lever was selected in the "up" position the right main landing gear moved about 6 inches before a grinding noise was heard. The test was suspended and the gear was extended.

When access panels were removed, it was found that the actuator beam arm inboard lugs and beam hanger had fractured. The rear wing spar, landing gear beam, aileron bus cable, pulley bracket, aileron and spoiler cables and hydraulic lines had been damaged extensively following the fracture of the lugs and hanger.

In August of 2003 a similar event occurred on an Aer Lingus 737-500 taking off from Amsterdam, The Netherlands. The airplane was landed safely in Dublin, Ireland. In this case the beam failure severed the spoiler cables, damaged the rear spar and landing gear beam, dislocated an aileron pulley bracket and pinched an aileron cable.

6.37.2 Cause

An inspection then revealed that the fracture of both lugs (made of high strength steel) was due to stress corrosion cracking.

Stress corrosion cracking of high strength steel components of aircraft landing gears occurs when the components are subjected to a sustained tensile stress and are exposed to an environment that allows stress corrosion cracking to begin. Components are susceptible to stress corrosion cracking when exposed to moisture and salt laden air as landing gears typically are. Stress corrosion cracking in the actuator beam arm lugs occurred as a result of the movement of the bushes installed in the lugs and the penetration of moisture into the gap created between the bushes and the lugs.

In the August 2003 event, stress corrosion cracking was again the cause.

6.37.3 Solution

Boeing had already developed a “fix” to this problem as explained in Service Bulletins 737-32A1224 and 737-32A1355 a part of which had been mandated by the FAA in an Airworthiness Directive 91-05-16. The AD required in situ inspection of the beam arm in 600 flight cycle intervals or a replacement with a new production arm.

The new arm assemblies incorporated the following changes:

- improved bushing with an increased interference fit
- an improved actuator beam bolt
- more extensive cadmium plating
- improved lubrication of components

The NTSB recommended that the FAA expedite the AD to prevent a more serious accident from happening.

6.37.4 Lessons

1. Stress corrosion cracking is a phenomenon that can be predicted to occur in certain environments and with certain combinations of materials and slack fits.

2. The proximity of the beam assembly to the rear wing spar, landing gear beam, aileron bus cable, pulley bracket, aileron and spoiler cables and hydraulic lines made this a potentially very dangerous situation which should be avoided in early layout design.

6.38 Leaks Into the Avionics Bay II

6.38.1 Problem

In June of 2000, a similar incident to that discussed in section 6.32 happened to a brand new Air Tran Boeing 717 (Figure 6.58).

Figure 6.58 Boeing 717 (Not accident aircraft, Courtesy www.geminijets.com)

6.38.2 Cause

The cause was leakage from the galley which penetrated the main avionics bay. A flight attended had a little accident and spilled a large amount of soft drinks on the floor. Sound familiar? The crew and passengers were lucky that the incident occurred on a bright day with excellent visibility. The airplane was safely landed.

6.38.3 Solution

Some years ago, when teaching a short course at AVRO in Manchester, England, they proudly showed me the design solution applied in the AVRO RJ series of transports. In that airplane, the entire galley is mounted over a shallow bathtub. This tub in turn has several positive drainage paths, leading away from the avionics bay. That is one way to avoid such problems.
The other way is NOT to locate the main avionics bay (or for that matter any electrical equipment) underneath a galley or a lavatory.

6.38.4 Lesson

See Section 6.32. This particular lesson does not seem to get learned easily by aircraft designers.

6.39 Fuel System Design II

6.39.1 Problem

In October of 2000, a Bombardier CL-604 Challenger (Figure 6.59) on a test flight, crashed immediately after lift-off from Wichita, Kansas.

Figure 6.59 Bombardier CL-604 Challenger (Not accident aircraft, Courtesy D. Pryde)

Two crewmembers died in the wreck, the third died a month later from severe burns.

6.39.2 Cause

According to Ref. 6.34 the NTSB established as the probable cause “the pilot’s excessive take-off rotation, during an aft center of gravity take-off, a rearward migration of fuel during acceleration and take-off and consequent shift of the c.g. aft of the aft allowable c.g. limit which caused the airplane to stall at an altitude too low for recovery.”
The objective of the accident airplane flight test was to test the effectiveness of a changed control force feel unit to allow for certification in the United Kingdom. The test had to be conducted at approximately aft c.g.

The flight test was planned for a low weight and the rear fuselage fuel tanks were empty when the take-off roll started. There are open fuel lines (no shut-off or check valves) between the wing tanks and the rear fuselage fuel tanks. When the take-off roll began, the forward acceleration was significant. According to NTSB, calculations during the 20 seconds take-off roll the fuel shifted enough to cause the c.g. to shift 2.5% m.g.c. aft of the aft allowable limit. This, coupled with a rapid rotation induced by the pilot caused the airplane to rotate to an angle of attack above stall. The stick shaker and pusher both activated but the right wing stalled first and the 40 degree bank could not be arrested before the airplane crashed.

6.39.3 Solution

Evidently, if the airplane had not been rotated so aggressively the accident might not have happened. However, if it had been made impossible (or very difficult) for the fuel to shift aft, the accident might not have happened either.

The fact that the fuel shifted so significantly could (and should) have been predicted and the effect on controllability after lift-off analyzed.

6.39.4 Lessons

1. Murphy's Law can be interpreted to say: "if fuel can shift in the presence of large forward accelerations, it will."

2. Design engineers should have been aware of this phenomenon, particularly after the event of Section 6.16.

6.40 Landing Gear Door Design

6.40.1 Problem

In October of 2001 a Quantas Link Boeing 717-200 (Figure 6.60) en route from Brisbane to Coolangatta (Australia) experienced a low right hydraulic quantity warning.

A description of what ensued is contained in Ref. 6.35. Since the rudder had now reverted to the manual mode and two ground spoilers would not operate as a result of the loss of hydraulic system No.2 the pilot decided to return to Brisbane where a longer runway was available. With the right hydraulic system turned to the off position, the landing gear had to be manually lowered using the emergency gear extension lever.

Figure 6.60 Boeing 717-200 (Not accident aircraft, Courtesy Carsten Bauer)

However, that operation did not close the main landing gear doors after the gear was extended. In accordance with the abnormal check list an attempt to close the doors was conducted by the crew after receiving the green down and locked indication for the landing gear. However, following the selection of the No.2 hydraulic system to “ON”, a rapid drop in hydraulic fluid quantity was noticed so the “OFF” position was immediately reselected before the doors had closed.

As the aircraft touched down, the main landing gear doors contacted the runway surface. Although the doors were fitted with non-sparking polyurethane rest bumpers, the runway centerline lights were contacted creating sparks that were observed by ground personnel. The aircraft was brought to a halt on the high speed taxiway where an engineer was requested to manually close the gear doors.

6.40.2 Cause

An inspection of the airplane revealed that a hydraulic line from the right engine-driven hydraulic pump had failed at its brazed fitting, resulting in the loss of hydraulic fluid from the No.2 system. Because this was not the first time the operator had experienced such a failure of hydraulic lines, the aircraft manufacturer was contacted. It was determined that the hydraulic lines were being subjected to vibration from the engine-driven hydraulic pump, which in some cases resulted in fracture of the line fittings.

6.40.3 Solution

Boeing issued an All Operators Letter (AOL) 717-048 on January 18, 2002, recommending the installation of a pulsation attenuator (damper) at the outlet of each engine-driven hydraulic pump.

6.40.4 Lessons

1. An interesting question is whether it is good design practice to have the landing gear hydraulic system powered only by the No.2 system (this feature was inherited from its DC-9 lineage). Why not both No.1 and 2 plus still an emergency system?

2. Main landing gear doors should be designed so that they do not scrape the runway if and when they cannot be retracted for whatever reason.

6.41 Moisture Ingress I

6.41.1 Problem

The following material was adapted from Ref. 6.36. In December of 2001 a Virgin Blue Boeing 737-33A (Figure 6.61) on a flight from Townsville to Brisbane in Queensland, Australia, experienced a master caution light illuminating, indicating failure of the cabin pressurization system. The airplane was at 33,000 ft at the time.

Figure 6.61 Boeing 737-33A (Not accident aircraft, Courtesy Jay Piboontum)

The crew completed the non-normal procedure and, as cabin pressurization was being maintained, they decided to continue the flight to Brisbane. About one half hour later the flight crew experienced physiological sensations which indicated that the flight deck was de-pressurizing. The crew donned their oxygen masks and the co-pilot noted that the cabin rate of climb indicator was displaying a rate of climb of 4,000 ft/min. When the cabin altitude reached 10,000 ft the cabin altitude warning horn sounded. An emergency descent was carried out and the airplane made an uneventful landing at Brisbane.

The flight crew noticed a slight un-commanded movement of the rudder pedals on two occasions during the descent. They also noticed that the map display on the (Electronic Horizontal Situation Indicator (EHSI) was incorrect: it disagreed with indications on other navigational instruments.

6.41.2 Cause

Following the incident, the electrical/electronic (E/E) bay was inspected and water was found to be dripping from the forward galley into the bay. There was also evidence of moisture leakage under the forward passenger door and service door. Moisture stains were found on the racks and ducting in the E/E bay. The inspection also found that the moisture shroud was missing from above the E1 rack in the forward part of the bay.

It was found that the airplane had been undergoing heavy maintenance and modifications.

One modification was the removal of the forward air stair from under the forward passenger entry door. During that work the air stair drip pan and moisture shroud were removed from the E/E bay. The documentation covering the removal of the air stair specified that the moisture shroud was to be replaced following the modification work. However, the shroud was not replaced because the kits were temporarily not available from the manufacturer.

One of the operator’s engineers, authorized by the Australian Civil Aviation Authority to approve a design modification or repair, assessed that the absence of the moisture shroud would not affect the safety of the aircraft. This engineer approved an amendment to the engineering release that permitted the installation of the shroud within 12 months of receipt of the parts.

During the investigation of this incident, Boeing advised that these shrouds are required in order to ensure the airworthiness of the airplane. Boeing 737 airplanes should not be used for revenue flight without the shroud installed.

The investigation also revealed that there have been several reports of 737 airplanes experiencing un-commanded flight movement due to moisture ingress into certain electronic components in the E/E bay.

6.41.3 Solution

The shroud was installed and the damaged equipment repaired.

6.41.4 Lessons

1. The use of removable shrouds to keep moisture out of E/E bays is not a good idea. Remember Murphy's Law? The E/E bay should have a permanent protection from moisture ingress.

2. Be careful when DER's sign off on modifications involving anything dealing with flight crucial equipment. An independent safety review should be conducted.

6.42 Electrical System Design II

6.42.1 Problem

The material in this section was adapted from Ref. 6.37. In December of 2001 a Saab SF-340B (Figure 6.62) experienced a failure of the co-pilot's two electronic flight information system (EFIS) screens.

An emergency descent was initiated. During this descent a number of cockpit warnings and cautions activated and some aircraft systems failed. The crew became aware that the right DC generation system was operating abnormally. Their attempts to rectify this were not successful and the airplane diverted to Cloncurry and landed.

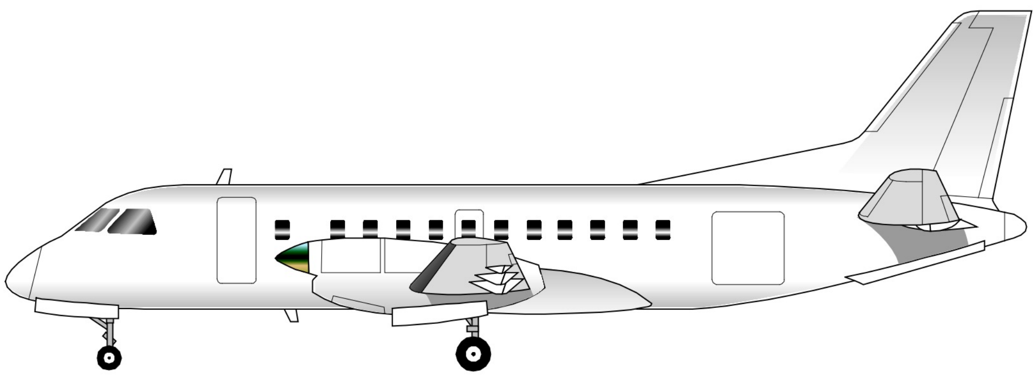


Figure 6.62 Saab SF-340B

6.42.2 Cause

The failure of the EFIS screens and the subsequent warnings were consistent with a right system voltage drop from the rated 28 VDC to below 18 VDC. During the investigation it became apparent that in some Saab SF-340 aircraft a starter generator could fail without taking the generator off line and alerting the crew, resulting in low system voltage. In this incident the crew overlooked the first item of the EFIS failure-disturbance check list, which required a check of the generator voltage. Consequently, the crew did not recognize the developing low voltage condition that led to the cascading series of warnings, cautions and failures. The bus tie relay, which was designed to automatically connect the two main electrical systems in the case of generator failure, did not operate.

6.42.3 Solution

An optional generator control unit modification, to prevent un-alerted low voltage conditions, had not been incorporated. The investigation determined that the modification to reduce the risk of the consequences of a delayed generator failure warning was highly desirable.

The investigation found that the operator's maintenance control system and approved system of maintenance did not ensure that the starter generator was maintained properly.

6.42.4 Lessons

It is not reasonable to expect flight crews to always do the right thing in case of an emergency. In this case the weather was good, but if it had not been, the cockpit confusion, poor visibility and instrument failure driven attention distracters (high pilot workload) could have resulted in a less benign ending. The basic system design would have made this possible.

Although a modification had been designed, it was optional. It should have been mandatory.

This occurrence demonstrates the need for pilots to be familiar with the systems of the aircraft they operate. Better yet, it demonstrates the need for designers to foresee such occurrences and design the system so that it cannot happen.

6.43 Icing of Stall Warning System

6.43.1 Problem

The following material has been adapted from Ref. 6.38. In June of 2002 a Saab SF-340 (Figure 6.63) experienced stall followed by a loss of control event while flying in icing conditions at 3,800 ft altitude.

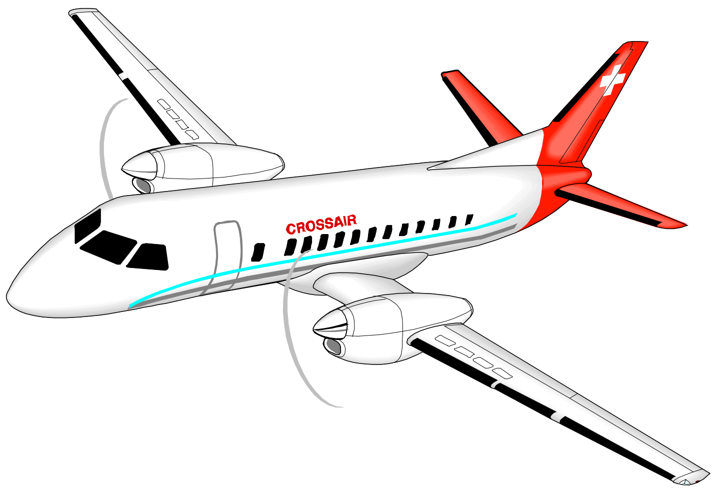


Figure 6.63 Saab SF-340

The airplane was recovered at 112 ft above the ground and did land without further incident.

6.43.2 Cause

The Australian Transport Safety Bureau (ATSB) found that the airplane can stall before activation of the stall warning system if there is ice on the wing and noted reports of similar incidents involving this type.

6.43.3 Solution

In Canada, Saab 340 aircraft are required to have a manually selected ice stall warning option. However, this system simply raises the airspeed at which the stall warning is given. It also presumes that the crew has activated the system when the aircraft enters icing conditions. At best this is a “patch-work” solution until a better one has been developed. The problem is being studied by Saab and by the Swedish Civil Aviation Administration.

6.43.4 Lesson

Icing continues to be a significant operational problem in certain types of airplanes. Manufacturers of such airplanes should conduct realistic flight tests to develop ways to generate stall warnings that are driven by the type of icing the airplane encounters.

6.44 Moisture Ingress II

6.44.1 Problem

The following material was adapted from Ref. 6.39. In January of 2003 a Fokker F-27 Mark 50 (Figure 6.64) experienced a failure of the cabin pressurization system while at 25,000 ft. The crew commenced a descent with the engines at flight idle.

Figure 6.64 Fokker F-27 Mark 50 (Not accident aircraft, Courtesy Simon Coates)

The flight descended to 10,000 ft and continued on to its destination since apparently none had suffered adverse effects.

6.44.2 Cause

The cause turned out to be moisture ingress. In this airplane there is it is an electrical junction box on the right main landing gear oleo. This junction box contains the wiring and connectors for the right main gear "weight-on-wheels" micro-switch. This micro-switch activates 12 different relays that are linked to avionics systems, warning and inhibit systems, the pressurization system and engine ground controls.

An inspection revealed that this junction box had been contaminated with moisture through inadequate sealing of the box cover following routine maintenance. The moisture ingestion led to spurious electrical signals being sent to the aircraft's pressurization system resulting in erratic cabin altitude control.

6.44.3 Solution

The solution was to increase the frequency of maintenance inspections on the main landing gear, including the junction boxes and introduced a detailed inspection that includes removal of the junction box cover, inspection of the connections and re-sealing of the cover.

6.44.4 Lesson

The wisdom of locating a critical electrical junction box in the landing gear area should be questioned. Landing gear wells always are subjected to water, slush and salt sprays.

6.45 Flap/Slat System Design

6.45.1 Problem

The following material has been adapted from Ref. 6.40. In May of 2003 a Quantas Link Boeing 717-200 (Figure 6.65) took off from Melbourne, Australia for a flight to Coolangatta.

Following a normal take-off the pilot-in-command (PIC), the handling pilot, called for the landing gear to be retracted. A short time later, he noticed an amber warning appear on the airspeed scale on his primary flight display (PFD). In response to that warning the PIC reduced the pitch attitude of the airplane. At about the same time, he noticed that the flaps/slats lever was at the "slats retract" position. The PIC immediately called for the flaps to be re-positioned, but the co-pilot selected the landing gear up. The PIC again called for the flaps to be re-positioned and the co-pilot then returned the flap selector to the take-off position.

Figure 6.65 Boeing 717-200 (Not accident aircraft, Courtesy Carsten Bauer)

The PIC reduced the pitch attitude further. The airspeed then quickly increased to 15 knots above the reference speed as the flaps reached the take-off position. The PIC re-established the normal climb attitude and the flaps and slats were subsequently retracted in accordance with the normal profile. The remainder of the flight was uneventful. Both the PIC and the co-pilot believed that the stick shaker had activated momentarily during the sequence.

Examination of the flight data recorder revealed:

- Three seconds after the aircraft became airborne, and at 30 ft above ground level, the flaps/slats lever was moved from the take-off position and the flaps began to retract.
- One second later, as the flaps/slats reached the retract position, the stick shaker warning commenced. At that time the aircraft pitch angle was 18.6 degrees and the computed airspeed was 157 kts. Over the next three seconds, the stick-shaker warning continued and the aircraft pitch angle reduced to 10.2 degrees. A second later the landing gear handle was recorded in the “up” position.
- The flaps/slats lever began to move from the fully retracted position about one second after the landing gear handle reached the up position. The flaps reached the fully retracted position in less than two seconds later, before immediately beginning to extend again. The slats began to retract but did not reach the fully retracted position before moving back to the extended position. The flaps/slats movement was accompanied by a very brief re-activation of the stick-shaker and a further reduction in aircraft pitch angle to about 6 degrees. Computed airspeed at that time was 165 kts. The aircraft then began to accelerate and quickly returned to a normal climb profile.
- One altitude loss of 5 ft, and lasting less than three seconds, coincided with the reduction in pitch angle that was made in response to the second stick-shaker activation. At that

time the airplane was more than 240 ft above ground level. There was no loss of altitude associated with the first stick-shaker activation.

6.45.2 Cause

According to Ref. 6.40 there was no obvious issue that may have led the co-pilot to retract the flaps/slats instead of the landing gear. Ref. 6.40 refers to this as an “action slip”. What is troubling is that further investigations revealed three other instances where this happened on this aircraft. All those events occurred above 3,000 ft altitude.

6.45.3 Solution

In response to these occurrences the airline amended its procedures for flaps/slats retraction to include the following CAUTION note:

“when retracting flaps/slats to UP/RET, pause at the UP/EXT position until the flaps indicate UP on the PFD prior to retracting the slats. Never move the flaps/slat handle to UP/RET in one motion.”

6.45.4 Lesson

The author believes that this a questionable solution to a problem that has occurred several times and which, at some future time when taking off into a dark night over water, can lead to a serious problem. It seems that this is an ergonomic design problem which needs to be solved urgently.

6.46 Galley Chiller Fan Blade and Wiring Failure Causes in Flight Fire

6.46.1 Problem

The following material has been adapted from Ref. 6.41. In December of 2003 a British Airways Boeing 747-436 (Figure 6.66) suffered an in-flight fire in the forward cargo bay while on approach to Sydney, Australia.

Figure 6.66 Boeing 747-436 (Not accident aircraft, Courtesy geminijets.com)

The fire was brought under control by the aircraft fire suppression system and the airplane landed in Sydney without injuries.

6.46.2 Cause

A galley chiller boost fan was installed in the airplane to provide forced air circulation over the forward galley chiller units to increase their cooling efficiency. The system incorporated a vane-axial-type three phase fan, powered by the aircraft's No.3 alternating current electrical system. Control power was supplied by the aircraft's direct current electrical system, with operation being automatic on selection of the galley chillers "ON".

The cause of the fire was determined to be chafing of the armature of the chiller fan motor. The investigation determined that this was most likely caused by a bearing failure. In turn this led to fan blade chafing which led to electrical arcing. The electrical arcing apparently set an insulation blanket on fire. The blanket was probably contaminated with combustible materials.

6.46.3 Solution

Boeing issued alert service bulletin SB747-21A2427, directing inspection and corrective routing of the electrical wire loom to the boost fan.

6.46.4 Lesson

When deciding on the routing and location of high power electrical wiring extra caution must be taken because electrical arcing is always a possibility. It would not make any sense to loose an airplane over a fire originating in a chiller boost fan. As always: the little things need detailed attention in airplane systems design.

Chapter 7

Lessons Drawn from Maintenance and Manufacturing

"Question for designers: Should an error in manufacturing or maintenance cause a fatal accident?"

Dr. Jan Roskam, 2007

7.1 Introduction

In this chapter a series of problems which arose in maintenance and manufacturing operations are reviewed. Where applicable, causes and solutions are described and lessons learned are stated. Connections with other areas of design are identified in Appendix A.

7.2 Propeller Blade Separation in Flight

7.2.1 Problem

In August of 1950 an American Airlines Douglas DC-6 (Figure 7.1) sustained a No. 3 propeller blade failure in cruise flight.

Part of the blade pierced the fuselage and depressurized the cabin. The resulting propeller unbalance tore the No. 3 engine loose which then fell from the airplane. A safe emergency landing was made at the Stapleton Airport in Denver, Colorado. There were 54 passengers and a crew of 5 on board. Five passengers and one cabin attendant sustained minor injuries. One passenger died, presumably from a heart attack. The aircraft was extensively damaged.

Figure 7.1 Douglas DC-6 (Not accident aircraft, Courtesy www.prop-liners.com)

There was no hydraulic pressure to operate the flaps nor the brakes. The captain made a flaps-up landing and stated that he would have done so regardless of the availability of flaps. Braking was done with reversal on the Nos. 1, 2 and 4 engines and emergency compressed air to operate the brakes.

7.2.2 Cause

In Ref. 7.1 the Safety Board determines that the probable cause of this accident was the internal gouging of a propeller blade during the manufacturing process which resulted in a fatigue fracture and subsequent failure during flight. Inspection of the airplane revealed a nearly vertical slit through the ice striker plate on the right side of the fuselage. This slit was about 36 inches long by 2 inches wide and located below the center of the fuselage slightly to the rear of the propeller plane of rotation.

There also was a large irregular opening on top of the fuselage of about 250 square feet. This extensive fuselage damage was why the captain became worried about the structural integrity of the airplane and decided to make a flaps-up landing. The failed propeller blade parts were recovered and sent to the National Bureau of Standards for examination. Their report stated:

"The failure was caused by a fatigue fracture which originated at one of several defects which were points of stress concentration on the inside surface of the flat side of the blade. These defects, which occurred prior to heat treatment and painting on the inside of the blade, appeared to have resulted from a gouging or galling action due to rubbing against another surface such as a mandrel."

Ref. 7.1 states:

"This propeller blade was manufactured by the Propeller Division of the Curtiss Wright Corporation. It is steel, hollow, Model 744-602-0, Serial No. 292695.

In the fabrication of this model blade, the two surfaces are formed and shaped separately and then welded together. During the welding the two parts are positioned by a mandrel within the blade controlling the distance between the two surfaces. This mandrel has extendable side mandrels controlling the weld locations. The entire device is rigid in use but is necessarily made collapsible so that it may subsequently be withdrawn from the relatively small opening in the shank end of the blade. The positioning of the side mandrels is by means of two cam adjustments in the center mandrel. These cams are locked in position by Allen head set screws. Two parallel and longitudinal gouges found in the inside surface of the flat (rear or thrust) side of the failed blade were located and spaced closely corresponding to the location and spacing of these Allen head set screws. The bottoms of these gouges were irregular, the maximum depth of gouge being approximately one-sixth of the blade's wall thickness at that station.

At the time the subject blade was manufactured, it was subjected to a number of tests and inspections. One such inspection was by means of X-ray photographs. These original X-ray negatives were on file with the manufacturer. On examination, they bore faint marks indicating internal defects."

7.2.3 Solution

An examination of similar records of other blades showed an extremely small percentage of blades with similar defects. Steps were taken to reject these blades. Measures were also taken to tighten up the inspection criteria for the acceptance or rejection of blades.

7.2.4 Lesson

When considering a manufacturing process, seemingly unimportant imperfections caused by "not quite flush" Allen head screws in a manufacturing tool can cause serious problems. This should be considered during the definition phase of such a manufacturing process.

7.3 Elevator Control Bolt Backed Out I

7.3.1 Problem

In September of 1953 a Resort Airlines Curtiss C-46F, Commando (Figure 7.2) crashed during landing. There were 25 fatalities, including the crew of three, while 16 passengers received serious injuries.

Figure 7.2 Curtiss C-46F Commando (Not accident aircraft or airline, Courtesy G. Helmer)

7.3.2 Cause

According to Ref. 7.2 the probable cause was the structural failure of the left elevator in flight, causing loss of control. This structural failure was brought about by the left outboard hinge bolt backing out of the assembly. The underlying cause was improper maintenance which resulted in the installation of hinge bolts and bearings which did not meet specifications. Inadequate inspection failed to detect this condition.

It turned out that the bolt which was installed in the No.1 hinge had 1/8 inch less grip length then the approved part. This resulted in several threads resting on the bushings of the hinge bracket. The bolt used also had a tolerance which permitted a slightly smaller diameter than what was recommended. Vibration of the elevator in its bushings was the result. Detailed inspection of the accident airplane revealed that three of four bearings on the left elevator were not of an approved type.

7.3.3 Solution

Strict adherence to manufacturer's recommendations is important, in particular where it concerns the primary flight control system of an airplane.

Maintenance on this aircraft was contracted out to Slick Airways at San Antonio, Texas. Review of maintenance records and procedures revealed significant problems and lack of compliance with CAA (Civil Aviation Authority) approved standards.

7.3.4 Lessons

1. Contracting out maintenance should be carefully watched both by the aircraft owner and by the FAA which followed the CAA.

2. From a design viewpoint: one maintenance error should not cause a catastrophe.

7.4 Elevator Servo Tab Bolt Backed Out

7.4.1 Problem

In January of 1955 a United Airlines Convair 340 (Figure 7.3) made a wheels-up emergency landing near Dexter, Iowa.

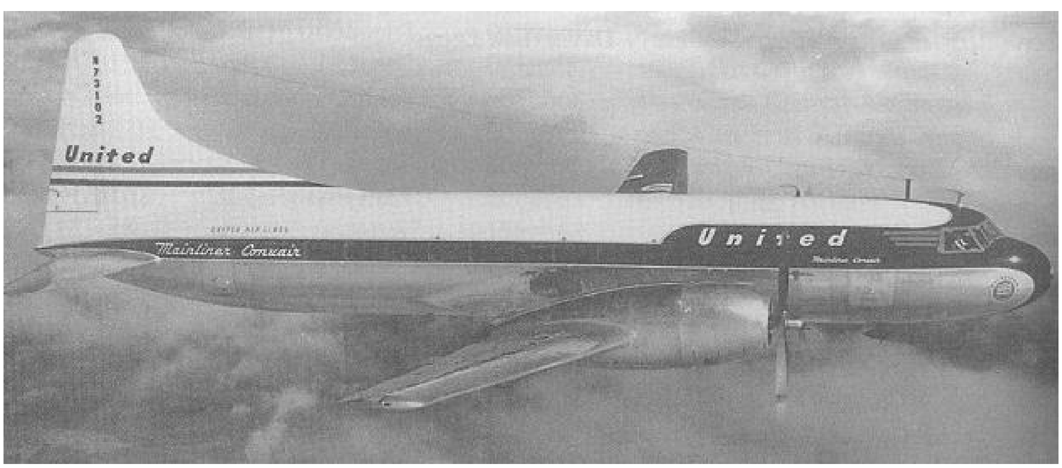


Figure 7.3 Convair 340 (Not accident aircraft, Courtesy www.prop-liners.com)

A few of the 36 passengers received minor injuries. The crew was not injured.

7.4.2 Cause

The following material was adapted from Ref. 7.3. The airplane had left Des Moines, Iowa and was carrying out a normal climb to 5,000 ft when the crew noticed vibration and a slight fore-and-aft movement of the control column. After trying to locate the problem elevator control was almost totally lost and an emergency was declared. The airplane made several severe nose-up and nose-down gyrations but the captain was successful in landing the airplane in a field.

The day before the crash the airplane was in a UAL shop for a 1,500 hour maintenance check. It was found that there was excessive play in the (left) elevator servo tab. A worn idler support bolt was found to be the cause. A replacement part was not in stock and a part had to be ordered from the UAL San Francisco base. Somehow the worn bolt found its way back into the system and was put back in place but not safetied. The job card contained no indication that this was a temporary installation, contrary to UAL maintenance policy.

Work on the airplane continued in a normal manner. When completed the supervisor noted that the subject non-routine job card had not been signed off as complete. The mechanic informed the supervisor that there was no longer excessive play in the tab. The supervisor also could not find any play and he signed the card adding the notation “OK for service.

The airplane was put back into service and the accident happened the next day: the un-keyed castellated nut which fastens the idler assembly support bolt in its bracket backed off because of vibration. This permitted the bolt to back out which eventually led to servo tab oscillations causing the forward and aft movement of the control column. The tab oscillation imparted an oscillation to the left elevator which now was out of phase with the right elevator. At some point this broke the torque tube between the two elevators and the pilots were left with only the left elevator but without the use of the servo tab.

The Board found that the probable cause was the release of the airplane from maintenance in an un-airworthy condition resulting in almost complete loss of elevator control in flight.

7.4.3 Solution

Follow the company maintenance procedures to the letter.

7.4.4 Lessons

1. It does not seem acceptable that the failure of one mechanic to do his job and a supervisor to not question job performance can lead to a serious crash.

2. From a design viewpoint: one maintenance error should not cause a catastrophe.

7.5 Engine Maintenance Error

7.5.1 Problem

In August of 1955 an American Airlines Convair 240 (Figure 7.4) crashed near Fort Leonard Wood in Missouri.

Figure 7.4 Convair 240 (Not accident aircraft, Courtesy of www.prop-liners.com)

The airplane was seen to be on fire on the right side and the right wing was seen to separate from the airplane before the crash. The three crew members and 27 passengers were killed.

7.5.2 Cause

Ref. 7.4 gives as the probable cause the installation of an un-airworthy engine cylinder, the fatigue failure of which resulted in an uncontrollable fire and subsequent loss of the right wing in flight.

The background of how the un-airworthy cylinder was installed in the No.2 engine of this airplane is disturbing. The history of how this and 23 other un-airworthy cylinders were re-installed in AA airplanes revealed a breakdown in proper maintenance procedures, including the required record keeping at the Tulsa, OK based maintenance facilities.

The detailed accident investigation of Ref. 7.3 showed that "failure of the No.12 engine cylinder was accompanied by the release of combustibles consisting of a fuel-air mixture from the disrupted intake pipe and oil from the crankcase section. The most likely source of ignition was the exhaust manifold which is routed rearward of the cylinders.

The No.12 cylinder straddles the mating line of the lower and inboard side orange peel cowls. After the cylinder failed fire passed rearward into zone 2 at the lower left corner of the diaphragm, which is aft of the No.12 cylinder. It is believed that fire progressed into zone 2 quite rapidly. The fire path in that zone is in accord with the zone 2 air flow pattern and the location of original entry of fire into zone 2. More significant is the exit of fire from zone 2, which occurred at the mating line between the lower cowl and both side cowls at and behind the rearmost fasteners. Fire on the inboard side burned the aluminum nacelle skin back of the firewall and between the upper and lower nacelle longerons, permitting fire entry into zone 3.
The crew must have become aware of the engine difficulty and initiated emergency procedures at once. Relatively minor damage to the No.12 link rod, which was free to flail after the cylinder let go, indicates an almost immediate feathering of the propeller. This would halt the release of combustibles in zone 1 and account for the comparatively light fire damage in that area.

That CO_2 was discharged in flight is evidenced by the fact that all CO_2 bottles were found empty with their heads, including the thermal disks, intact. It is therefore reasonable to assume that the fire extinguishing system was actuated at the time called for in the emergency procedure checklist.

The emergency procedure for in-flight fire consists of two phases, the second part being a "cleanup" list for items considered less urgent than those related to controlling and putting out the fire. One of these items near the end of the list is to close the main fuel tank shutoff valve."

The accident investigation also revealed that this right main tank fuel shut-off valve was found in the open position. This certainly contributed to the severity of the fire. The Safety Board opined that consideration should be given to moving the closing of this main tank shutoff valve way up on the emergency procedures checklist.

After the fire spread into zone 3 it contacted the front spar and weakened the aluminum material to the point of wing failure.

7.5.3 Solutions

The maintenance procedures and record keeping were improved.

The closing of the main tank fuel shut-off valve was moved up on the emergency procedures checklist.

7.5.4 Lessons

1. Maintenance procedures and record keeping for flight crucial items should be watertight and rigidly followed. At that time the CAA had insufficient manpower to see to it that this is enforced. That still is the case with the FAA today.

2. Consideration should be given to better fire proofing of engine installations. The scenario of this accident could have been predicted and additional stainless steel firewalls protecting the wing spar from the fire might have prevented most casualties.

7.6 Propeller Reversal in Flight

7.6.1 Problem

In November of 1958 a Seaboard & Western Airlines Lockheed L-1049D Constellation (Figure 7.5) on a training flight became uncontrollable right after take-off, struck a parked Viscount airliner and skidded to a stop. The ensuing fire destroyed both airplanes.

Figure 7.5 Lockheed L-1049 Constellation (Not accident aircraft, Courtesy Mel Lawrence)

One of the five man crew of the Constellation received minor injuries and the flight attendant on the Viscount was slightly injured while running from the airplane.

7.6.2 Cause

According to Ref. 7.5 the probable cause was an unwanted propeller reversal at a low altitude occurring immediately after take-off. A contributing factor was the inadequate overhaul procedure used by the propeller manufacturer.

The investigation showed that right after take-off the No.1 propeller reversed at 117 knots and 25 feet above the ground. The left wing dropped and struck the runway causing the airplane to sharply veer to the left. Efforts of the crew to control the airplane were not successful and after crossing a taxiway and two perimeter strips the airplane collided with a parked Viscount airliner which was close to boarding passengers. Both airplanes were destroyed by fire despite efforts of fire fighters to contain the fires.

Investigation showed that the No.1 propeller was in the full reverse pitch position of 11.7 degrees. The wear of the rotor spline and mating speed reducer sleeve in the propeller hub was of sufficient magnitude to cause complete disengagement between the power unit motor assembly and the speed reducer, thus preventing electrical control of the propeller. These conditions would permit the centrifugal forces on the blades to move them to the flat pitch position and beyond.

It was found that this was not an isolated incident. In six other cases propeller reversals had occurred in cruising flight with temporary loss of control. However, in all those cases control was re-established and safe landings were made.

Also, when Seaboard & Western inspected their other propellers with 1,000 to 1,200 hours of operating time it was found that 14 out of 26 units had to be rejected because of excessive wear and or damaged oil seals. Clearly this is a case of pre-cursor incidents which, had they been properly acted upon might have prevented this accident.

7.6.3 Solution

The following is quoted from Ref. 7.5:

As a result of this accident, the Board submitted two recommendations for corrective action to the FAA. The first called for immediate inspection and re-lubrication of the splines of the armature rotor and speed reducer sleeve assemblies, as well as incorporation of the mechanical low pitch stop assembly, as soon as possible. On December 15, 1958, Airworthiness Directive AD 58-25-2 was issued requiring the mandatory inspection of the affected parts, not to exceed 1,250 operating hours.

Since issuance of this AD, one additional case of excessive spline wear was reported with less than 600 hours of service. As a result, AD 59-7-1, issued April 6, 1959, superseded the original AD. The AD 59-7-1 called for inspection of the armature and sleeve bearing fits at each 600 hours of service and in addition eliminated the use of molybdenum disulfide as a spline lubricant, thus approving only Lubriplate 315 as the approved lubricant. On September 8, 1959, AD 59-18-3 was issued requiring the installation of an improved model power unit on all Curtiss C34S-C400 and –C500 propellers. This model assembly incorporates a new armature rotor assembly with a longer shaft, with splines of a larger pitch diameter, and a new mating speed reducer splined sleeve, and high speed drive gear. Lubrication is provided by the speed reducer oil supply.

The Board's second recommendation to the FAA resulted from the high rejection rate by Seaboard & Western Airlines of the subject parts. It was evident from this fleet campaign that the inspection and quality control procedures of the propeller manufacturer were not conducive to required standards of airworthiness. A review board established by the FAA examined the overhaul facility of the Propeller Division of the Curtiss-Wright Corporation and made recommendations to the management for prompt corrective action of unsatisfactory procedures and conditions.

7.6.4 Lessons

1. One maintenance deficiency caused a serious accident.

2. Designers should take this into account when formulating maintenance and inspection procedures. The result of such an analysis may well be a redesign of the component at hand.

7.7 Landing Gear Truck Beam Failure

7.7.1 Problem

In July of 1959 a Pan American World Airways Boeing 707 (Figure 7.6) made a successful emergency landing at New York International Airport after losing two of the four wheels on the left main landing gear.

Figure 7.6 Model of Boeing 707 (Courtesy geminijets.com)

7.7.2 Cause

According to Ref. 7.6 the probable cause of this accident was the failure of the forward truck beam of the left landing gear.

The loss of the two wheels was caused by a failure of the forward truck beam. This failure was induced by undetected damage to the beam when it contacted the lower torsion link assembly anti-rotation bolt. This could only have occurred on one of two previous flights. On one of the previous flights the snubber[2] failed on take-off and on the other flight the snubber had been removed. Either condition would have permitted the truck beam to pitch up sufficiently to cause this contact. The manufacturer has prepared several engineering changes which will prevent recurrence of this type of failure.

A review of the history of the landing gear on this aircraft revealed that a snubber failure had occurred on take-off in July of 1959. After landing the snubber was replaced. Two days later it was noted upon landing that the snubber terminal attach bolts were sheared. The bolts were replaced with temporary bolts and the airplane returned to service. After the next landing, inspection showed that the temporary bolts had sheared.[2]

7.7.3 Solution

As a result of a study of snubber failures, the manufacturer has prepared several changes which it is anticipated will eliminate further difficulty. The size of the orifices in the hydraulic piston of the snubber assembly are to be reduced to increase its load rate, and the pressure relief setting is to be increased from 8,000 p.s.i. to 12,500 p.s.i. This modification will increase the effectiveness of the damping action of the snubber assembly.

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[2] A snubber is a small hydraulic shock absorber mounted in such a way as to damp oscillations of the landing gear truck beam relative to the main oleo strut. It also serves to limit the displacement of the truck beam to a maximum of 10 degrees above and 15 degrees below the horizontal. The design of this landing gear is such that this type of damage can occur only if the snubber is removed or broken from one end of its fittings. The effect of beam over-rotation causes the beam to be peened by the torsion link assembly anti-rotation bolt. The frequency of occurrence of these incidents indicates, according to the Safety Board, improper maintenance practice.

The automatic wheel brake valve setting is to be reduced from 450 p.s.i. to approximately 175 p.s.i. This should decrease the tendency of unequal braking causing the oscillation of the truck beam.

Finally, the pressure in the leveling cylinder assembly is to be increased from 925 p.s.i. to 1,500 p.s.i.

7.7.4 Lesson

Proper simulation of the beam motion as a result of unequal braking might have led to an earlier discovery of this problem.

7.8 Elevator Control Bolt Backed Out II

7.8.1 Problem

In September of 1959 an AAXICO Curtiss C-46 (Figure 7.7) crashed on the runway at Dyess AFB near Abilene, TX. The crew of two were killed.

Figure 7.7 Curtiss C-46 (Not accident aircraft or airline, Courtesy G. Helmer)

7.8.2 Cause

In Ref. 7.7 the CAB determines that the probable cause of this accident was the loss of elevator control because of an improperly secured bolt, a condition which was undetected because of inadequate maintenance.

The airplane had flown two hours since the last No.2 inspection during which the unsecured bolt was not detected.

7.8.3 Solution

Better discipline, checking and cross-checking of maintenance on all items dealing with flight crucial systems.

7.8.4 Lesson

Primary flight control paths should be designed with at least one level of redundancy. Of course, there is no guarantee that mechanics will not leave two bolts unsecured either.

7.9 Loss of Roll Control

7.9.1 Problem

In September of 1961 a Northwest Airlines Lockheed L-188C Electra (Figure 7.8) crashed right after take-off from O'Hare International Airport near Chicago, IL. All 32 passengers and the crew of five sustained fatal injuries.

Figure 7.8 Lockheed L-188C Electra (Not accident aircraft, Courtesy Bob Garrard)

7.9.2 Cause

According to Ref. 7.8 the probable cause of this accident was a mechanical failure in the aileron primary control system due to an improper replacement of the aileron boost assembly, resulting in loss of lateral control at an altitude too low to effect recovery. Figure 7.9 shows a layout sketch of the Electra lateral control system.

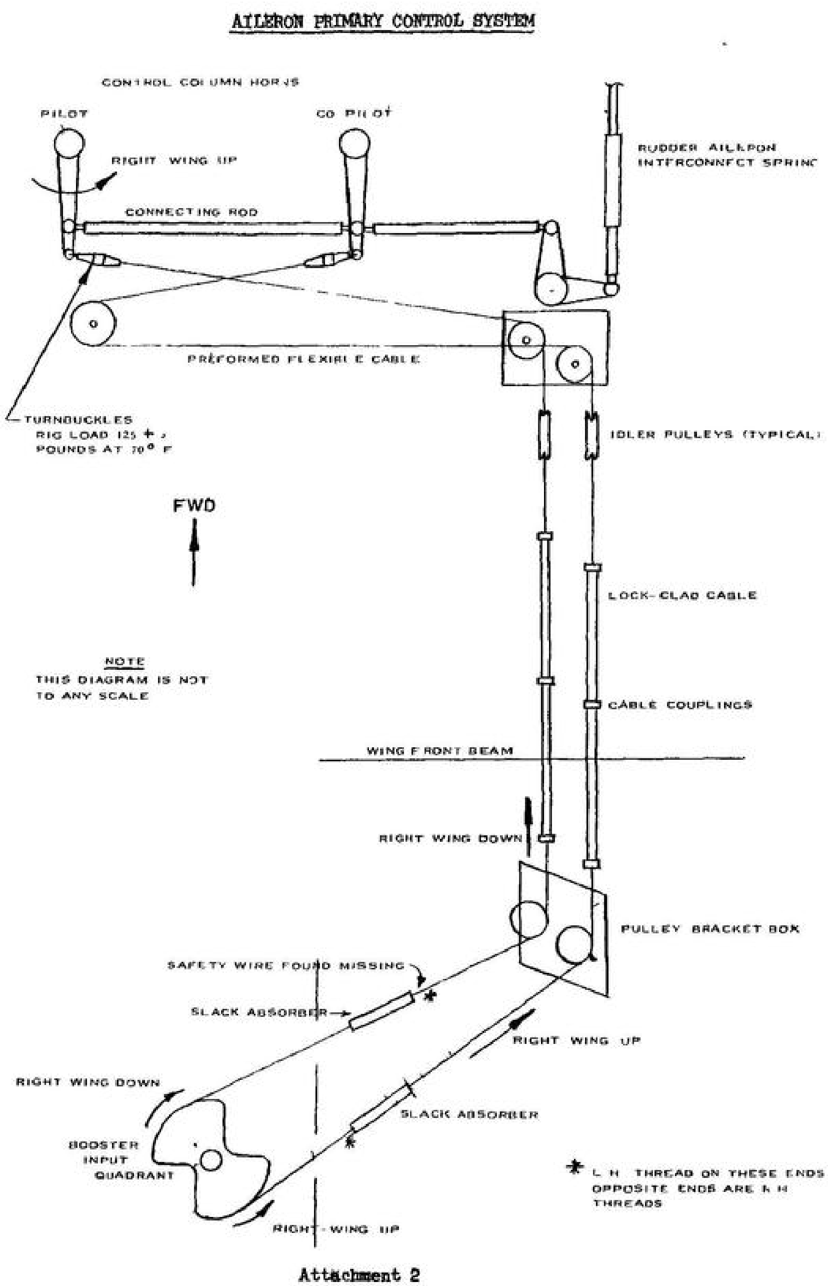

Figure 7.9 Lateral Control System Lockheed L-188 Electra
(Courtesy Civil Aeronautics Board)

Note the area where the safety wire was found missing. As a result of this the cable end unscrewed itself from the cable slack absorber. This in turn imparted a signal to the aileron boost unit for a right wing down movement. The resulting uncommanded roll could not be arrested by the crew, leading to the crash.

The following material has been adapted from Ref. 7.8 to indicate the serious breakdown in maintenance procedures and checks which led to this accident.

Between June 27 and July 11 eight aileron control discrepancies were reported by pilots on the logs of N137US. During this time period 29 scheduled flights were completed in this airplane.

The aircraft logs recording corrective actions taken indicate that little effort was made to analyze the cause of these discrepancies and to correct them. This type of operation reflects a casual attitude on the part of maintenance personnel toward a potentially hazardous condition. The aileron boost unit was removed on July 11. To facilitate this, the safety wire on the slack absorber was removed. Ref. 7.8 indicates that the removal of the boost unit and a later installation of another unit was done without adhering to detailed procedures prescribed by both Northwest and by Lockheed. The time interval between removal and installation covered four work shifts. Procedures to inform mechanics and supervisors for following shifts were violated. Also, many mechanics were found not to have been properly trained for the type of work they were asked to perform.

7.9.3 Solution

The maintenance department of the airline had to clean up their act.

7.9.4 Lesson

For designers the lesson is that one human failure in the installation of a critical path in a flight crucial system should not cause a fatal crash.

7.10 Quenching

7.10.1 Problem

This material is adapted from Ref. 7.9 (Kelly, p.142). In 1965, during the early production phase of the Lockheed SR-71 (Figure 7.10) it was noticed that certain parts, which had to be quenched in water to obtain the correct crystalline arrangement, failed when tested.

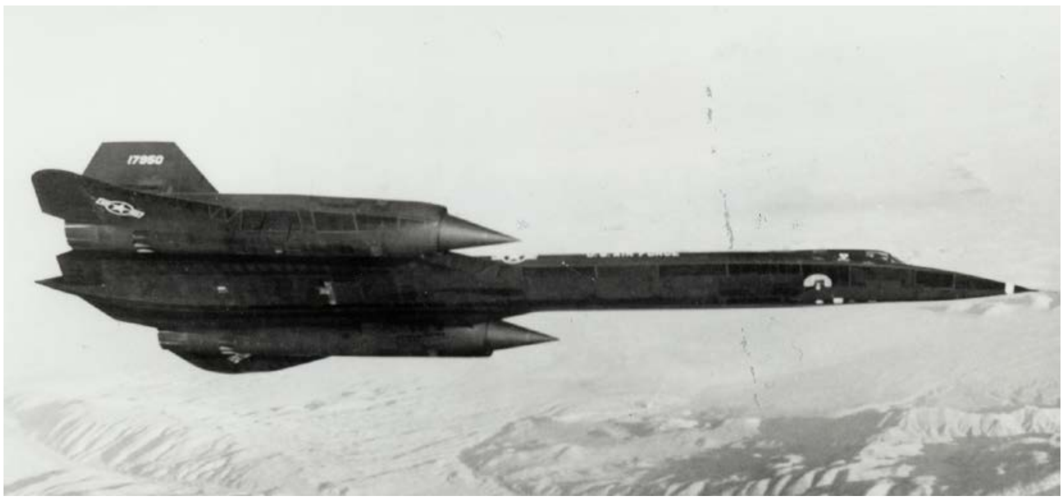

Figure 7.10 Lockheed SR-71 (Courtesy Royal Aeronautical Society)

7.10.2 Cause

It was observed that these problems only occurred during summer months, not during winter months. After quite a bit of detective work it was found that the Burbank City water department added certain chlorine compounds to the city water during the summer. These chlorine compounds caused alterations in the formation of crystals during the quenching process.

7.10.3 Solution

Once the cause was identified the solution was simple: filter the chlorine compounds out of the water before the quenching operations.

7.10.4 Lesson

The lesson is that sometimes a minute detail can really foul things up. It takes a lot of work to uncover such details.

7.11 Weight Control

7.11.1 Problem

The Windecker Eagle (Figure 7.11) was arguably the first certified general aviation airplane to be manufactured primarily of glass fiber composites.

Figure 7.11 Windecker Eagle (Not accident aircraft, Photo by Mark Avino, National Air and Space Museum, Smithsonian Institution (SI 85-11222-15))

In production it turned out that the empty weight could not be controlled within a reasonable margin (+/- 0.1%).

7.11.2 Cause

The cause was the choice of manufacturing process. A hand lay-up procedure with manually applied resin was used for manufacturing the airplane. The resulting airframes were certainly satisfactory from a structural and aerodynamic viewpoint. However, it proved impossible to control the empty weight within an acceptable margin. As a result the useful load varied from one airframe to another. FAR 23 certified airplanes may not exceed a certain maximum design take-off weight. If the empty weight varies, the useful load (difference between design take-off weight and empty weight) also varies. In practical operations that is not acceptable.

7.11.3 Solution

The solution is found in the selection of the manufacturing process. By automating the lay-up and resin impregnation process the weight might have been be controlled. However, the company ran out of money and went bankrupt.

7.11.4 Lesson

Designers of airplanes should have some familiarity with manufacturing procedures and consequences of their selection.

7.12 Incomplete Skin Bonding

7.12.1 Problem

In 1977 one accident and one incident occurred with the Cessna Conquest (Figure 7.12). Apparently, in both cases there was an intense vibration at the horizontal tail. In the first case the airplane fluttered apart and there were no survivors.

In the second case the pilot managed to slow the airplane down rapidly, the flutter subsided and a landing was made despite severe damage to the horizontal tail.

Figure 7.12 Cessna 425 Conquest (Not accident aircraft, Courtesy E. Marmet)

7.12.2 Cause

As is often the case, a combination of several factors caused the problem. These factors will be discussed in some detail to help raise the awareness of design engineers.

Factor 1: The horizontal tail is in the propeller slipstream. In a highly powered airplane this does lead to continuous excitation of the tail. The elevator trim-tab was actuated with a single control linkage (that is all that is required in a FAR 23 airplane). If the linkage becomes disconnected, the tab is free to oscillate.

Factor 2: The horizontal stabilizer in this airplane was of metal-bonded construction. Figure 7.13 shows a cross-section of the stabilizer. The skin was assumed to be bonded to the spar and rib flanges as indicated in the sketch. On the airplanes involved the bonding in the leading edge area was found to be incomplete.

Factor 3: In determining the flutter characteristics of an airplane structure the torsional stiffness is an important factor. This torsional stiffness depends on the ability of a structure to sustain a shear-flow along the periphery of the structure, in this case the torque box and the leading edge. In this case, the shear flowed around the torque box but not around the leading edge. Therefore, unbeknown to the engineers, the torsional stiffness was less than what was assumed. This had all been verified in a prototype where all bonds were satisfactory.

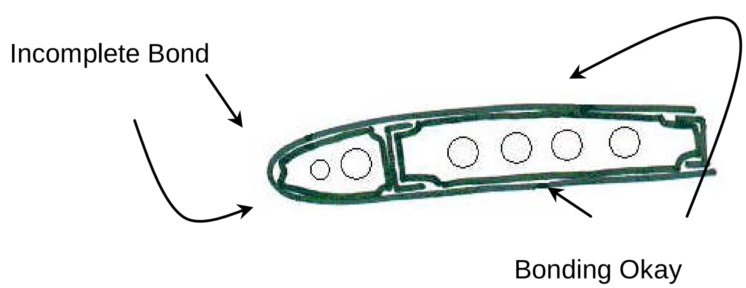

Figure 7.13 Sketch of Incomplete Bonding Areas

Factor 4: In a bonded structure with areas of small radii (such as the leading edge of a stabilizer) it will be found that it is very difficult to maintain adequate bonding in those areas. One sample may bond correctly but the next one may not.

Factor 5: When a flutter analysis of the stabilizer was conducted with the actual lower torsional stiffness AND with a disconnected tab the result was that flutter could occur at cruising speed. This is what happened.

7.12.3 Solution

The problem was solved by:

- Changing the tab design to include a dual control linkage (required on FAR 25 airplanes)
- The leading edge skins were riveted to the rib and spar flanges

The latter clearly was overkill but it has prevented recurrences. The airworthiness certificate was re-issued on 2-20-78.

7.12.4 Lesson

When stiffness inputs to a flutter calculation are critical (and they usually are), a double check of the attainability in view of manufacturing processes (in this case: bonding) and/or maintenance procedures (in this case the bolt connection to a tab) should be kept in mind. These questions should be raised during critical design reviews.

7.13 Drain Holes Forgotten

7.13.1 Problem

Some time in 1977 a pilot flying a new Cessna 441 (Figure 7.14) reported loss of elevator control during high altitude cruise.

Since he did have a bit of trim authority left he began a descent to make a precautionary landing. On final approach he noticed that elevator control authority had returned. Inspection of the flight controls after landing did not reveal any problem. Therefore the decision was made to continue his flight. During high altitude cruise the same thing happened again ending once more in a satisfactory landing with full control regained. That time Cessna was contacted.

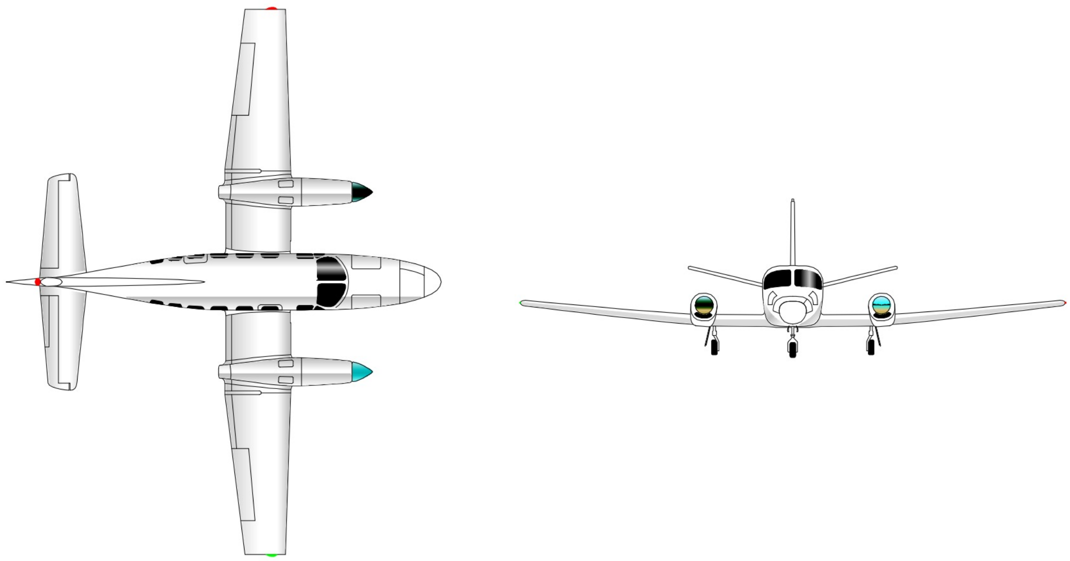

Figure 7.14 Cessna 441

7.13.2 Cause

With the help of Cessna expertise, it was soon found that at high altitude, water that had accumulated behind the rear pressure bulkhead had frozen as shown in Figure 7.15.

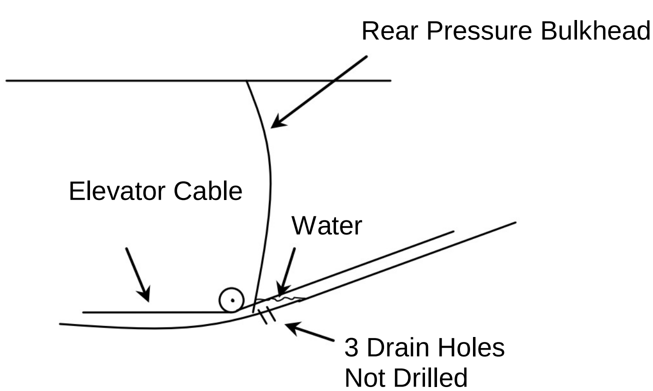

Figure 7.15 Sketch of Area Near the Rear Pressure Bulkhead

The frozen water simply locked the elevator control cable. When the airplane descended the warmer air unfroze the water and normal control was re-established.

It was found that the reason water had accumulated was that in this particular airplane three drain holes behind the rear pressure bulkhead had not been drilled during a manufacturing operation and inspection had not caught this. Other airplanes were checked and no problems were found.

7.13.3 Solution

Follow manufacturing and inspection procedures.

7.13.4 Lesson

Drain holes are usually extremely important. Accumulated water can affect the center of gravity and, if it freezes it can affect controllability. It is a smart idea to flag the need for drain holes in airplanes to emphasize their importance.

7.14 Maintenance Man-Hours per Flight Hour

7.14.1 Problem

In many versions of the F4 fighter airplane (Figure 7.16) the Identification of Friend or Foe (IFF) receiver was located underneath the rear ejection seat.

Figure 7.16 McDonnell Douglas F-4 (Not accident aircraft, Courtesy Royal Aeronautical Society Library)

The failure rate of that IFF system was rather high, and understood to be high in the early design phase. As a result, the maintenance man-hours associated with replacing the IFF were also very high.

Imagine the following maintenance sequence:

- Disarm the rear ejection seat
- Remove the ejection seat
- Remove the IFF
- Install a new IFF
- Re-install the ejection seat
- Re-arm the seat

7.14.2 Solution and Lesson

Items which require frequent replacement should be easily accessible. It should not be necessary to remove other equipment before access is possible. It is a good practice to write down maintenance sequences as soon as a decision is made to locate a component somewhere in the airplane. Doing this may help a designer realize that maybe the installation should be changed.

7.15 Placards on Inspection Covers

7.15.1 Problem

Sometime in 1992 the author was sitting in a new Delta Airlines twin-jet Boeing 737-300 (Figure 7.17) at Salt Lake City awaiting push-back.

Through the window the big cowl of the number one engine could be seen. There is a large cowl door which opens upward and is located between the nacelle and the fuselage. This door provides access to certain engine components.

Figure 7.17 Boeing 737-300 (Not accident aircraft, Courtesy Mark Van Drunen)

When the airplane is parked at the gate and the hydraulic system is not operating, the Krueger flaps between the nacelles and the fuselage tend to sag downward a bit. If the cowl door were opened in such a case it would interfere with (i.e. crunch into) the Krueger flap. Therefore, a large placard was painted on the cowl door. The placard read: “Do not open this cowl door unless the leading edge devices have been secured in the up position. For detailed instructions see inside of cowl door."

7.15.2 Cause

The problem, to my knowledge, never happened. The author then called a friend at Boeing and suggested that they may wish to change the placard. This was done, at least on Delta airplanes.

7.15.3 Solution

The solution is simple: engineers composing placards should anticipate what the consequences of literal interpretation of a placard can be and draw the appropriate conclusions.

7.15.4 Lesson

Engineers who compose placard instructions on airplanes should think about the consequences of their composition. Always go the extra mile: anticipate problems.

7.16 Inspection Cover not Large Enough

7.16.1 Problem

Early in January of 1993 the author was awaiting push-back in a Boeing 747 (Figure 7.18 shows the -400 version) at Schiphol Airport, near Amsterdam, The Netherlands.

It was bitterly cold and snowing. The airplane had been duly de-iced. Just before push-back the pilot announced that he wanted maintenance to check the oil in the number one engine. This would delay push-back a bit.

Pretty soon here came a blue KLM maintenance pick-up truck. A mechanic stepped out, dressed in full winter clothing including nice warm gloves. When he approached the nacelle cowl door he could not open it with his gloves on. So he took them off. When he touched the cowl his hand froze to it. It took awhile that day before take-off occurred.

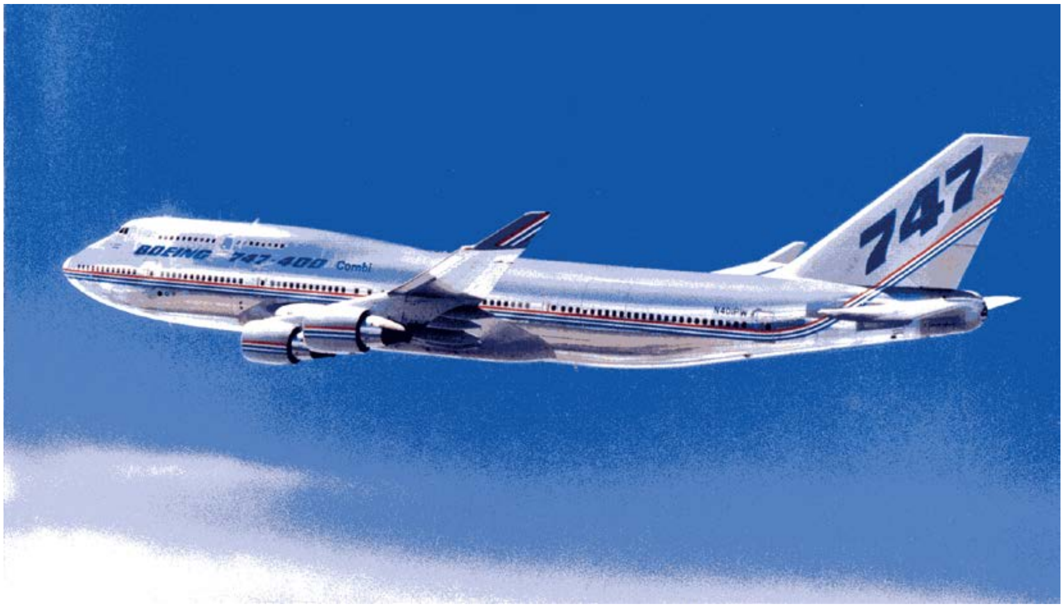

Figure 7.18 Boeing 747-400 (Courtesy Boeing)

7.16.2 Cause

The cause of this foreseeable problem was that the inspection cover latches had not been designed with the possibility in mind that someone would have to open it while wearing winter gloves.

7.16.3 Solution

Design inspection covers which have to be frequently opened for with quick opening latches so that maintenance personnel can open them while wearing winter clothing.

7.16.4 Lesson

Design all access covers for items requiring frequent ramp access so that people wearing winter clothing and gloves can open them. This lesson should be memorized by engineers involved in detail design activities.

7.17 Landing Gear Corrosion

7.17.1 Problem

In March of 1996 a Piper PA-23 Aztec (Figure 7.19) had landed in De Kooy, The Netherlands, slowed down and was turning of the runway.

Figure 7.19 Piper PA-23 Aztec (Not accident aircraft, Courtesy Digimicra@airliners.net)

At that point the left main landing gear fork assembly failed causing the airplane to rotate 180 degrees about its top axis before coming to a halt. The pilot and passengers were evacuated without injuries.

7.17.2 Cause

Ref. 7.10 states that the failure was caused by cracks in the collar attached to the oleo strut. The remaining material in the collar could not withstand the loads imposed during the turn off the runway.

A detailed investigation by the KLM Engineering Department revealed that corrosion cracking emanating from bolt holes triggered galvanic corrosion between the steel bolt and the aluminum collar. The fact that there was a lack of paint and corrosion protection of these parts assisted the corrosion.

7.17.3 Solution

Before assembly of parts made of dissimilar materials corrosion proofing must take place. Either the manufacturing/assembly drawings did not specify this for these parts or the corrosion proofing was accidentally omitted.

7.17.4 Lesson

When designing components made of dissimilar metals corrosion cracking is a real possibility. This is always the case with bolt holes in aluminum with steel bolts. Any textbook on the subject matter can tell you that.

7.18 Grit Blasting

7.18.1 Problem

In May of 2001 a McDonnell-Douglas MD-83 (Figure 7.20) of Spanair experienced a main landing gear oleo cylinder failure right after touchdown at London Heathrow.

Figure 7.20 McDonnell-Douglas MD-83 (Not accident aircraft, Courtesy L. Willems)

Despite the fact that there was a full load of passengers on board there were no injuries during the evacuation.

7.18.2 Cause

According to Ref. 7.11 there were two causal factors:

- The right main landing gear failed immediately upon touchdown due to the application of spin-up drag loads on a section of the oleo cylinder containing a major fatigue crack of 3.2 mm in length and 1 mm deep. There were several other cracks associated with this main crack.
- Although the origin of these fatigue cracks could not be positively identified, other embryonic cracks were found which were associated with surface irregularities arising from a grit-blasting process during manufacturing.

The following additional findings in Ref. 7.11 are noteworthy:

1. After the accident, the left main landing gear cylinder from the same aircraft was found to have two small cracks in the critical area, but these were discovered only after removal of the Cadmium plating.
2. One overhaul agency had found three confirmed cases of cracking of main landing gear cylinders but, due to an oversight, these had not been reported to the manufacturer, the FAA nor the national airworthiness authority.
3. The fracture bore many similarities to those of another cylinder failure which occurred in April of 1995 and which was investigated by the AAIB. A further failure which occurred in China in 1997 also bore many common features.
4. Measures undertaken by the aircraft manufacturer and the FAA following the above two accidents did not prevent this third failure. Assumptions that cylinders would be crack-free following four negative NDE inspections, and after installation of brake hydraulic line restrictors, appear to have been erroneous.
5. The small critical crack size, coupled with an unknown growth rate, made detection of cracking by NDE, before complete failure, extremely difficult using then-specified procedures.
6. A proposed revised inspection procedure and periodicity, should enhance the probability of detecting a crack prior to failure of the cylinder.

7.18.3 Solution

Time will tell whether or not item 6 will have solved this problem.

7.18.4 Lesson

This type of failure can result in a disastrous accident in the future. Landing gear manufacturing, maintenance and inspection procedures are critical to safety.

7.19 The Wrong Hydraulic Pump

7.19.1 Problem

In August of 2001 an Air Transat Airbus 330 (Figure 7.21) ran out of fuel causing both engines to flame out. The crew managed to land the airplane as a glider on the island of Lagos in the Azores.

Figure 7.21 Airbus A330 (Not accident aircraft, Courtesy www.al-airliners.be)

7.19.2 Cause

The airplane had recently received a new starboard engine. The engine was delivered to the airline without a hydraulic pump installed. Airline mechanics had a similar (but not the same) hydraulic pump in stock which did fit on the engine and had the correct pumping capacity. When installed on the engine the hydraulic fluid line from this pump was closer to a fuel line than would have been the case with the proper pump installed. Because of the smaller clearance between the hydraulic line and the adjacent fuel line it became possible for the hydraulic fluid line to vibrate and rub against the fuel line. During this particular flight a large leak occurred and fuel was spilled overboard.

The crew noticed a fuel imbalance at some time during the flight. Since they did not suspect a leak a cross-feed valve was opened which allowed fuel to transfer from one wing to the other to correct the imbalance.

Eventually, the airplane ran out of fuel and both engines flamed out. Luckily the airplane was within gliding distance of the Lagos airport. The landing (without the benefit of flaps and thrust reversers) was severe and all tires blew out. The airplane did come to a stop on the runway and all passengers and crew were evacuated.

7.19.3 Solution

The maintenance practices of this airline were questioned, presumably improved and the airline was fined.

7.19.4 Lesson

When carrying out maintenance on a flight critical item it is not acceptable to deviate from improved procedures. In this instance the maintenance department had the choice to order the proper part (and wait two weeks before returning the airplane to service) or assume that nothing would go wrong when installing the wrong part and return the airplane to revenue service.

7.20 Faulty Structural Repair

7.20.1 Problem

In May of 2002 a China Airlines Boeing 747-200 (Figure 7.22) experienced a structural disintegration at the top of its climb to cruise altitude. All aboard perished.

The in-flight break-up of this 23-year old airplane is believed to have started in the rear fuselage.

Figure 7.22 Boeing 747-200 (Not accident aircraft, Courtesy www.al-airliners.be)

7.20.2 Cause

The following material has been adapted from Ref. 7.12.

Investigators believe that the structural failure originated at cracks underneath a tail strike repair doubler located on the upswept part of the pressurized fuselage. The accident occurred within the inspection interval for that area of the structure. The last detailed inspection was carried out 3.5 years before the accident within the 4 years interval required by Boeing.

Figure 7.23 shows the general location of the tail strike repair doubler at the bottom of the lower rear fuselage. Figure 7.24 shows a close-up of the repair doubler. It apparently covered a fatigue crack from which air leaked from the pressurized fuselage. Streaks emanating from such doublers are often signs of underlying fatigue cracks which can only be visually detected by a periodic bilge inspection.

7.20.3 Solution and Lesson

Structural repairs carried out with the help of external doublers remain a source of hidden fatigue cracks. The periods between scheduled inspections of such areas should probably be shortened, particularly in the case of geriatric airframes.

An important lesson is that if streaks can be seen on the outside of doublers the airplane should probably be grounded for a closer inspection. Streaks tend to be a sign of leakage. Leakage often is caused by undetected fatigue cracks.

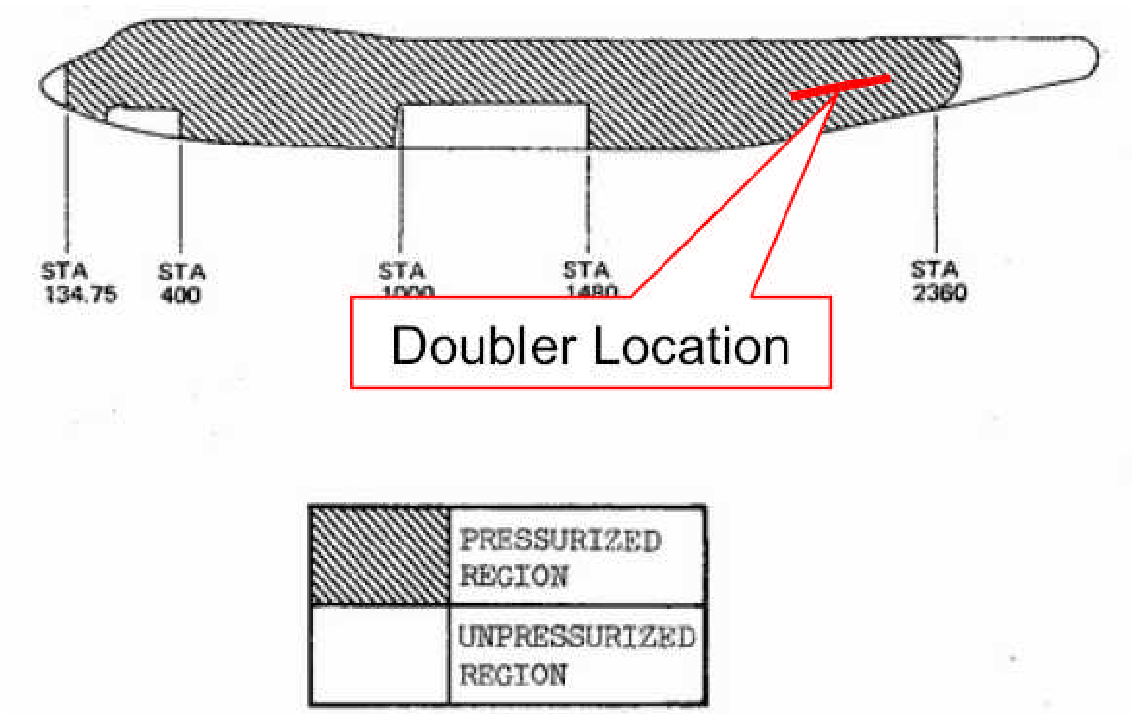

Figure 7.23 Location of Tail Strike Repair Doubler (Courtesy Republic of China Aviation Safety Council)

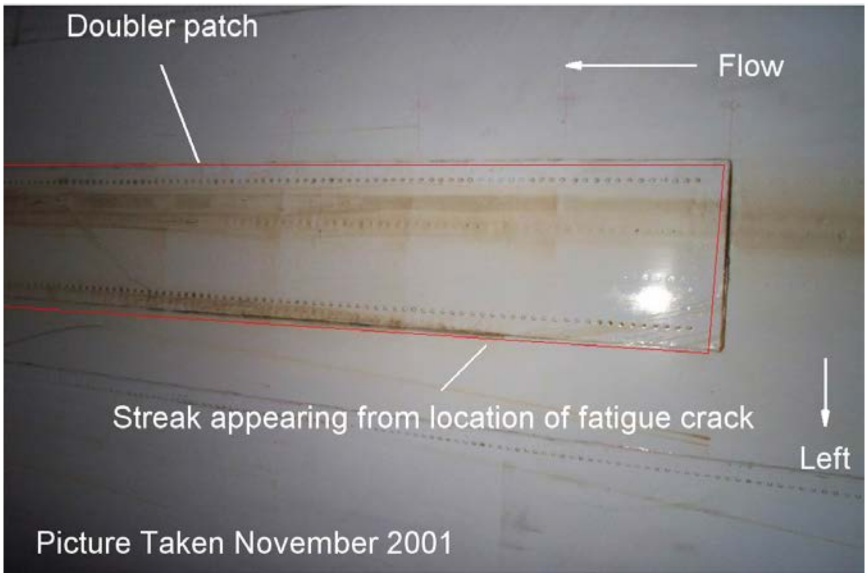

Figure 7.24 Repair Doubler Showing Streaks Emanating from Fatigue Crack (Courtesy Republic of China Aviation Safety Council)

7.21 Fuel Tank Purge Door Left Open

7.21.1 Problem

On June 10, 2004 a British Airways Boeing 777 (Figure 7.25) returned to Heathrow after the crew of another airplane reported they saw a large contrail of smoke coupled with the smell of fuel vapor streaming from the airplane. There were no injuries and there was no reported damage to the airplane.

Figure 7.25 Boeing 777 (Not accident aircraft, Courtesy www.al-airliners.be)

7.21.2 Cause

An engineering inspection following landing revealed that the center fuel tank purge door in the left main landing gear door was missing (Figure 7.26).

The following is quoted from Ref. 7.13: “In maintenance, a hose is attached to the purge hole to flow fresh air into the tank for technicians inside. The door was hanging by a lanyard inside the actual fuel tank. The purge door screws were in a plastic bag attached to the opening as shown in Figure 7.26.

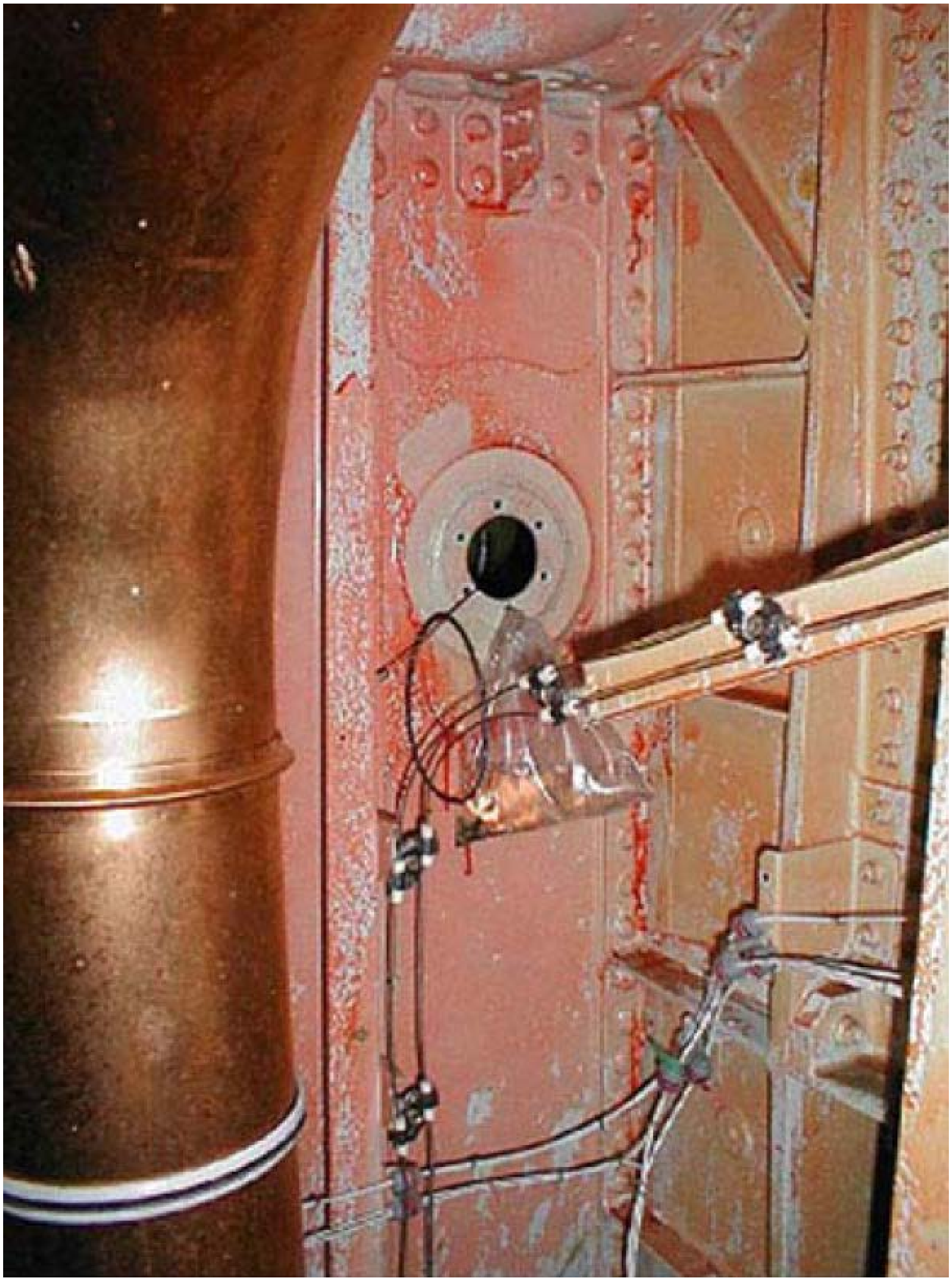

Figure 7.26 View into the Left Main Landing Gear Well (Courtesy AAIB)

The center fuel tank can hold 80 tons of jet fuel. While the BA aircraft's center fuel tank was only just over half full, it held 43 tons. Pitch attitude during initial climb-out was enough to allow the fuel to flow through the opening."

7.21.3 Lesson

1. Human maintenance did it again.

2. Now look at this from a design viewpoint. Suppose this happens again. Also suppose that during a prolonged taxi operation on a hot field the tires are very hot. The gear retracts and one tire explodes. With the purge door hole open, a fuel tank explosion and loss of the aircraft is the probable result.

That is not an acceptable scenario.

Chapter 8

Lessons Drawn from Aerodynamic Design, Configuration Design and Aircraft Sizing

"Aerodynamic Design: if it looks good, it will fly well
Configuration Design: Innovative configurations are hard to sell
Aircraft Sizing: Size airplanes for growth"

Dr. Jan Roskam, 2007

8.1 Introduction

In this chapter a series of problems which arose as a result of aerodynamic design decisions are reviewed. Where applicable, causes and solutions are described and lessons learned are stated. Connections with other areas of design are identified in Appendix A.

8.2 Empennage Changed Due to Insufficient Longitudinal and Directional Stability

8.2.1 Problem

In 1946, after the first flight of the AVRO Tudor 1 (Figure 8.1) the pilots commented that longitudinal and directional stability and control were insufficient (Ref. 8.1, p. 384 and Ref. 8.2, pages 11-12). Note in Figure 8.1 the small vertical tail and the side area of the main landing gear (forward of the c.g.) which does not help directional stability.

Figure 8.1 AVRO TUDOR 1 (Not accident aircraft,
Courtesy Royal Aeronautical Society Library)

8.2.2 Cause

The cause was the ineffective vertical tail size and aspect ratio as well as the horizontal tail which was too small for effective control.

8.2.3 Solution

A very much larger vertical tail with higher aspect ratio and a slightly larger horizontal tail were installed as shown in Figure 8.2.

Figure 8.2 Model of the AVRO Tudor 1 with Modified Vertical Tail
(Courtesy www.collectorsaircraft.com)

8.2.4 Lessons

1. Designers should never release an airplane for first flight without being sure that the basic stability and control levels are acceptable. In this case the pilots, after the first flight, made it clear to engineering that directional stability was not acceptable. A design department should realize that calculations are fine (and necessary) but can be erroneous. Therefore, wind tunnel tests should be performed to verify basic stability and control levels.
2. Reportedly, the AVRO team was in a hurry to get the airplane in the air and therefore this step was skipped. Design management should realize that major modifications to an airplane after first flight are very expensive and very calendar time consuming. It is "penny wise" and "dollar foolish" to believe otherwise.

8.3 Dorsal Fin Suppresses Rudder Lock

8.3.1 Problem

In May of 1949 a Bristol Freighter airplane crashed into the sea. At that time the cause of the crash was not established. All on board were lost. In March of 1950 another Bristol Freighter crashed during a single-engine climb under circumstances similar to the earlier crash (Ref. 8.3, pp 337-338). Figure 8.3 shows an early picture of the Bristol Freighter (note the absence of the dorsal fin).

Figure 8.3 Early Version of the Bristol Freighter (Not accident aircraft, Courtesy John M. Wheatley)

8.3.2 Cause

The cause of the crash was established to be rudder lock followed by structural failure of the vertical tail. A technical description of the rudder lock phenomenon is given on pages 268-270 of Ref. 8.4.

The air flowing over the top of the rear fuselage is prone to form a very thick boundary layer in the vicinity of the vertical tail. This is suggested by the sketch in Figure 8.4.

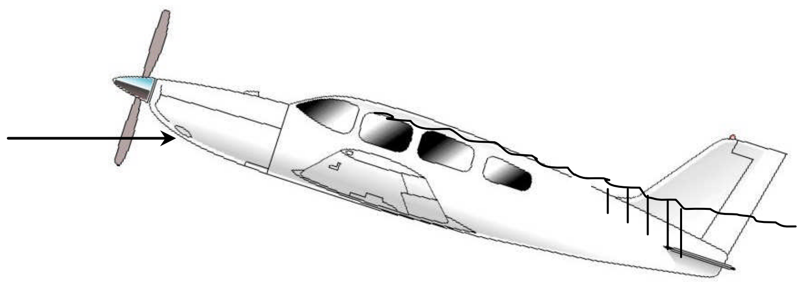

Figure 8.4 Boundary Layer Thickening and Potential Separation over the Aft Fuselage

At a modest angle of attack the flow may even separate. Either way, the bottom of the vertical tail is likely to be in very low energy flow and therefore not effective. This not only reduces directional stability but makes the rudder hinge-moment coefficients nonlinear, causing rudder lock. With the rudder aerodynamically locked against its hard-stop the loads on the vertical tail tend to be very high and can lead to structural failure.

8.3.3 Solution

The problem was eliminated by the addition of a dorsal fin to the airplane. Figure 8.5 shows an example of the Bristol Freighter with the added dorsal fin.

Figure 8.5 Bristol Freighter with Dorsal Fin (Courtesy E. Marmet)

An explanation of why a dorsal fin can solve this problem is found in the shedding of strong leading edge vortices of highly swept, thin leading edge, delta wings. Figure 8.6 shows two sketches:

a. The leading edge vortex system developed by a highly swept, delta wing and

b. The relationship between lift coefficient and angle of attack of such a wing

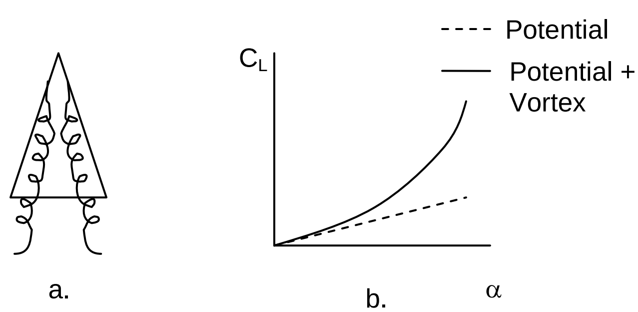

Figure 8.6 Vortex Lift Developed by a Highly Swept, Thin Leading Edge, Delta Wing

Any wing, according to potential flow theory, develops so-called potential lift. If a wing also develops a leading edge vortex system, the external, potential flow moves around this stable vortex (think of this vortex as a small tornado). The vortex can be thought of as increasing the effective camber of the wing thereby enhancing its lift. The vortex component of the lift tends to become stronger with increasing angle of attack.

Next, examine Figure 8.7 which shows a highly swept dorsal fin placed ahead of a vertical tail.

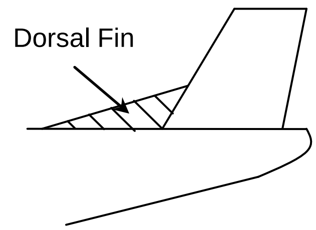

Figure 8.7 Dorsal Fin Interpreted as Half of a Highly Swept Delta Wing

The dorsal fin can be thought of to act as one half of the delta wing in Figure 8.6a.

It should be noted that if the leading edge of the fin is not thin but well rounded, its effectiveness largely disappears.

8.3.4 Lesson

The lesson learned here is that designers should consider adding a dorsal fin to a configuration from day one. If a wind tunnel test is planned, do the test with and without the dorsal to establish its effectiveness.

It is important to repeat the advice that the leading edge radius of the dorsal must be kept as low as practical for the fin to work as advertised.

In a recent wind tunnel test program conducted by DARcorporation on a new business jet design a well-rounded dorsal was used (against our advice) and it proved to be not very effective. When a sharp edged dorsal was substituted it worked. The difference can truly be remarkable.

Examples of airplanes with thin leading edge dorsal fins (also known as dorsals) are: Vickers Vanguard, DeHavilland Canada DHC-7, Fokker F-27, Fokker F-28, Fokker 70, Fokker 100 and the Beechcraft King Air 200.

An interesting historical observation is that the prototype of the Boeing Model 307 Stratoliner (the world's first pressurized piston-propeller transport) crashed in 1938 partly because the rudder locked hard-over during an attempt to recover from an unintentionally induced upset. At the time of this flight test, the rudder-lock potential of this airplane had already been discovered by Boeing engineers in the wind tunnel. Their solution, a dorsal fin (albeit well rounded), had also been tested in the tunnel but not yet fitted to the accident airplane. It was fitted to all 10 production Stratoliners.

It is also interesting to recall that all B-17 bombers of WWII were fitted with these dorsals for the same reason.

8.4 Commonality Lost

8.4.1 Problem

In 1953 Percival Aircraft of Luton, England was working on a jet powered version of the Provost, called the Jet Provost. The basic idea was to create a low cost, jet-powered version of the very successful Provost, standard propeller-driven trainer of the RAF which was in production at that time. Figure 8.8 and Figure 8.9 show the two airplanes.

8.4.2 Solution

To achieve this objective it was decided to utilize a high degree of commonality between the two airplanes.

8.4.3 Lesson

It is evident from Figure 8.8 and Figure 8.9 that commonality between the two airplanes in this case is largely an illusion. As things turned out, only the tail-planes of the airplanes had some commonality. Retaining commonality between two designs with widely differing power-plants and widely differing flight envelopes (altitude and Mach number) often proves impossible. History shows that in most cases a better and lower cost solution is to start with a fresh sheet of paper (or today, with a new CAD file).

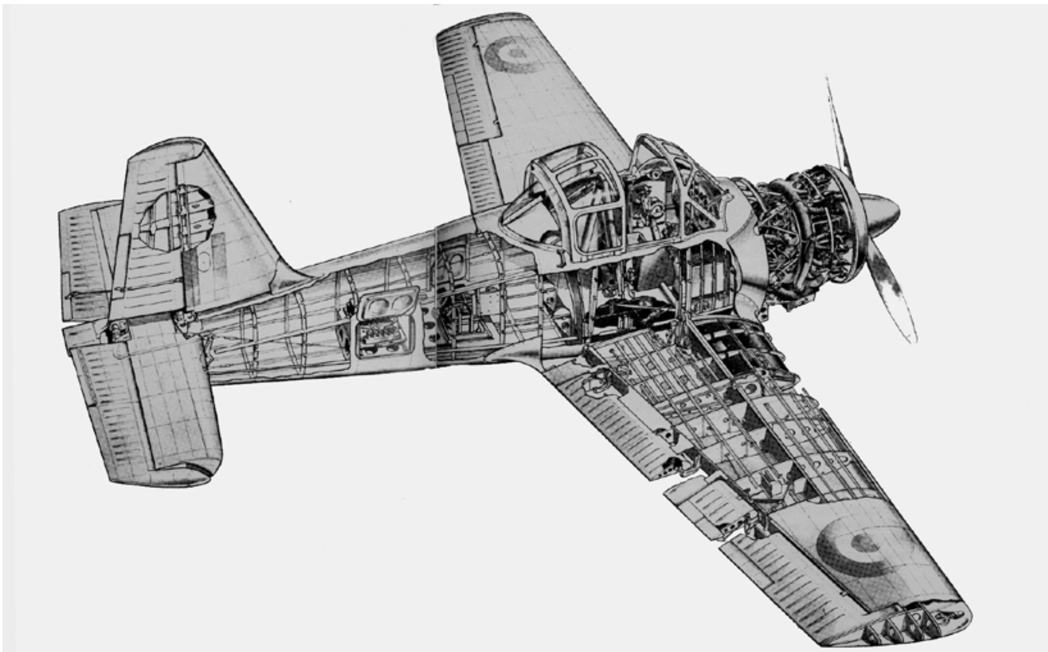

Figure 8.8 Percival Provost (From Ref. 8.5 with Permission from Mrs. B. Silvester)

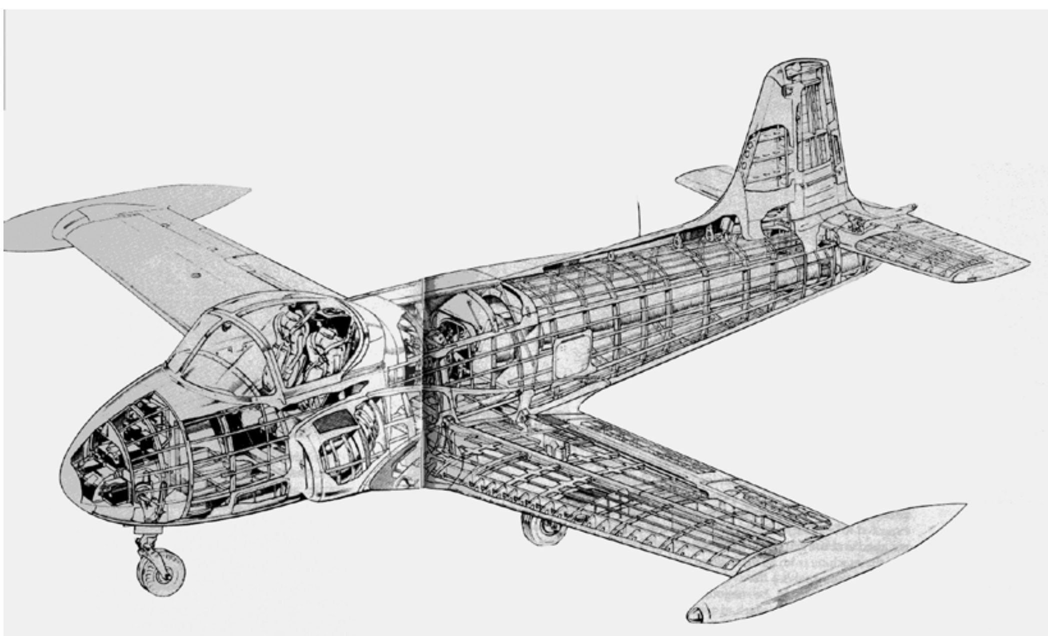

Figure 8.9 Percival Jet-Provost (From Ref. 8.5 with Permission from Mrs. B. Silvester)

8.5 Deep Stall I

8.5.1 Problem

In 1953 a Gloster Javelin (Figure 8.10) was lost during a test flight at aft c.g. The test pilot did not survive (Ref. 8.6, pages 320-321).

Figure 8.10 Gloster Javelin (Not accident aircraft, Courtesy F. Croom)

8.5.2 Cause

The airplane had entered a so-called super-stall condition (stable, slightly nose-up with a very high rate of descent) from which recovery was not possible. In airplanes with a relatively small horizontal T-tail which also has a short moment arm relative to the c.g. it is possible that at high angles of attack the tail becomes enveloped in the low energy wing wake. For all practical purposes control power is then also compromised.

8.5.3 Solution

One solution is to limit the ability of the pilot to achieve angles of attack beyond some critical value. That can be done with a stick-shaker and/or a stick-pusher. See also Section 8.12.3. Another solution is to make the tail larger and/or give the tail a larger span. Moving the tail aft (increasing its moment arm relative to the c.g.) also helps.

The Douglas A4D-1 Skyhawk (Figure 8.11) of the same era had a similar configuration. However, it was a single engine design and its fuselage was much narrower. Also, the tail had a larger moment arm and was in a cruciform rather than a T-tail position. There are no known super-stall incidents with that airplane.

Figure 8.11 Douglas A4D-1 Skyhawk
(Not accident aircraft, Courtesy Royal Aeronautical Society Library)

8.5.4 Lesson

Super-stall or deep stall is a phenomenon that designers should be aware of. Wind tunnel tests conducted to very high angle of attack are the only known way to identify and cure problems.

8.6 Sizing the Cabin Cross Section in a Competitive Environment

8.6.1 Problem

Around 1955, in the race to develop the Douglas DC-8 and the Boeing 707 transports (Figure 8.12 and Figure 8.13), an important design decision was that of the cabin cross section. This decision has an important bearing on cruise drag and empty weight (through wetted area) and on passenger comfort. Airline customer input proved crucial in this case.

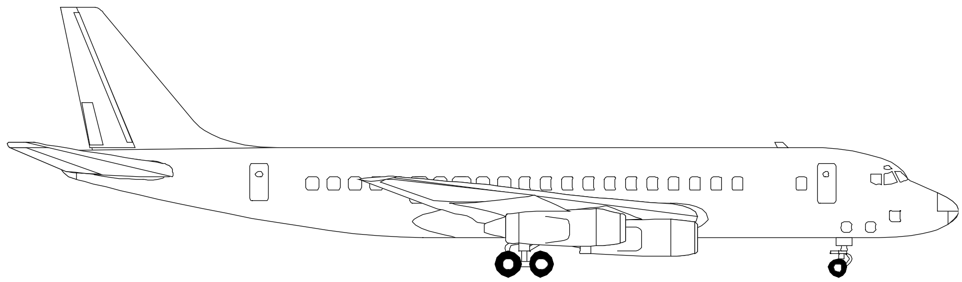

Figure 8.12 Douglas DC-8

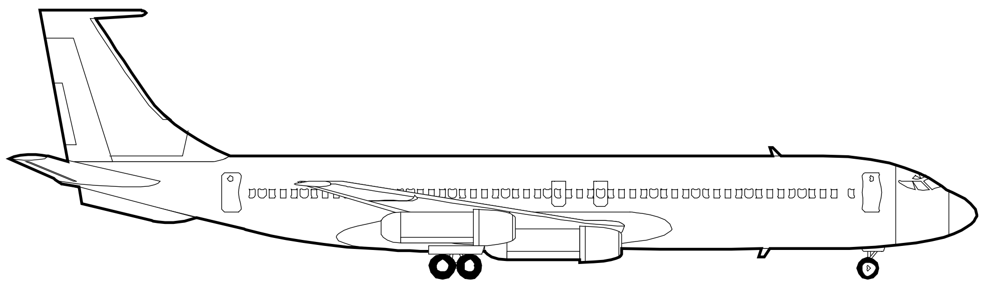

Figure 8.13 Boeing 707-120

Douglas decided on a fuselage diameter of 147 inches knowing that Boeing was using 132 inches on the Dash-80 prototype. The Boeing Stratocruiser also had a 132 inch fuselage diameter. With 147 inches of width the DC-8 could accommodate six-abreast seating while, with 132 inches, only five-abreast seating was possible. This gave Douglas a competitive advantage.

At some point during the design decision making it had become clear to Boeing management that Douglas was committed to the cabin cross section of 147 inches. Once fuselage jig and tool manufacturing has begun it is very expensive to change such a decision. Since Boeing had built the Dash 80 on “soft” tooling as opposed to “hard” (production) tooling, they still had this freedom and it was decided to adopt a cabin cross section of 148.5 inches (Ref. 8.7, Chapter 13). This gave Boeing a fairly significant competitive advantage.

8.6.2 Lesson

When sizing an airplane in a competitive environment management should keep in mind that certain design decisions are very difficult (or expensive) to reverse. It is advisable to retain flexibility in design decision making as long as possible.

8.7 Sizing an Airplane to the Requirements of One Customer

8.7.1 Problem

In 1956 the Vickers VC-10 was designed as a long range, commercial transport to compete against the Boeing 707 and the Douglas DC-8. Figure 8.14 shows the airplane.

Figure 8.14 Vickers VC-10 (Courtesy G. Helmer)

The prime customer, BOAC (British Overseas Airways Corporation, predecessor to British Airways) indicated that it wanted to acquire the 707 for its transatlantic operations. However, the 707 had one problem: it was not sized to take off from relatively short runways under hot day conditions. Such conditions prevailed in those days on the so-called Empire-Route airports such as Kano, Nairobi and Johannesburg. At the insistence of BOAC the VC-10 was sized to do that job.

8.7.2 Cause

It is shown in Ref. 8.8, that sizing an airplane for relatively short runways generally results in a very large wing area. In turn this leads to low values of the cruise lift coefficient and, typically, a poor cruise lift-to-drag ratio. The take-off wing loading of the VC-10 was around 92 lbs/sqft. This compares with 105 lbs/sqft on the 707-320.

As a result of these early design decisions the airplane could not compete very well against the 707 and DC-8 across the North Atlantic which led to a short and unprofitable production program: only 43 were built.

8.7.3 Lesson

Sizing an airplane to the requirements of one customer can (particularly if the customer can order only a small amount of airplanes) results in an uncompetitive airplane. To breakeven on a commercial transport program typically requires that at least 250 airplanes are sold. The reason is that amortization of the non-recurring costs in an airplane program yields a large unit cost increment until the "magic" number of 250 is reached.

8.8 Spin strips

8.8.1 Problem

The first prototype of the T-37 (Figure 8.15) ran into a serious problem: the test pilot had to bail out because the airplane would not recover from a spin (Ref. 8.9, p.81).

Figure 8.15 Cessna T-37 with Nose Mounted Spin Strips (Courtesy NASA)

For a training airplane that was a very serious problem which absolutely had to be solved.

8.8.2 Cause

It was conjectured that the problem was caused by the rudder not being effective in spin recovery. The head of the aerodynamics department, Harry Clements, suggested that a flat strip of aluminum wrapped around the nose of the airplane might break up the flow around the nose at high angle of attack. This could reenergize the air flowing toward the rudder, and just might provide enough rudder control power to allow spin recovery.

8.8.3 Solution

The author was assigned the task of drawing up the strip and seeing it installed on the second prototype that same day. This was done. Our test pilot, Bob Hagan flew the airplane the next day in and out of spins without any trouble.

I remember being asked by our chief engineer, Hank Waring, what it would take to find out why and how this strip worked. I suggested a wind tunnel test but Hank did not want to spend money finding out why a solution which was known to work, really did work.

So, to this day, we can only conjecture why the T-37 nose strip helps in spin recovery.

8.8.4 Lesson

If a solution to a problem works, do not spend any more money finding out why, particularly when you are in a tight budget spot. On the other hand, that way you never find out what the cause of the problem really was.

8.9 Transonic Aerodynamic Center Shift

8.9.1 Problem

The Convair B-58 Hustler (Figure 8.16) was the first operational supersonic bomber in the USAF inventory. The prototype made its first flight in November of 1956.

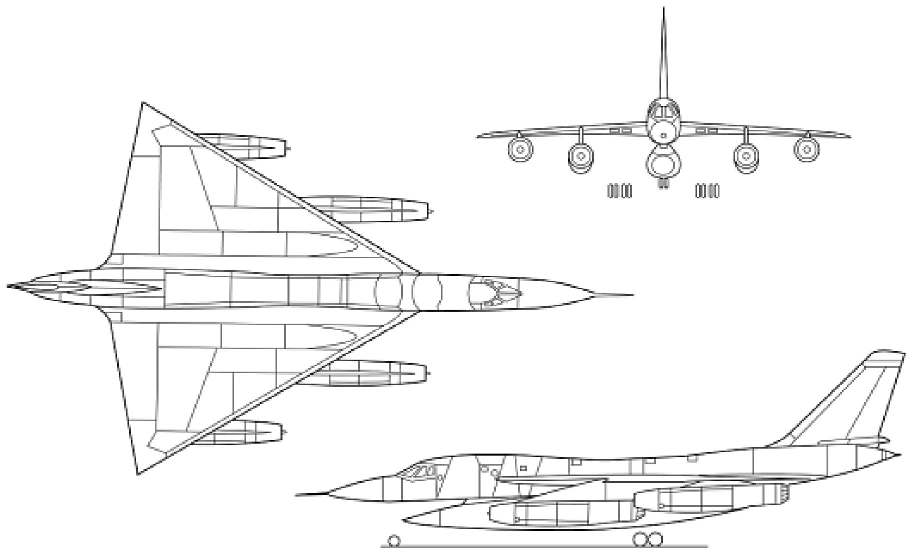

Figure 8.16 Three-view of the Convair B-58 Hustler (Courtesy NASA)

The airplane had a very high accident rate (18 out of 86 delivered) and was withdrawn from service early into its operational life (Ref. 8.10, p.213).

Several accidents occurred as a result of engine flame-out during supersonic (M=2.0) flight, after which the airplane became longitudinally unstable.

8.9.2 Cause

The cause was a rapid forward shift in aerodynamic center as the airplane slows down following engine flame-out at supersonic speed. An explanation of this phenomenon is in order. Figure 8.17 shows the typical aft aerodynamic center shift with increasing Mach number which occurs on most wings.

Line A (Figure 8.17) represents the aft shift in aerodynamic center. The physics behind this trend may be explained by assuming the wing to have a thin and symmetrical airfoil. For thin, symmetrical airfoils the center of pressure and the aerodynamic center are almost coincident. At subsonic speeds such an airfoil tends to have a center of pressure located at the 25% chord point. At supersonic speeds, a system of shocks such as shown in Figure 8.18 will appear.

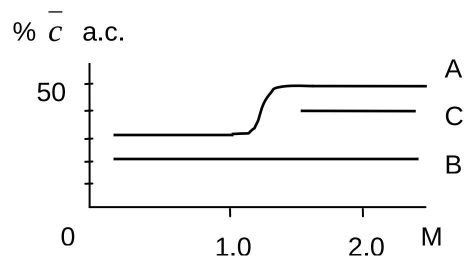

Figure 8.17 Aft Shift of Aerodynamic Center with Mach Number

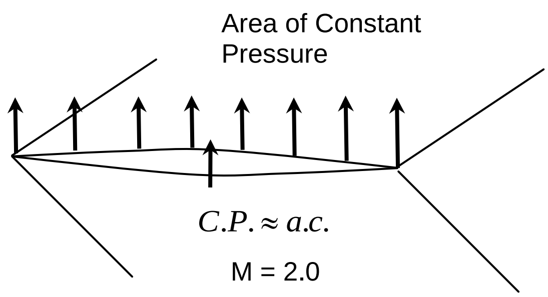

Figure 8.18 Leading and Trailing Edge Shocks at Supersonic Mach Numbers

According to supersonic flow theory (substantiated experimentally) the pressure between two shocks is constant. Therefore, above M=1.0 the center of pressure will be at the 50% chord point. This explains line A in Figure 8.17.

In the era during which the B-58 was built, airplanes had to have inherent static longitudinal stability (fly-by-wire, digital flight controls and automatic feedback to obtain de-facto stability had not yet been developed). That in turn requires the center of gravity of the airplane to be forward of the aerodynamic center. Therefore, at subsonic Mach numbers the c.g. is indicated by line B.

Note that if the c.g. stays at line B the stability level at supersonic Mach numbers is very large. The resulting, large, nose-down pitching moment must be trimmed out. In an airplane with a horizontal tail that trim is accomplished with a download on the tail. The B-58 was tailless. Therefore the trim would have to be obtained by deflecting the elevons, located at the wing trailing edge, downward. That would result in a very unfavorable lift-to-drag ratio and unacceptably low range.

To fix this problem a so-called trim-tank was installed in the rear fuselage. After the airplane has arrived at M=2.0, fuel is pumped into the rear tank which moves the c.g. to correspond to line C in Figure 8.17. That effectively solves the trim problem, but results in the following scenario.

If an engine flames out at M=2.0, a bow shock will appear at the inlet of that engine. This creates a very large increase in drag, which rapidly slows the airplane down. It turned out not to be possible to transfer fuel forward again to prevent the airplane from becoming unstable, which leads to a resulting loss of control and structural break-up.

8.9.3 Solution

Three solutions are possible.

1. Add a horizontal tail
2. Rotate the wingtips down at supersonic speeds. This reduces the static margin and also helps increase directional stability at M=2.0.
3. Allow inherent instability at subsonic speeds and use a digital flight control system to control and stabilize the airplane

Solution 1 would have been practical only if the problem had been foreseen in the early design phase.

Solution 2 would have been practical but was not implemented because of the early service withdrawal. The airplane that was intended to follow the B-58, the North American XB-70A (Figure 8.19) was equipped with fold-down wingtips for this reason.

Figure 8.19 North American XB-70A with Wing Tips Folded (Courtesy Department of Defense)

Solution 3 was not feasible in that era.

8.9.4 Lesson

The problem should have been foreseen in the preliminary design stage. Engineers should always play the what-if game long before a project is committed to hardware.

8.10 Swept Vertical Tail on a Propeller Driven Airplane

8.10.1 Problem

The following material was adapted from Ref. 8.9. In 1957 the Cessna 172 had an unswept vertical tail as shown in Figure 8.20.

Figure 8.20 Cessna 172 with Unswept Vertical Tail (Courtesy Cessna)

Cessna's commercial marketing director felt that sales could be further stimulated by changing the vertical tail to a swept one. His idea met with a lot of resistance from the commercial airplane engineering department. These engineers opined that more weight, more drag, less effectiveness and less performance would be the result. At that time the author was working in the military airplane engineering department on derivatives of the Cessna T-37. Nevertheless, I was asked to spend some time and analyze this swept tail.

8.10.2 Solution

Figure 8.21 shows what happens if one takes an un-swept vertical tail and gradually sweeps the leading edge by rotating about Point A.

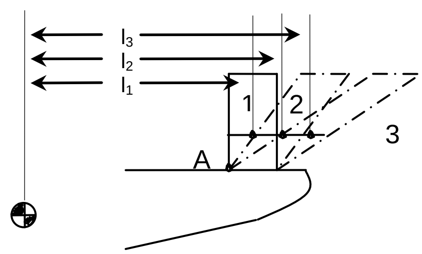

Figure 8.21 Effect of Vertical Tail Sweep Angle on Moment Arm

It is seen that the moment arm (distance from the airplane c.g. to the vertical tail aerodynamic center) of the vertical tail, ℓ_v, increases with increasing sweep angle. This is shown graphically in Figure 8.22a. Note, that the area, S_v, stays constant.

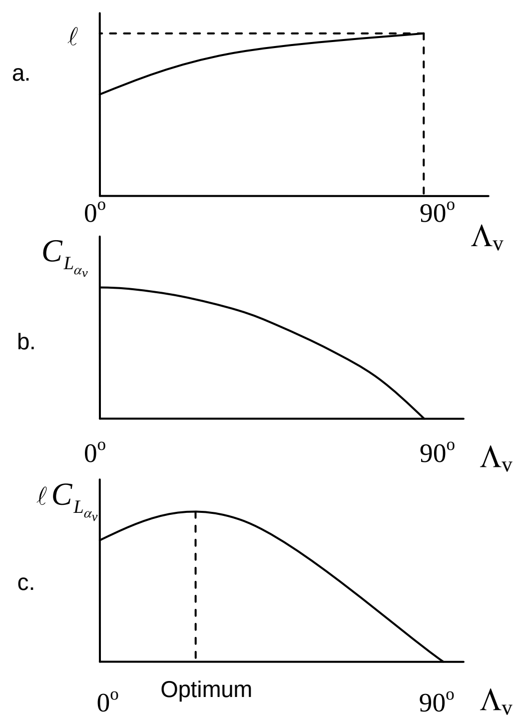

Figure 8.22 Effect of Vertical Tail Sweep Angle on Moment Arm,
Lift-Curve-Slope and their Product

It is well known that the lift-curve-slope of the vertical tail decreases with increasing sweep angle. That effect is shown in Figure 8.22b.

The directional stability and control effectiveness of a vertical tail and rudder are proportional to the product of these two quantities. Figure 8.22c shows that the product has an optimum at a non-zero sweep angle. Therefore, from a vertical tail effectiveness viewpoint, some sweep is favorable. Note that the optimum is fairly flat. That means that a slightly larger than optimum sweep angle would not hurt much.

Next, the matter of structural weight and drag need to be addressed. By increasing the sweep angle, the torque box of the vertical tail would become longer which leads to increased weight. On the other hand, if the un-swept tail has adequate effectiveness and acceptable weight, one could elect to use a smaller tail area to keep the product $S_v \ell_v C_{L_{\alpha_v}}$ the same! That would then tend to reduce the weight and the drag.

It turns out that in the case of the C-172 the sweep angle of the vertical tail was a washout in terms of weight, drag, effectiveness and performance. But with the enhanced "looks" Cessna probably would sell more airplanes.

I briefed the marketing director on my results and he took my data back to the Commercial Division and managed to convince the decision-making hanchos that they should adopt the swept tail. That fall, Cessna proudly announced the "new swept look" for the 172 model. It improved sales by 30% and the swept tail became a standard feature on other Cessna models (Figure 8.23).

Figure 8.23 Cessna 172 with Swept Vertical Tail (Courtesy Cessna)

8.10.3 Lesson

Good looks can really improve airplane sales, particularly if nothing else gets harmed. When analyzing a change in an airplane all factors should be looked at carefully.

8.11 Transonic Drag I

8.11.1 Problem

The Convair 990 (Figure 8.24) was designed to be a jet transport with a cruise Mach number of M=0.91. For 1958 that was a very tall order.

The drag in cruise flight turned out to be so high that the airplane was far from meeting its performance guarantees. The cause had to be identified and a solution had to be found.

Figure 8.24 Convair 990 Coronado (Courtesy NASA)

8.11.2 Cause

It turned out that in cruise flight of the prototype there was extensive flow separation from the wing trailing edge due to large areas of supersonic flow over the inboard wing. The area of supersonic flow had to be eliminated.

8.11.3 Solution

With the help of extensive wind tunnel testing at NASA the solution which was evolved was the adaptation of so-called "Küchemann Carrots" (Ref. 8.10, pages 217-218). These are also referred to as anti-shock bodies. They basically provide local area ruling to prevent the flow from going supersonic too soon. Figure 8.24 shows the airplane equipped with these trailing edge anti-shock bodies. It turns out that with these anti-shock bodies in place the "local area ruling" of the wing is significantly improved.

8.11.4 Lesson

This type of problem should never have occurred in the first place. By carrying out a wind tunnel test program before arriving at a final configuration the high drag at M=0.91 would have been found. Solutions are a lot cheaper to develop on a wind tunnel model than on a flying prototype.

Of course, in modern days it is possible to predict this problem by using CFD early in the aerodynamic design phase of the wing.

8.12 Deep Stall II

8.12.1 Problem

Several airplanes with fuselage mounted nacelles and T-tails have been known to experience the so-called "deep-stall" problem. In some airplanes this has led to an uncontrollable situation and a crash. An early example of this was the BAC-111 shown in Figure 8.25. The airplane experienced two crashes (one with fatal results), the first one in 1963.

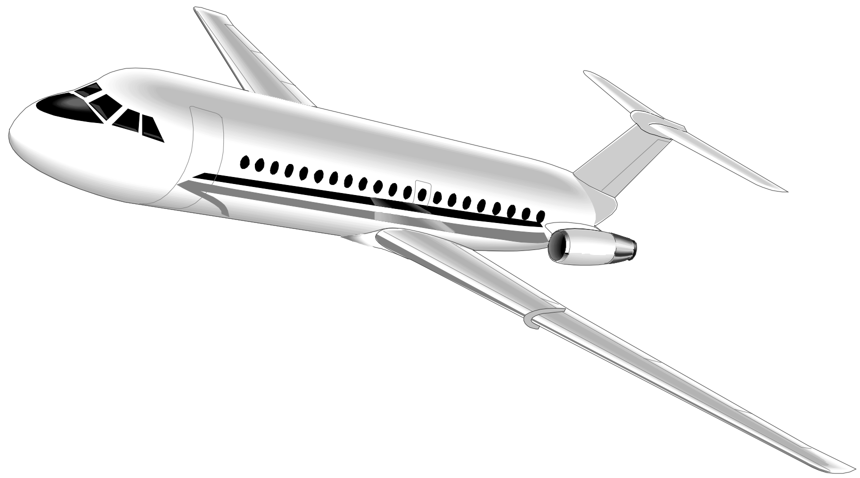

Figure 8.25 BAC-111

8.12.2 Cause

Figure 8.26 shows a generic sketch of a trim diagram for such a configuration. For a detailed derivation and explanation of airplane trim diagrams, see Ref. 8.11, Chapter 4.

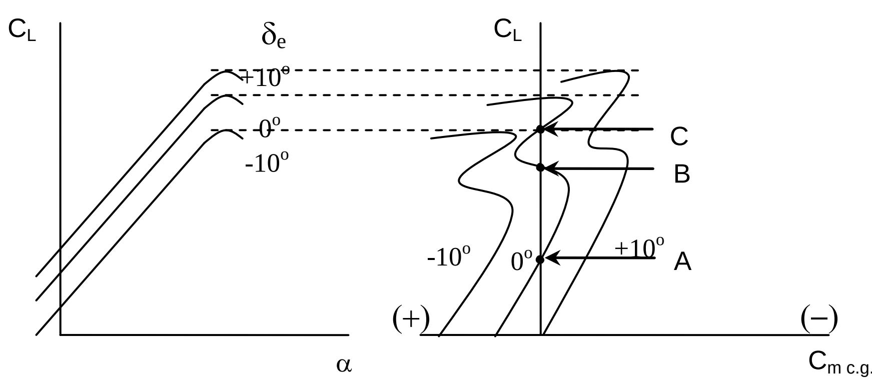

Figure 8.26 Generic Trim Diagram for a Configuration with Aft Mounted Nacelles and a T-Tail

For a given c.g. location, Point A represents a stable trim point with adequate control power around the trim point. At Point B the airplane has become unstable. The physical reason for this is separated flow from the nacelle enveloping part or most of the horizontal tail. This is illustrated in Figure 8.27.

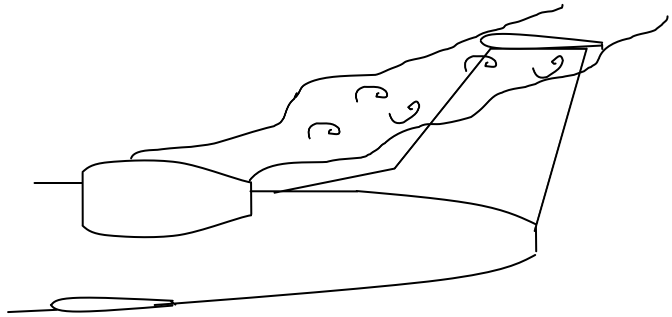

Figure 8.27 Separated Flow from the Nacelle Envelops the Horizontal Tail

The airplane therefore acts as if most of the horizontal tail is not present. Between Point B and Point C the nacelle wake has moved to the left and the tail is again in healthy flow. Point C represents the so-called “deep-stall” trim point. The airplane is stable and trimmed. Beyond Point C the separated wake from the wing has enveloped the horizontal tail and the airplane again becomes unstable.

A significant corollary problem is that longitudinal control power in the region of points B and C tends to be significantly reduced. This is particularly true if the nacelle span is about the same as the horizontal tail span. By making the tail-span significantly larger than the nacelles-span some control power and some stability can be retained. This is illustrated in Figure 8.28.

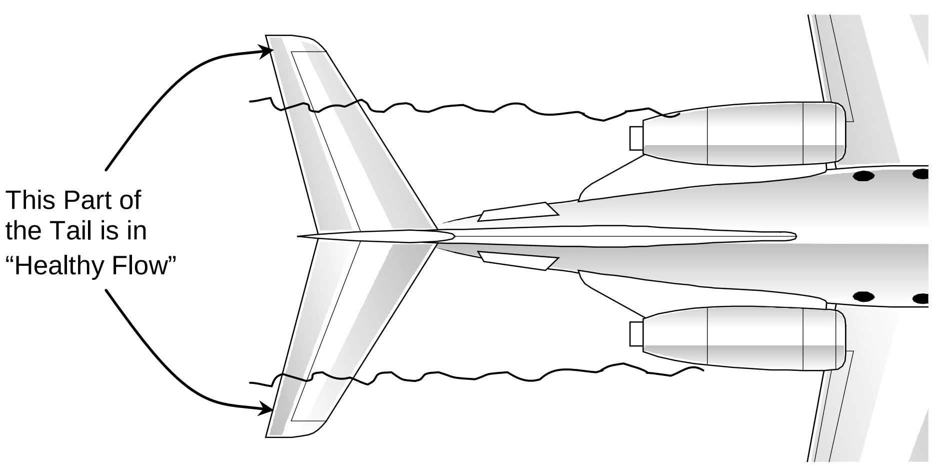

Figure 8.28 Horizontal Tail Span should be Larger than Nacelle Span in a T-Tail Configuration

This principle was applied by Boeing during the wind tunnel development of the 727 airplane (See Figure 8.29).

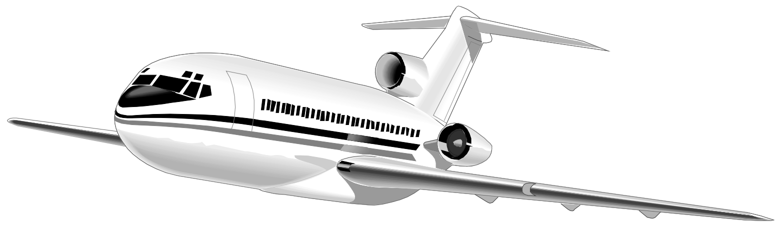

Figure 8.29 Boeing 727

Refs. 8.12, 8.13 and 8.14 contain more information on the aerodynamic behavior at high angles of attack of T-tail configurations.

8.12.3 Solution

There are several solutions to “solve” the “deep stall” problem. The two most frequently used are:

1. Stick shakers and stick pushers

In this solution the pilot is warned by a stick shaker whenever the angle of attack comes close to that at Point B. If a pilot was to ignore this warning and continues to pull the cockpit controls aft, the stick pusher automatically moves the control column forward.
Stick shakers and pushers have been used on airplanes such as the BAC-11, Boeing 727, Douglas DC-9, Boeing MD-80-series and Learjet Models 23, 24, 25, 36 and 55. These systems must be designed with at least dual redundancy.

A consequence of using this solution is that the field-length performance of the airplane gets penalized. Since the limiting angle of attack is below the aerodynamic stall of the wing, approach and lift-off speeds are both higher which results in longer runway requirements. This can make a significant competitive difference.

2. Ventral fins

Figure 8.30 shows an illustration of ventral fins. Note that these fins are highly swept. They also should have sharp leading edges. Aerodynamically they behave just like the dorsal fins explained in Section 8.2.

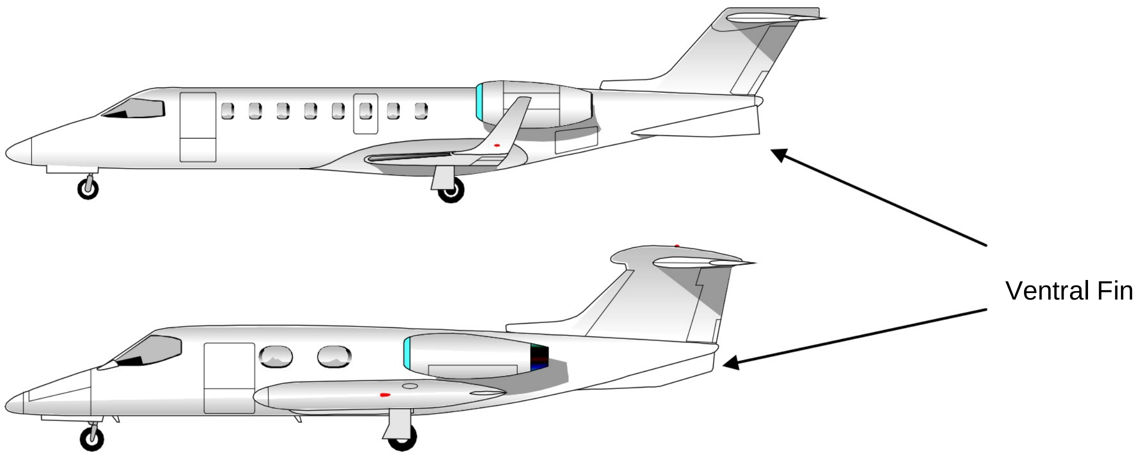

Figure 8.30 Example of a Ventral Fin Installation

The ventral fins are positioned on the airplane so that in cruise they do not contribute any lift. They obviously do add friction drag. Because in cruise no lift is generated by the fin, its longitudinal stability input around the cruise condition is negligible.

At high angles of attack when the tail input to stability falters the ventral fins develop strong vortex lift. Therefore, with the horizontal tail blanketed by either the nacelle or the wing, the ventral fins are in healthy flow and contribute to static longitudinal stability. With proper fin sizing it is possible to eliminate the pitch-up tendencies altogether and thereby certify the airplane without a stick pusher or shaker. This significantly improves the field-length.

It is left to the reader to explain why these ventral fins, when canted about 45 degrees, also contribute to directional stability at high angles of attack.

The author was involved in the development of these fins on the Piaggio P-180 Avanti (Figure 8.31).

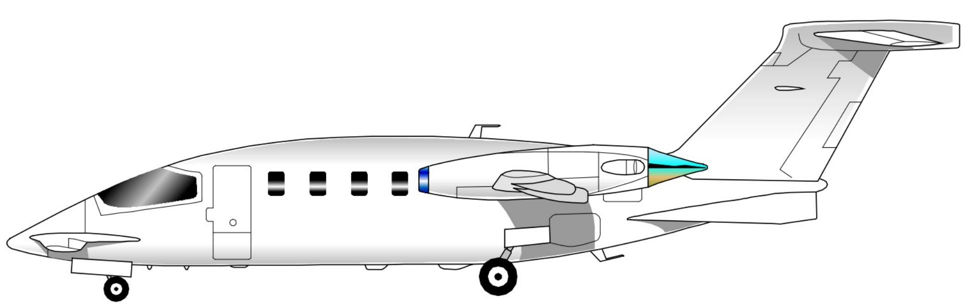

Figure 8.31 Piaggio P-180 Avanti

At that time Gates-Learjet Corp. and Piaggio were cooperating in the development of the P-180. That is the reason why Learjet Models 31, 45 and 60 also employ these fins.

Note: in principle, it is possible to give these fins a variable incidence and thereby also use them for control power. This has not yet been tried to the author's knowledge.

8.12.4 Lessons

1. Deep stall problems can and should be identified in great detail during early wind tunnel testing. These tests should be carried out to about 35 - 45 degrees of angle of attack.

2. Designers should carefully weigh using the systems solution 1 as opposed to an aerodynamic solution 2.

Two advantages of the aerodynamic solution are:

- system failures and/or redundancies are eliminated
- field-length performance is improved.

A disadvantage of the aerodynamic solution is slightly increased cruise drag (and thus fuel consumption). Also, there are people who simply do not like the looks of ventral fins.

8.13 Snaking Oscillation Due to Local Directional Instability

8.13.1 Problem

In 1965, during several test flights at high altitude the XB-70A (Figure 8.32) was observed to have a directional "snaking oscillation".

Figure 8.32 North American XB-70A (Courtesy NASA)

The amplitudes of the oscillation were small but noticeably annoying in the cockpit which is located far forward of the center of gravity. The test pilots described the oscillation as a typical "limit cycle" oscillation. The problem had to be fixed.

8.13.2 Cause

Figure 8.33 shows a plot (without scale) of yawing moment coefficient versus sideslip angle. The line drawn through the measured points (from a wind tunnel test) suggests a positive value for directional stability within the range of sideslip angles tested.

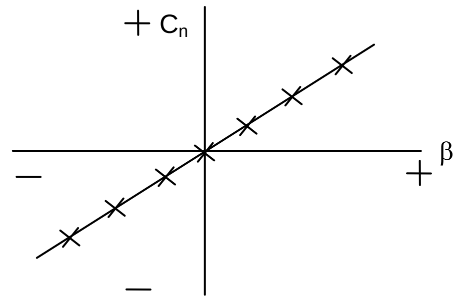

Figure 8.33 Yawing Moment Coefficient versus Sideslip Angle Taken at Large Increments

The data points taken were three degrees apart. Using the "slope-through-zero" the snaking oscillation was not predicted. It was decided to repeat the wind tunnel test but with smaller increments of sideslip. Figure 8.34 shows the result: the airplane had a mild directional instability within +/- three degrees of sideslip and became stable beyond.

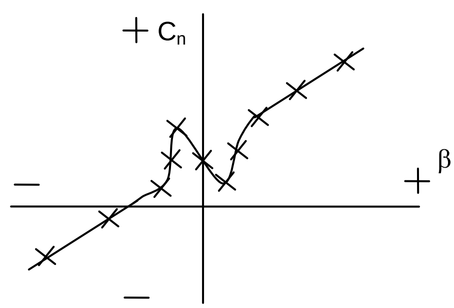

Figure 8.34 Yawing Moment Coefficient versus Sideslip Angle Taken at Small Increments

That explained the limit cycle behavior in flight. The reason for the nonlinear behavior was found to be vortex shedding from the canards. For a small positive sideslip angle, the tip vortex from the left canard was found to blanket the left vertical tail. For negative sideslip the same thing happened on the right. Figure 8.35 shows what happened.

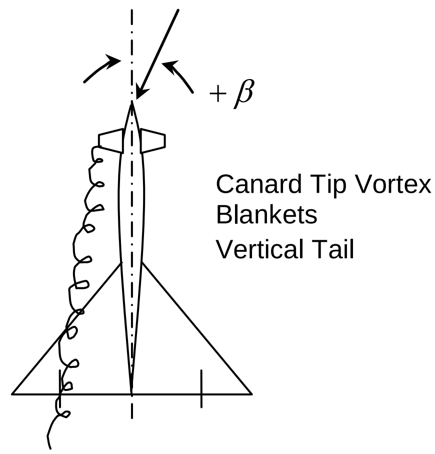

Figure 8.35 Canard Vortex Blankets the Vertical Tail at a Small Sideslip Angle

8.13.3 Solution

A solution was not implemented because of program cancellation (for unrelated reasons).

A potential solution might have been to move the vertical fins closer together or farther apart to miss a shed vortex from the canard tips which blanketed one of the fins. Another solution might have been to make the existing fins larger.

8.13.4 Lesson

When planning a wind tunnel test on a novel configuration it is not a good idea to save money by measuring data at large intervals of sideslip or angle of attack. Doing so can mask existing nonlinearities. Invariably these show up later in flight test. Solving problems in flight test is much more expensive than adding more data points in a wind tunnel test program.

8.14 Aileron Reversal due to Tail Interference

8.14.1 Problem

In 1966 Boeing was involved in various advanced fighter-bomber design studies. Designing adequate lateral control power into highly swept wings is always a problem for two reasons:

- the potential exists for aero-elastically induced aileron reversal. That is why in many subsonic transports the outboard ailerons are locked in place during cruising flight.

- at very high sweep angles there is significant outboard flow in the general direction of the hinge-line of a conventional aileron thereby negating its effectiveness

In England, English Electric was successful with a special wing tip aileron on a 60 degree swept wing installed on the P1A/B Lightning (Figure 8.36).

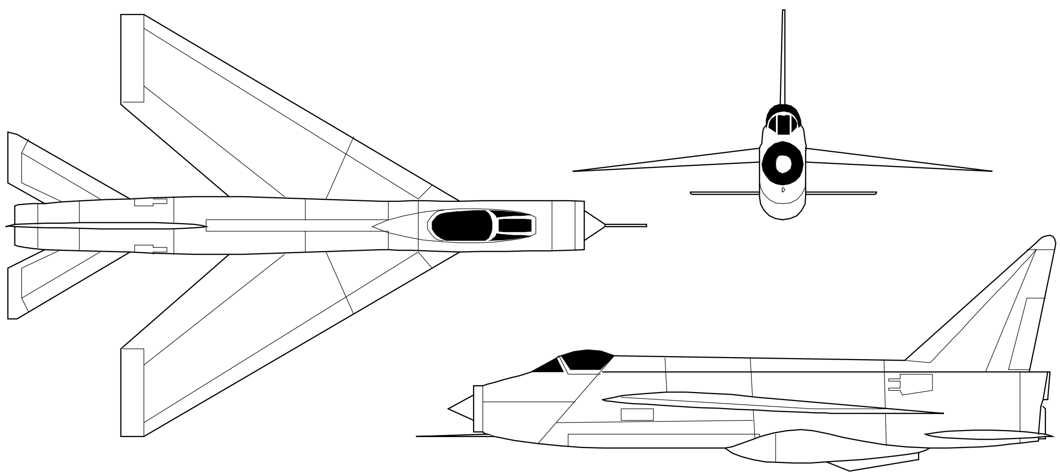

Figure 8.36 Three-view of the English Electric Lightning P1A Lightning

Note the wing tip ailerons. They are located so that the aileron span "straddles" the wing elastic axis. This minimizes any aero-elastic effects. Also, note that the outboard flow due to the high wing sweep is more or less perpendicular to the aileron hinge-line in the case of Figure 8.36.

Finally, note the fact that the inboard aileron station is slightly outboard of the horizontal tail span.

Because of the success of the P1A/B aileron configuration it was decided to emulate this in one of the Boeing design studies. Figure 8.37 shows a sketch of one of those studied.

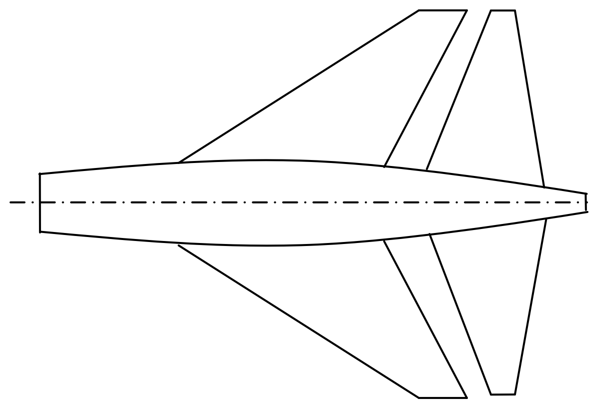

Figure 8.37 Top-view Sketch of a Fighter Configuration

When a wind tunnel test was run to check the aileron effectiveness it was found that the ailerons produced zero rolling moment at any deflection.

8.14.2 Cause

The reason why zero rolling effectiveness was found is evident from Figure 8.38.

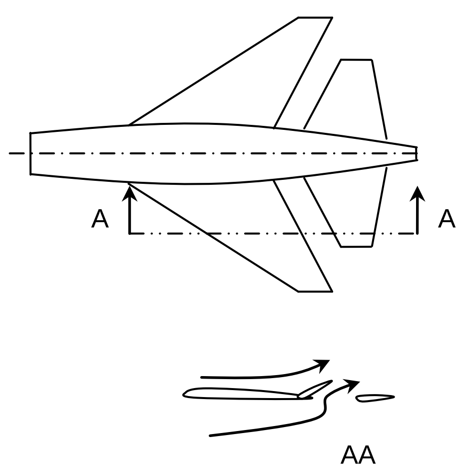

Figure 8.38 Cross Section through Wing-Aileron-Horizontal-Tail

Any trailing edge up deflection of the aileron causes an up-flow at the horizontal tail. As a result, the down-load on the wing is cancelled by an upload on the tail and no net rolling moment results.

8.14.3 Solution

The horizontal tail span should be kept inside the inboard aileron station.

8.14.4 Lesson

The finding in the wind tunnel was entirely predictable with a little common sense. It would help if aerodynamic design engineers attempt a flow visualization (all it takes is to produce a sketch as in Figure 8.38) before committing to an expensive test.

8.15 From Vatlit to Avanti

8.15.1 Problem

The following material was adapted from Ref. 8.9. During 1976 the author took a sabbatical leave from K.U. and spent half a year at The Delft University of Technology in The Netherlands. While there I had the opportunity to work with a large group of Dutch students who wanted to design an advanced general aviation airplane under my supervision. I laid down the design specifications of this airplane. It had to be a very fast, twin turboprop, tractor, high wing airplane aimed at the Beechcraft King Air Market. The airplane was to have a cruise speed of around 400 kts, at 40,000 ft altitude. It also had to be lighter and quieter than the King Air.

8.15.2 Solution

After many trade studies and after trying various configurations the airplane ended up with supercritical, slightly swept wing, with winglets. It became the known as the VATLIT, for Very Advanced Technology Light Twin (Figure 8.39).

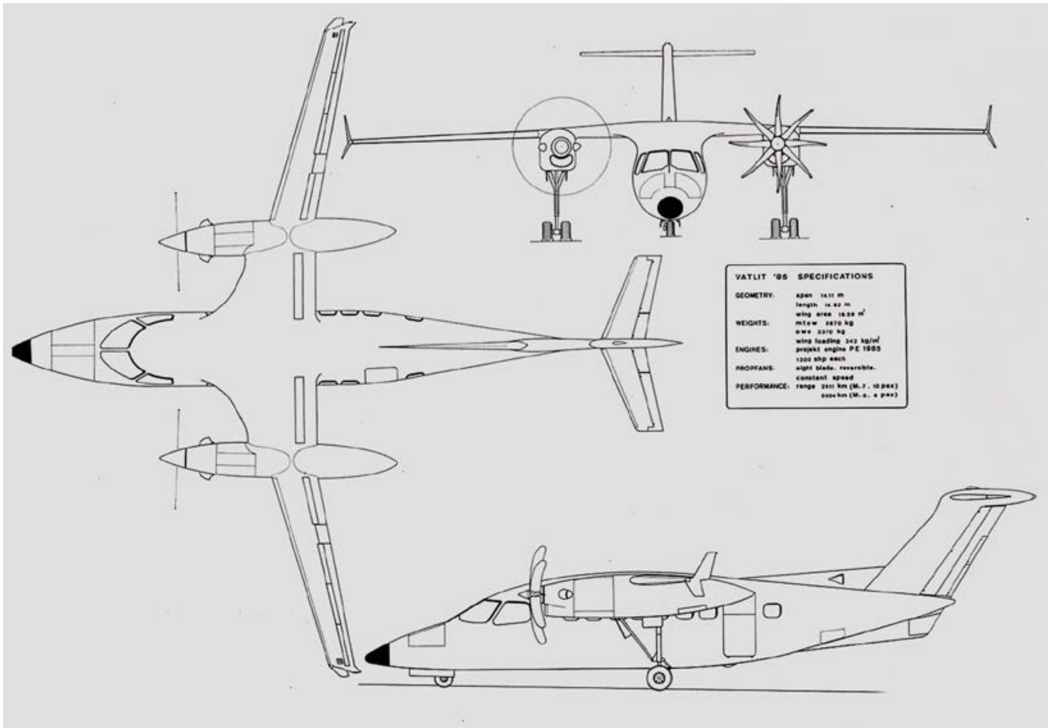

Figure 8.39 The VATLIT (Very Advanced Light Twin)

Although the VATLIT was never built, it had a significant consequence. In 1978, I was invited by Dr. Piaggio, CEO and owner of Piaggio Aircraft Company, to come to his plant for some discussions. It turned out that the VATLIT project had come to the attention of Dr. Piaggio whose design team had been working on a very similar design specification.

My first of many visits to Piaggio resulted in a lot of consulting engineering work on the Piaggio P-180 Avanti (Figure 8.40). The Avanti configuration had been evolved by Piaggio's Director of Engineering, Dr. Mazzoni and his design team. Their big question to me was: which is the better approach: VATLIT or P-180? It did not take me and my own design team in the USA long to determine that the Piaggio approach was the superior one. One very clever aspect of their three-surface approach was the structural and aerodynamic synergism that was achieved:

- the wing torque box and the main landing gear are both attached to the rear pressure bulkhead. This saves a mainframe in the fuselage and therefore quite a bit of empty weight.
- The wing torque box passes through the fuselage as a mid-wing. Mid wings have considerably lower interference drag than either high wings or low wings.

- With a three-surface configuration one can always achieve trim with all surfaces lifting upward.
- The maximum trimmable lift coefficient of a three-surface configuration is higher than that of a conventional or a pure canard configuration.

Figure 8.40 Piaggio P-180 Avanti (Courtesy of Piaggio)

8.15.3 Lesson

Bold configuration thinking can sometimes result in a dramatic improvement of airplane performance. The P-180 as being produced in Italy in 2007 is the world's fastest commercially certified turboprop. It is faster than some business jets, has a quieter, stand-up cabin and uses 25% less fuel

8.16 Horizontal Tail Sizing I

8.16.1 Problem

In 1978, when the first F-18 fighter (Figure 8.41) was flight tested at Patuxent River, MD, it became evident that the airplane would not rotate at the predicted take-off rotation speed. This made the field performance of the airplane for land based applications unacceptable.

Figure 8.41 Early Version of the McDonnell-Douglas F-18, Hornet (Courtesy Boeing)

8.16.2 Cause

The problem was traced to an error in the calculation of aerodynamic forces in ground effect. This is particularly severe in case of a low placed horizontal stabilizer. As a result there was insufficient tail down-load capability to effect rotation at the predicted take-off rotation speed.

8.16.3 Solution

The problem was fixed by toe-in of the rudders. A squat-switch on the main gear biases the rudders to deflect inward while on the ground. This creates enough positive pressure over the aft fuselage to help effect take-off rotation.

This fix, although impressive, came at a price. All flight control software had to be re-validated. Also, the squat-switches and their added controls logic represented additional system complexity.

8.16.4 Lesson

Engineers should not forget about ground effect.

8.17 Horizontal Tail Sizing II

8.17.1 Problem

The following material was adapted from Ref. 8.9. One day in 1977, the author received a call from Ron Neal, the vice president of engineering for Canadair in Montreal. His job was to get the new Challenger 600 business jet program going. Figure 8.42 shows that airplane on its first take-off.

Figure 8.42 Canadair Model 600, Challenger (Courtesy Canadair)

Ron told me that his engineering department was trying to convince him that the horizontal tail of the Challenger was too small by about 30%. The reason given was that, according to their engineering calculations, the airplane would not be able to rotate within the advertised field-length constraints. Ron felt that the tail was probably sized properly and that a mistake had been made somewhere, but, he did not know where. So he asked what I might be able to do to help.

8.17.2 Solution

I suggested that he send me a copy of their tail-sizing report and include the work done on the take-off rotation. When the report arrived it contained a side-view of the airplane with all forces and moments which act on the airplane drawn in. Figure 8.43 shows what such a side-view drawing should be like (albeit for a different airplane).

The problem with the drawing in their report was that the so-called Newtonian reaction force (airplane mass times forward acceleration), which acts through the c.g. and helps in the rotation process, had been left off. I checked their equations of motion and, sure enough, this force was absent. I called Ron and suggested that upon putting that force into the equation it would be my guess that the tail was indeed adequately sized. This turned out to be the case.

A mathematical discussion and a method for solving the take-off rotation problem can be found in Ref. 8.11, pages 288-292.

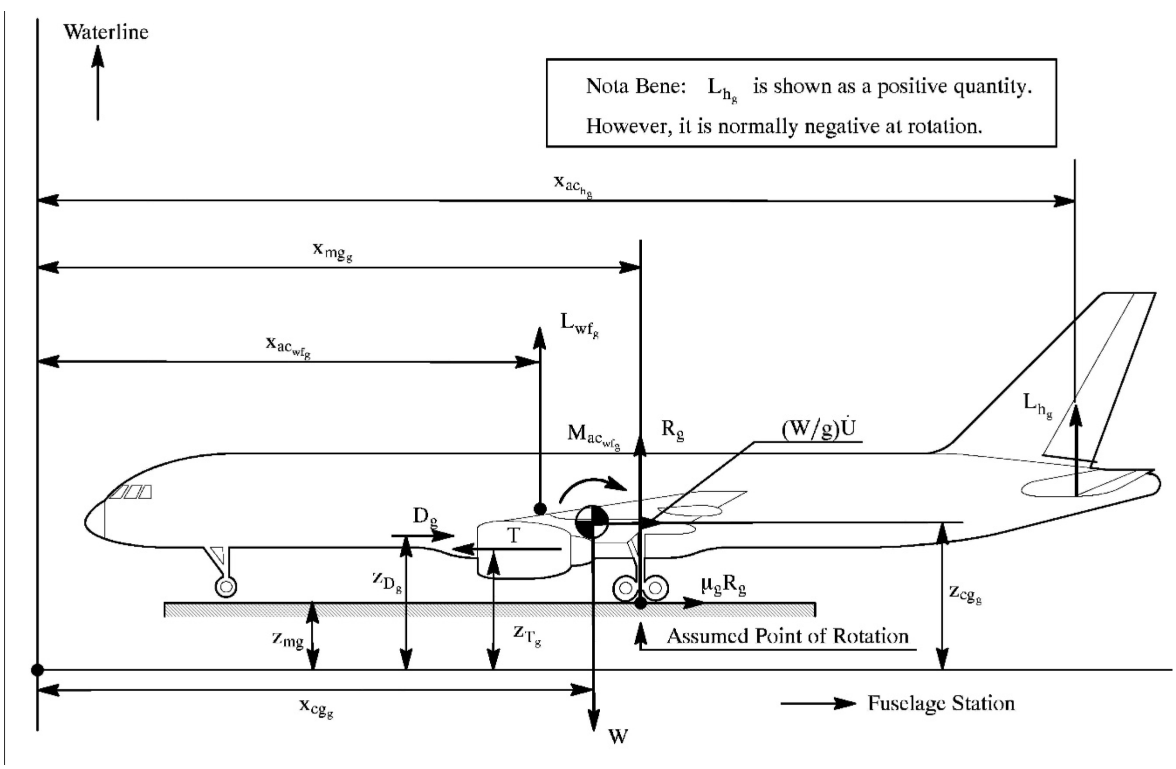

Figure 8.43 Forces and Moments Acting on an Airplane at Initiation of Take-off Rotation

8.17.3 Lesson

Engineers should understand Newton's Laws before tackling problems relating to airplane dynamics. Also this: note the subscript "g" associated with all aerodynamic forces and moments in Figure 8.43. This means that those forces and moments must be evaluated in the presence of the ground ("ground effect"). See also Section 8.16.

8.18 The XFV-12

8.18.1 Problem and Cause

Many years ago the North American Columbus Division conceived a new idea for a supersonic, VSTOL fighter, the XFV-112. Figure 8.44 shows the concept.

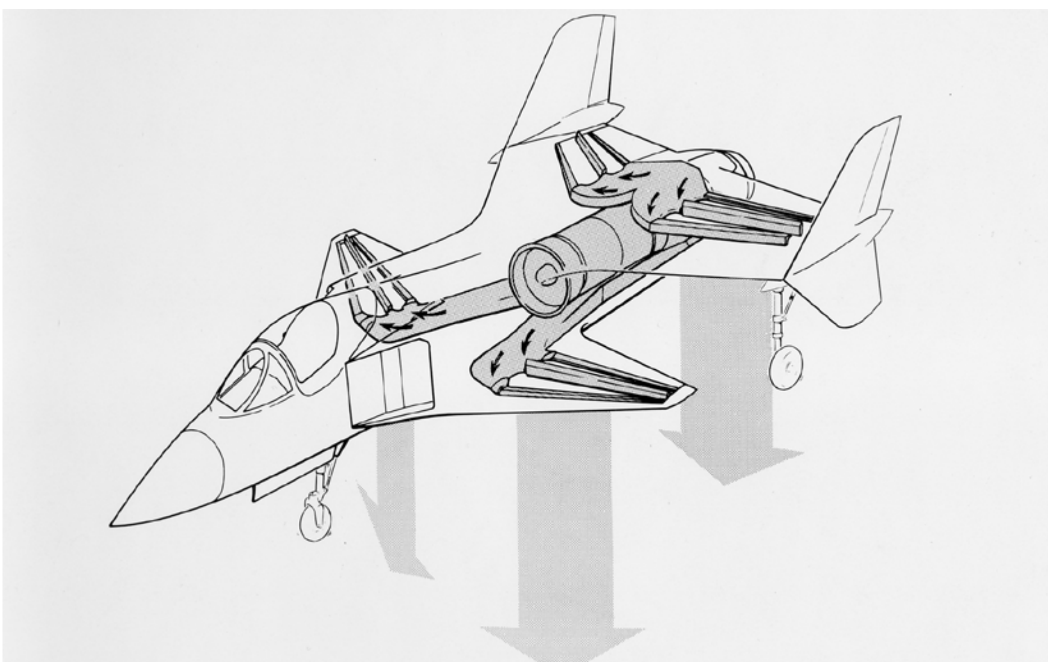

Figure 8.44 Conceptual Sketch of the North American XFV-12
(Courtesy North American)

A contract was received for the construction and flight testing of two prototypes. The prototype was to be evaluated in 1977.

It is of interest to observe that there is only one engine. In the hovering mode the vertical acceleration capability at maximum take-off weight should be at least 1.2g. That means that the ejectors in the wings and tails must generate a large proportion of the required vertical force. It turned out during tethering tests that the system was not able to do that.

Also, in transitioning from the hovering mode to the flight mode, it is well to remember that the air which is drawn into the louvered wing and canard has all of its forward momentum stopped. This causes a rather large momentum drag. To accelerate to forward flight, the total installed thrust (from the vectoring louvers and from the engine inlet/exhaust system) must be able to overcome the total drag and provide a reasonable forward acceleration.

As it turned out, the system was not able to do this either. The project was cancelled. Figure 8.45 shows one of two prototypes which were built.

Figure 8.45 Prototype of the XFV-12 (Courtesy San Diego Aerospace Museum)

8.18.2 Lesson

Momentum drag can be significant. This is predictable and should not be forgotten.

8.19 Do Forward Swept Wings Make Sense?

8.19.1 Problem

The following material was adapted from Ref. 8.9. In 1943, during WWII, the German Luftwaffe flew a four-engine, 35-degree, forward swept wing, jet bomber named the Junkers Model 287. Figure 8.46 shows a three-view of this airplane.

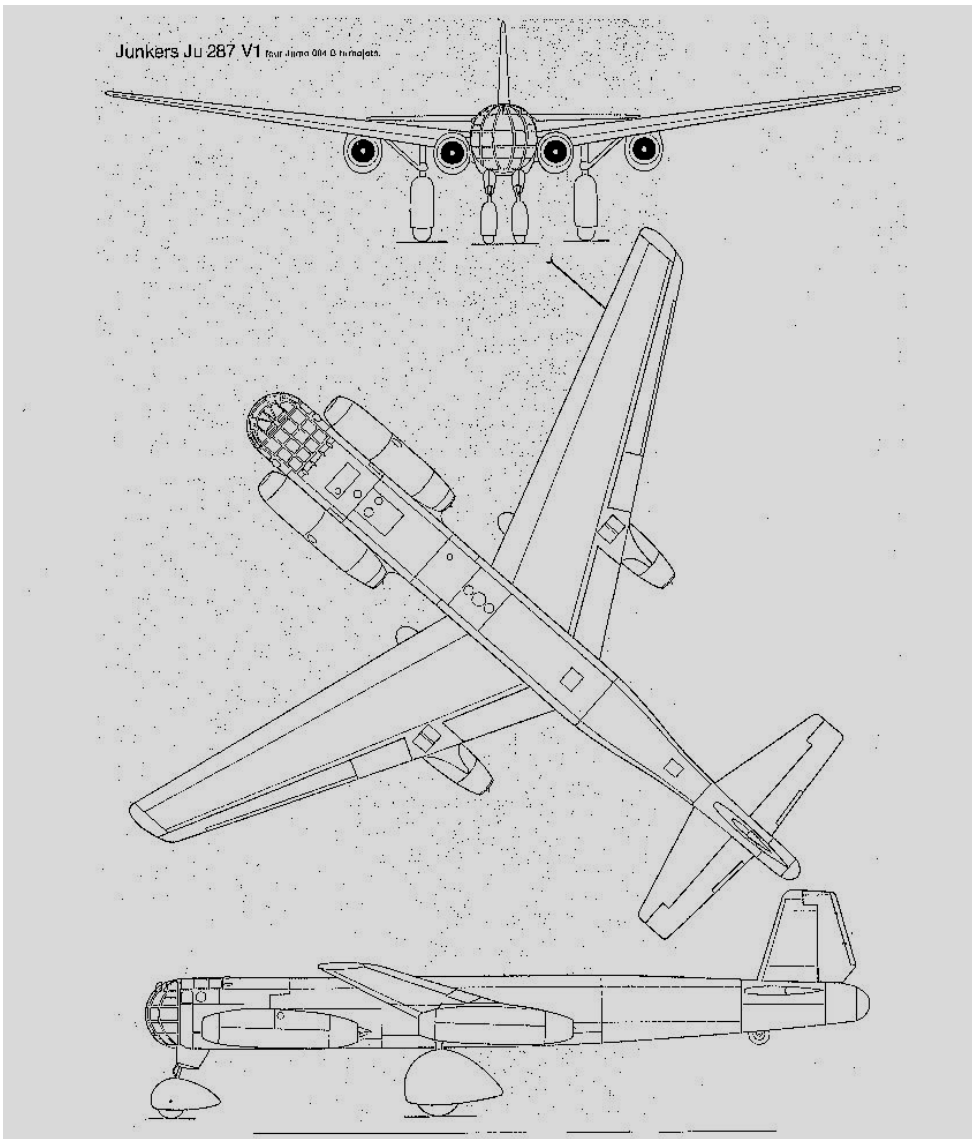

Figure 8.46 Three-view of the Junkers 287 (from: Jet Planes of the Third Reich; Smith & Creek; with Permission from Monogram Aviation Publications; Sturbridge, MA, USA)

Luckily for the Allied war effort, the Ju-287 never achieved operational status although it was extensively test flown.

Following the German surrender in 1945, several US manufacturers became familiar with the Ju-287 technology. Shortly after WWII it became clear that the Soviet Union had sinister designs on Europe. As coined by Winston Churchill, an Iron Curtain descended over Eastern Europe. It became necessary for the USA to start re-arming.

Consequently, the USAF received many design proposals from aircraft manufacturers for forward swept wing fighters and bombers. The phenomenon of structural wing divergence was known to be a potential problem. It was feared that to overcome the divergence issue, large increases in wing weight would be inevitable. To allow for a rational evaluation of the proposed weights for these new airplanes, the USAF enlisted the help of NACA Langley. At Langley, F.W. Diederich and B. Budiansky were given the task of coming up with a series of design graphs from which airplane designers could quickly determine the weight penalties involved in either aft or forward wing sweep.

Their work was published in Ref. 8.15. Major assumptions made were:

- Wings are made of aluminum, a homogeneous, isotropic material.
- Wings consist of two spars with caps and shear-webs and an upper and lower skin supported by stringers which run along constant percentage chord lines. This is the classical semi-monocoque arrangement found in many wings, even today.
- From an aero-elastic viewpoint it is assumed that the wing is rigidly clamped at the centerline.

Based on these assumptions a methodology was produced for arriving at trade studies, showing the effect of wing aspect ratio and sweep angle on wing weight. The results were dramatic, and showed that very large weight penalties were associated with forward swept wings. A generic trend of these data is shown in Figure 8.47.

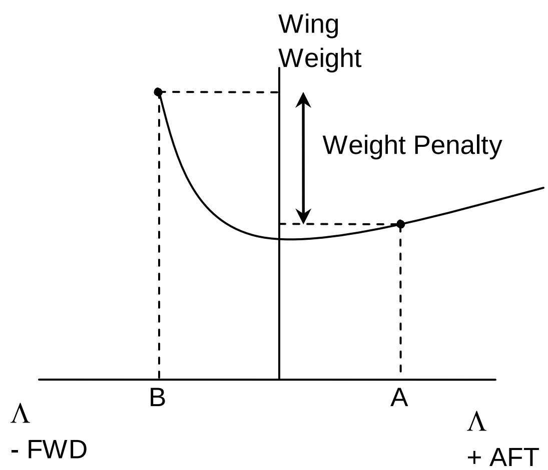

Figure 8.47 Generic Wing Weight Trend with Sweep Angle

As can be seen from Figure 8.47 the weight penalty of the forward swept wing is very large compared with the aft swept wing. It turned out that this report put the nail in the coffin of the forward swept wing until the 70's.

In view of all this, it is reasonable to ask whether or not there are any advantages to the forward swept wing? The answer is yes there are four:

1. A forward swept wing tends to have inboard flow, which at high angle of attack, promotes root stall. An aft swept wing produces tip stall and requires large negative twist to alleviate the effect of roll-off at the stall so typical for aft swept wings. The negative twist in turn causes an induced drag penalty in cruise.
2. For the same reason, ailerons on a forward swept wing retain their effectiveness to high angles of attack. Ailerons on an aft swept wing when used close to the stall tend to aggravate the tendency toward roll-off.
3. At high Mach numbers the shock sweep line on a forward swept wing has a larger sweep angle than on an aft swept wing. This allows a forward swept wing to be designed with a smaller sweep angle.
4. From a cross-sectional area distribution point of view, a forward swept wing has a lower wave drag than an aft swept wing. This effect becomes important in fighter aircraft. This is one of several reasons why Sukhoi used a forward swept wing on the Sukhoi S-37 Berkut of Figure 8.48.

Figure 8.48 Sukhoi S-37 Berkut (San Diego Aerospace Museum)

8.19.2 Solution

In 1974, Colonel Norris Krone of the USAF, finished his PhD dissertation, *Divergence Elimination with Advanced Composites*, at Purdue University under the guidance of Professor Terry Weisshaar. Before completing his dissertation, Krone questioned the assumptions made by Diederich and Budiansky with the following back-up rationale:

1. By using composite materials the homogeneous, isotropic assumption is no longer correct. This allows the tailoring of material properties to any particular requirement. In a forward swept wing, one would orient the fibers in such a manner that torsional stiffness is enhanced while sacrificing some bending stiffness. This method was used in the manufacturing of the wings of the X-29 (Figure 8.49).

Figure 8.49 Grumman X-29 (Courtesy Grumman)

2. It turns out that by machining a wing skin so that the rib-cap and stringers are located in a geodetic fashion a trade in favor of torsion stiffness also can be made.

3. If a forward swept wing airplane in a given flight condition would tend to diverge, the vertical load on the wing increases. This would accelerate the airplane upward. It then depends on how the span-wise mass distribution of the wing is arranged as to whether the divergence is enhanced or suppressed.

In other words, an airplane in flight does not behave like the "rigid clamping model." On the X-29 it was found that at some point the wing divergence couples into the rigid body freedom mode to produce a new type of flutter mode.

8.19.3 Lesson

When dealing with analyses of new technology, make sure all assumptions are carefully understood and justified. We now know that for certain types of airplanes the forward swept wing is definitely a suitable and even desirable candidate in configuration design decision making.

8.20 Unique Solution to an Extreme Range Requirement

8.20.1 Problem

In the 1980's Burt Rutan, President of Scaled Composites in Mojave, CA, decided to build a piston-propeller airplane that could fly around the world non-stop. This became the Voyager of Figure 8.50 which made its record non-stop-around-the-world flight in 1986. The configuration of that airplane was a radical departure from what was conventional.

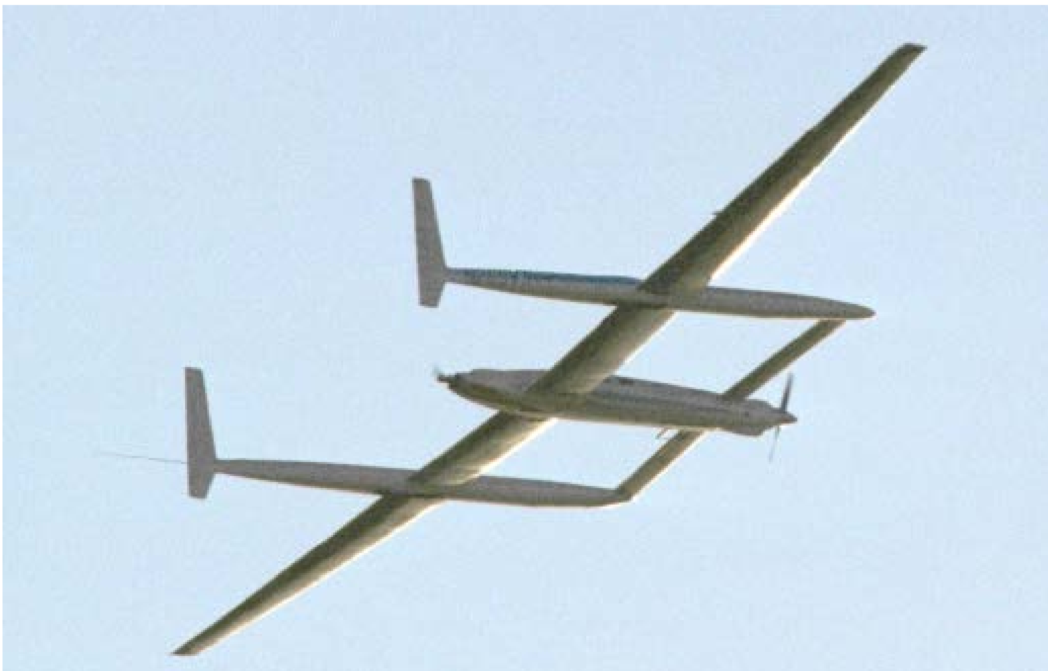

Figure 8.50 Rutan Voyager (Courtesy NASA)

Examine the Breguet range equation for propeller driven airplanes:

$$R = 375(\eta_p / c_p)(L/D)\ln(W_{begin}/W_{end}) \tag{1}$$

The distance around the world, R, is roughly 28,000 statute miles. Examine what the conventional airplane technology of the 1980's has to offer:

1. for the propeller efficiency: $\eta_p = 0.90$.
2. for the engine specific fuel consumption: $c_p = 0.40 \;\; lbs/hr/shp$.
3. for the airplane lift-to-drag ratio: $L/D = 25$.

For a very long range airplane it is not unreasonable to assume that the begin-weight-to-end-weight ratio is about that of the take-off-weight-to-empty-weight ratio. Jet transports are some of the best in terms of this ratio which typically is: $W_{begin} / W_{end} = 2.0$.

If these optimistic assumptions are substituted into Eqn (1) the result is a range of 14,621 statute miles, clearly far short of the objective. Because there is not much that could be done to improve items 1-3 significantly the only option a designer has is to see if a major improvement in the weight ratio is feasible. Substituting the 28,000 s.m. range into Eqn (1) with the assumptions 1-3, it is found that a weight ratio of 3.8 is required. Such a large weight ratio would require a drastic departure in structural and/or configuration design.

8.20.2 Solution

In conventional airplanes the design root-bending moment of the wing is a major factor in the design of the wing structure and, therefore, of the empty weight of an airplane. By using large masses away from the airplane centerline a relief of the design root bending moment is attainable. Figure 8.51 illustrates the principle.

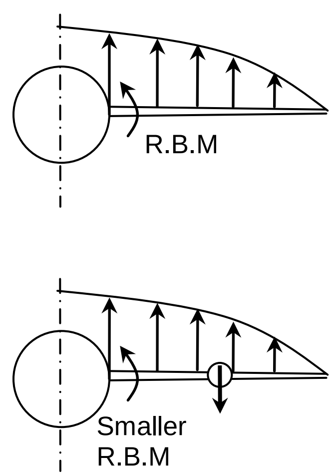

Figure 8.51 Root Bending Moment (R.B.M.) with and without Outboard Mass

By itself there was nothing new about using that design principle. Boeing had been using it for decades in the design of the B-47, B-52 and 707 transports, the engines representing the outboard masses. Also, fuel in wings has always been used to help alleviate root bending moments.

What Rutan did represented nevertheless a significant step forward. By using a twin-boom configuration with most of the fuel located in the booms a very large reduction in design root bending moment was attained. By also using advanced composite construction coupled with reduced design load factors Rutan was able to achieve the astounding weight ratio of 4.5. With that and a courageous flight crew the non-stop flight around the world was a success.

8.20.3 Lesson

Extreme mission requirements can often be met by coming up with a new and unique airplane configuration. For more reading on this topic, see Ref. 8.16.

8.21 More Examples of Area Ruling

8.21.1 Problem and Solution

Examples of other cases where excessive transonic drag was reduced by adjusting the local area ruling of a configuration are:

1. The Northrop F-5 tip-tank-to-wing integration

Figure 8.52 shows the local area ruling applied to that intersection. The figure also shows the very pronounced area ruling applied to the wing-fuselage intersection.

Figure 8.52 Northrop F-5 with Local Tip-tank-to-wing Area Ruling (Courtesy U.S. Air Force)

2. The Piaggio P-180 Avanti wing-nacelle integration

Figure 8.53 shows the local area ruling in the wing-nacelle intersection.

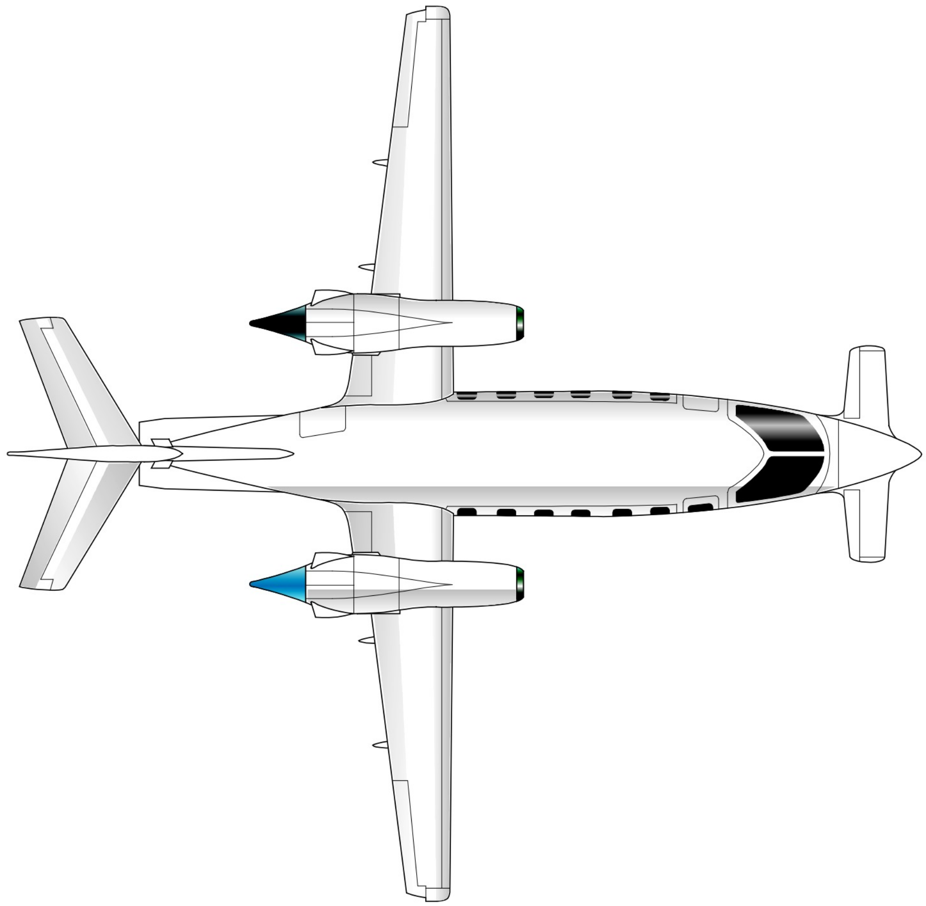

Figure 8.53 Piaggio P-180 Avanti with Area Ruled Nacelles

It is of interest to note that this airplane was designed to cruise at M=0.70 a Mach number where the need for area ruling was not expected. The need for this area ruling was identified during transonic testing of a wind tunnel model of the airplane in the Boeing Transonic Wind Tunnel.

3. The Cessna Citation X nacelle-fuselage integration

This airplane, in 2004 the fastest subsonic commercial airplane, required very extensive area ruling in the nacelle-fuselage area to obtain acceptable transonic drag characteristics. CFD applications had identified the need for this during early design and subsequent wind tunnel testing confirmed this. Figure 8.54 shows the extensive area ruling in the nacelle-fuselage area.

Figure 8.54 Cessna Citation X (Courtesy Tim Perkins)

8.21.2 Lesson

Designers of transonic airplanes need to keep this in mind before finalizing the external lines of any new airplane.

8.22 Canard with Close Coupled Propeller

8.22.1 Problem

In 1983 an experimental PAT-1 airplane (Figure 8.55), on a demonstration flight with four people on board, crashed while a recovery from an approach to stall maneuver was being performed by the pilot. There were no survivors.

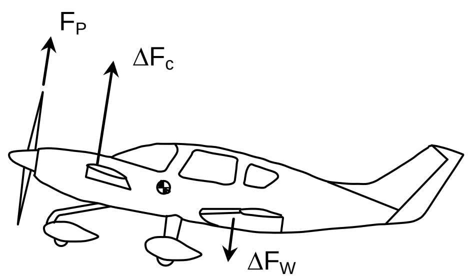

Figure 8.55 Side-view of the PAT-1 Airplane

8.22.2 Cause

The cause of the accident lies in the fundamentally undesirable pitching moment response of this configuration. Assume that the airplane is in an approach-to-stall flight condition with the power at idle. Next, assume that the pilot initiates a recovery by applying full power. The following three effects result:

- the lift over the canard increases due to the sudden slipstream. This results in a nose-up pitching moment
- the increased propeller normal force also produces a nose-up pitching moment
- the downwash field from the canard produces a down lift on the wing which also produces a nose-up pitching moment.

As a result, the airplane drives itself into a stall unless the pilot applies immediate full nose down controls. The slightest hesitation on the part of the pilot may stall the airplane. If this occurs at low altitude a crash is almost inevitable.

For a discussion of a wind tunnel investigation of this configuration, see Ref. 8.17.

8.22.3 Solution

The pitching moment due to the three forces identified in Section 8.22.2 (shown in Figure 8.55) should be recognized in the early design phase and analyzed for effect. Flight test is no time to uncover this type of problem. Adequate control power should be designed into the airplane to allow a pilot to cope with this foreseeable situation.

A preferred solution is to avoid the problem by using the pusher propeller configuration as was done in the Rutan Varieze of Figure 8.56.

Figure 8.56 Rutan Varieze (© 1999, EAA, Mark Schaible)

8.22.4 Lesson

A very important lesson is that the scenario described in Section 8.22.1 is predictable. Airplane designers should be expected to think through such scenarios before adopting a novel configuration.

8.23 Directional Stability Should be Required

8.23.1 Problem

In May of 1995 a Quad City Challenger (Figure 8.57) kit-built ultra-light spiraled into the ground near Bridgeport, CT. The pilot survived without injuries.

Figure 8.57 Quad City Challenger (Not Accident Airplane, Courtesy A. Brown)

The following is adapted from Ref. 8.18: "The pilot reported that while making a right climbing turn after take-off, at approximately 500 ft altitude the airplane started to bank to the left, despite the use of full right rudder. Unable to regain control the airplane entered an ever tightening left, flat spiral into the ground."

8.23.2 Cause

Examine Figure 8.57. Most experienced airplane designers would agree that the airplane probably lacks adequate directional stability because of the large amount of side-surface area ahead of the c.g. and the relatively small vertical tail. Close examination of Figure 8.57 shows that there are no side doors (open cockpit). Therefore, air will be able to flow through the side openings on one side and out the other side. The effect of that is to effectively reduce the destabilizing side area forward of the c.g.

This particular airplane had been modified by the installation of doors to enclose the cockpit area. Now the large side area forward of the c.g. will indeed reduce directional stability. In fact, the kit manufacturer stated that when the doors are installed the airplane becomes “rudder dominant” (this is a rather unscientific term not found in stability and control textbooks) and that pilots must not take their feet off the rudder pedals.

The airplane lacks directional stability and becomes difficult to control.

8.23.3 Solution

The solution is to either leave the doors off or enlarge the vertical tail.

8.23.4 Lessons

1. The pilot of the accident airplane was very lucky. Certified airplanes are required to exhibit directional stability anywhere in the flight envelope. There are no regulations for this covering ultra-light aircraft. The author believes that all such airplanes should be required to have inherent directional (and, for that matter, longitudinal) stability.

2. Lacking regulations, it should be the ethical responsibility of designers of non-certified airplanes to make sure that their designs are at least stable. Whether or not one certifies a light airplane, the FAR 23 requirements should be considered a minimum standard to work toward.

Chapter 9

Lessons Drawn from Marketing, Pricing and Program Decision Making

"Marketing is not an exact science"

Dr. Jan Roskam, 1990

9.1 Introduction

In this chapter a series of problems and missteps which occurred in marketing, pricing and program decision making are described. Finally, lessons learned are stated.

9.2 Cessna 620

9.2.1 Problem

In 1952 the Commercial Aircraft Division of Cessna Aircraft Company was working on a new airplane, the Cessna 620. This was an impressive, stand-up cabin, four-engine, propeller driven transport aimed at the executive market (Figure 9.1). Ralph Harmon, a project engineer of Lockheed was offered the job of managing the entire design and development.

Because there were no piston engines available of the right size to make the airplane into a twin it was decided to use an existing engine and accept a four-engine configuration. Cessna was well aware of the turboprop studies which were underway at Pratt & Whitney which eventually led to the famous PT-6 series.

Figure 9.1 Cessna 620 (Courtesy Cessna)

The judgment was made that these engines would take too long to develop. Two prototypes were built with "soft" tooling to minimize the required development investment. The airplane flew very well, met all performance expectations and had good handling qualities. Ref. 9.1 contains a detailed account of the development of this airplane.

In 1957 Cessna was confronted with the decision to invest in "hard" tooling to launch the airplane into production. Before doing this, Cessna management decided it wanted a definitive marketing study done. This job was given to a well-known marketing firm in Chicago. That firm came up with a very negative report about the marketability of that type of executive transport. Cessna cancelled the project and all engineers working on the 620 were given "pink slips." Management also made the strange decision to have both prototypes destroyed.

9.2.2 Lesson

Marketing is not an exact science. Also, management should preserve prototypes. The least they could do is donate them to a museum.

Some years later this market really blossomed. Beech Aircraft eventually walked away with that market with an airplane called the King-Air of which thousands were built.

9.3 Convair 880/990

9.3.1 Problem

One of the most expensive commercial failures in the history of aircraft programs was the Convair 880/990 program. Refs. 9.2 and 9.3 contain interesting details on the development history of both airplanes. The design of the Convair 880 (Figure 9.2) was started in 1954 in response to a request from billionaire Howard Hughes for his airline, Trans World Airlines (TWA).

Figure 9.2 Model of Convair 880 (Courtesy geminijets.com)

Because Mr. Hughes procrastinated, Boeing and Douglas had brought out their 707 and DC-8 designs before Convair (at that time a division of General Dynamics Corporation) could finalize the design of the 880 (which at first was called the Model 22 Skylark). The airplane wing was sized to give the airplane a 5,000 ft field-length capability. This resulted in a wing loading of 97 psf (rather low for a jet transport) but gave the airplane sufficient fuel volume to give it trans-atlantic range. A problem would be that the fuselage was sized for 5 abreast operations as opposed to the 707 and DC-8 which were sized for 6 abreast.

During high speed flutter flight testing the airplane lost much of its vertical tail and rudder but was landed successfully by the test crew. With proper flutter calculations and ground vibration testing this should never have happened.

It turned out that the sales price of the airplane was far below its cost and only 65 were manufactured.

The Model 990 (Figure 9.3) was a faster, longer version of the 880 and was billed for M=0.91 cruise speed.

Figure 9.3 Model of Convair 990 (Courtesy geminijets.com)

It turned out that the transonic drag rise was much more rapid than predicted and with the help of NASA, aerodynamic modifications were made in the form of the trailing edge anti-shock bodies, seen in Figure 9.3. Even with this the cruise drag was higher than predicted and only 37 were manufactured.

Reportedly, General Dynamics lost over $450,000,000 on this combined program by the time it was terminated. The loss amounted to about 4.4 million per airplane produced. At that time, that was about the production cost alone!

9.3.2 Lesson

Developing commercial aircraft is very expensive. If performance guarantees are not met and/or deliveries are not made on schedule, major losses will be incurred.
A simple way to look at an affordable unit cost per airplane is to consider Figure 9.4. In this figure the assumption is made that the total non-recurring cost is $500,000,000. The cost

contribution per unit produced obviously varies hyperbolically with the number of airplanes built. Figure 9.4 shows that if the market price of an airplane is, say $40,000,000 and the unit cost to produce (including all engines, avionics and customer equipment) is $38,000,000 then it takes at least 250 airplanes produced to break even. That general area of the curve is referred to as the "knee" of the curve.

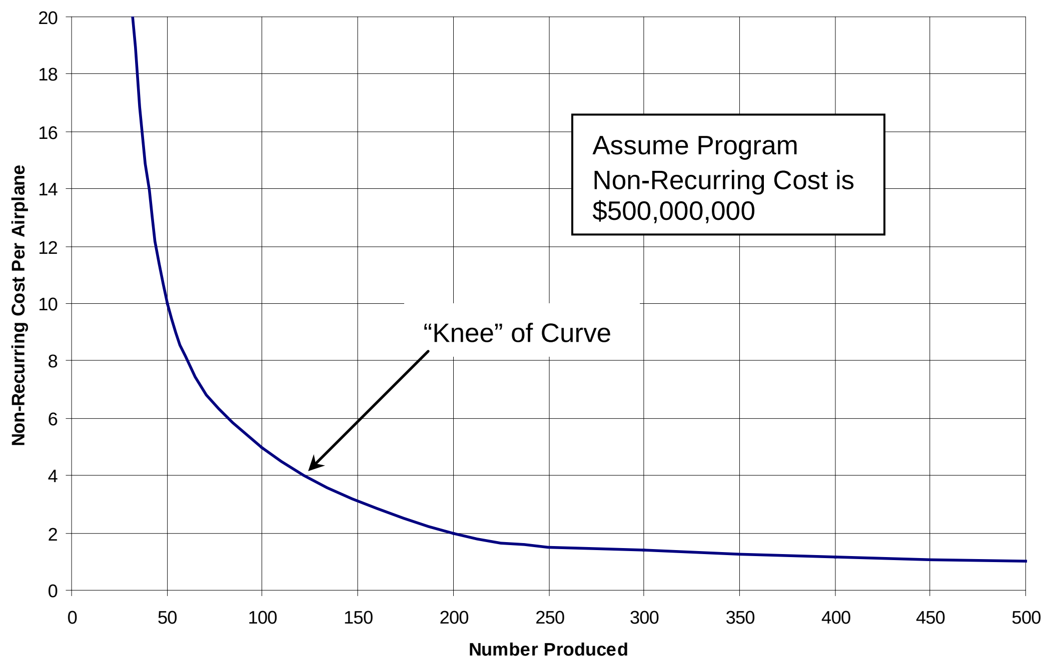

Figure 9.4 Non-recurring Cost per Airplane versus Numbers Produced

The big lesson here is to not start a program until reasonable assurance exists that:

- the airplane can be produced for a known cost
- the market exists for a number of airplanes beyond the knee of the curve.

9.4 McDonnell 119 and 220

9.4.1 Problem

In 1957 Ralph Harmon (of Cessna 620 fame) was offered the job to manage design studies at McDonnell Aircraft Corporation in St. Louis, MO. These studies led to the Models 119 and 120 (Figure 9.5). See Ref. 9.4 for more details.

Figure 9.5 McDonnell 119 (Courtesy Boeing)

The project was launched in the hope to win the USAF UCX (Experimental Utility Aircraft) program. A 707 type configuration was selected powered by four Fairchild J-83 turbojets. As it turned out Fairchild abandoned the J-83 project and as an interim solution the Westinghouse J-34 engine was selected. The production version of the airplane was to have the P&W JT12A engines.

The prototype flew in January of 1959. The flight tests indicated that the airplane would meet all expectations but the proximity of the nacelles to the ground presented an unacceptable FOD problem. The USAF opted for the Lockheed Jetstar (Figure 9.6) which had its four engines installed on the aft fuselage and for the North American Sabreliner (Figure 9.7) which had two engines mounted on the aft fuselage.

Figure 9.6 Lockheed Jetstar (Courtesy D. Schulman)

Figure 9.7 North American Sabreliner (Courtesy F. Duarte Jr.)

McDonnell management then decided to go after the executive market and modified the airplane internally to seat up to 29 persons. The airplane received its type certificate in 1960. It was used as a demonstrator and as a McDonnell corporate transport. However, because of the slow development of the corporate market the project was terminated in 1966. The sole Model 120 was sold to the Flight Safety Foundation of Arizona. In 1979 the airplane was reported languishing at the Albuquerque Airport in New Mexico.

9.4.2 Lessons

Starting an airplane program with an as yet undeveloped engine is inviting major program delays. FOD is an important consideration for airplanes that can be expected to be operated out of airfields without constant runway cleaning and monitoring. The 707, DC-8, 737, 747 etc. are all operated from “clean” runways and taxiways and FOD has not been a major problem.

9.5 Boeing 909

9.5.1 Problem

In 1958, Boeing operated an Industrial Products Division in Seattle. That Division had a gas generator in production for various types of armored vehicles. By adding a gear box and a propeller to such a machine one could develop a turboprop engine. The next logical thought was to marry that turboprop to an airplane: the 909 (Figure 9.8).

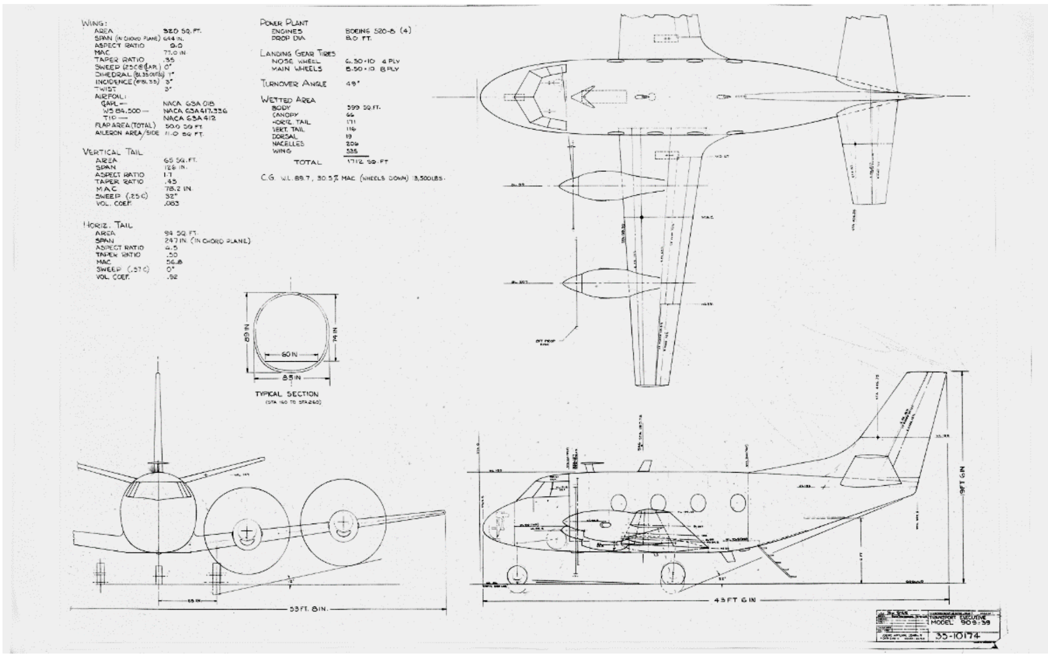

Figure 9.8 Three-view of the Boeing 909 (Courtesy Boeing)

The task to design and develop that airplane was assigned to the Wichita Division and the author was the lead engineer for stability and control. We did quite a bit of wind tunnel testing and ended up with what looked like a viable airplane. However, Seattle management had a change of heart. A marketing study (done by the same firm which torpedoed the Cessna 620 discussed in Section 9.1) showed that there would not be a profitable market for such an airplane. The project was cancelled.

9.5.2 Lesson

One more time: marketing is not an exact science.

9.6 Boeing 707 and Douglas DC-8

The Boeing 707 (Figure 9.9) and Douglas DC-8 (Figure 9.10) series of commercial airplanes were in direct competition with each other.

Figure 9.9 Model of Boeing 707 *Figure 9.10 Model of Douglas DC-8*
(Both images courtesy of geminijets.com)

The following dates and production numbers are noteworthy:

- Boeing 707 production deliveries began in 1958 and ended in 1987 with 827 built.
- Douglas DC-8 production deliveries began in 1958 and ended in 1972 with 556 built.

9.6.1 Problem

In the early 1970's the airliner market had hit a major slump. Order books at Boeing and Douglas for commercial jet liners decreased significantly.

For management the question then is: "is this product still viable and, if not, when do we terminate production." At Douglas the decision was made to terminate production, at Boeing the decision was made to continue building the 707 at the minimum economical production rate and park some of them in the desert as "white tails".

The fact is that sales forecasts are often wrong. In this case, Boeing management guessed right and Douglas management guessed wrong.

Boeing did have one advantage over Douglas: the military version of the commercial 707 kept on selling fairly well. A total of 120 cargo versions and 820 tanker versions were built with the last one delivered in 1991.

9.6.2 Lesson

Staying power is often necessary to have a truly successful production program.

The reader is encouraged to read Refs. 9.3 and 9.5 for more detailed insight into the competitive development of the 707 and DC-8 airplanes.

9.7 Boeing 720

9.7.1 Problem

The Boeing 720 (Figure 9.11) was a derivative of the 707. A total of 154 were sold. Production of the 720 ended in 1967.

Figure 9.11 Boeing 720 (Courtesy D. Schulman)

The airplane was aimed at shorter stage lengths with fewer passengers than the 707 and a much lower empty weight was critical to its success. This was achieved by shortening the fuselage. The inboard wing sweep angle was increased and full span Krueger flaps were fitted to the leading edge. A major reduction in weight was achieved by the incorporation of 7075ST skin panels.

That turned out to be a major mistake. During winter operations these panels were subjected to wheel sprays laden with salt and other chemicals. As a result, minute corrosion pitting occurred which spread rapidly to cause significant structural integrity concerns. Boeing ended up retrofitting more conventional aluminum alloy panels to the lower side of all 720 wings. Since the airplane was sold with certain range-payload performance guarantees Boeing also had to make up the difference.

This derivative program by itself caused significant losses which were happily offset by the continuing success of the 707 program.

9.7.2 Lesson

The lesson here is that when a new material is to be used in an environment where it was not used before it is a good idea to run tests of the behavior of this material in that new environment. This is referred to as "technology validation". It should not be forgotten.

9.8 Dassault Mercure

9.8.1 Problem

In 1971 the Dassault Mercure transport (Figure 9.12) made its first flight.

The Mercure was to be the first commercial jet transport designed and built by Dassault, known primarily for its jet fighters and business jets. The airplane was primarily aimed at Air Inter, the French domestic carrier which was operating the 99 seat Caravelle and wanted a larger airplane. It took considerably longer to certify the airplane than Dassault had promised forcing Dassault to pay for a period of two years the additional leasing cost of five Caravelles for Air Inter.

Figure 9.12 Dassault Mercure (Courtesy E. Marmet)

This, coupled with the fact that the airplane could not compete with the DC-9 or the 737 from a range-payload viewpoint made the airplane not attractive outside France.

Production was terminated after a total of twelve were produced.

9.8.2 Lesson

Another example of an airplane sized to the requirements of one customer. If it had not been for subsidies from the French government, Dassault would not have been able to stay in business.

9.9 Lockheed 1011 and its Rolls Royce RB-211 Engines

9.9.1 Problem

The Lockheed 1011 (Figure 9.13) made its first flight in 1970.

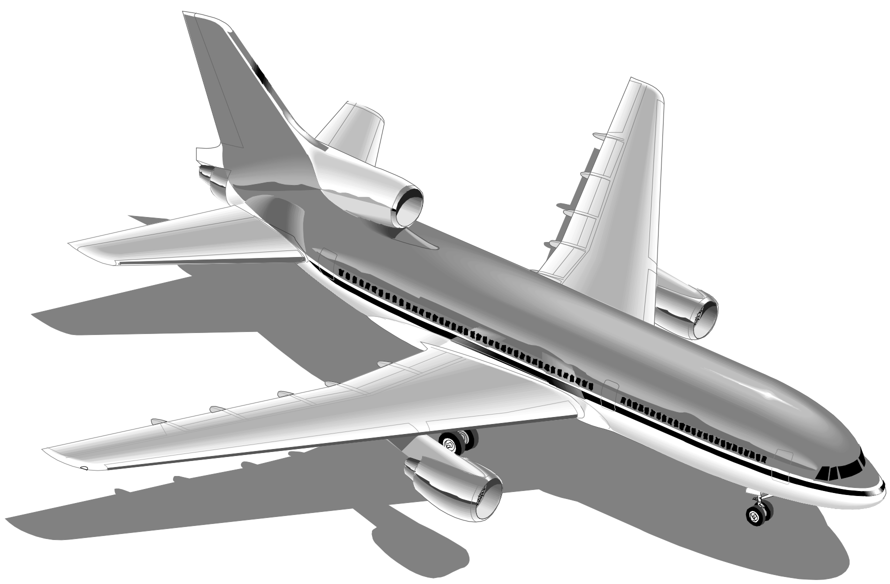

Figure 9.13 Lockheed 1011

The program was officially announced in 1968. The airplane was a direct competitor of the Douglas DC-10. To make the airplane competitive with the DC-10, Lockheed decided to use the new Rolls Royce RB211 engines. These were advertised as significantly lighter than similar engines offered by Pratt & Whitney and General Electric. The enabling engine technologies were the Rolls Royce patented “hyfill” fan blades and a three stage low pressure turbine. Well into the development of the new RB211 a major problem arose: the engine could not withstand the so-called bird-strike requirement. As a result Rolls Royce ran into financial problems: the cost of the RB211 program had gone way over the estimates. Rolls Royce went bankrupt and was salvaged by the British government. The resulting program delays were such a problem for Lockheed that it needed U.S. government loan guarantees to survive as a company.

9.9.2 Lesson

When designing an airplane program around a new, as yet uncertified engine which uses new technology, beware. Any hick-ups in the engine development program will have significant consequences to the viability of the airplane program.

Rolls Royce and Lockheed both eventually recovered and the RB211 series of engines did become a success. The Lockheed 1011 did turn out to be a good airplane but only 249 were built, not enough to make this a profitable program. Refs. 9.6 and 9.7 contain details on the development and production history of the Lockheed 1011.

9.10 Pricing Yourself out of the Market

9.10.1 Problem

The Piaggio P-180 Avanti (Figure 9.14), not unlike any new airplane type, was rather expensive to develop.

Figure 9.14 Piaggio P-180 Avanti (Courtesy L. Willems)

When selecting a new configuration there is not much historical background available: much testing will have to be done to prove “equivalent safety”. The total cost to study, design, develop, build prototypes and do all the required certification flight testing is referred to as the Research, Development, Test and Evaluation (RDTE) cost of a new airplane program. This cost, in economic terms, is a non-recurring cost.

The objective of a typical commercial enterprise is to make money. Making money on an airplane program means that the selling price consists of the cost to produce, the amortized non-recurring cost plus a profit margin.

Say that the total non-recurring cost of a program is $200,000,000 dollars. If the decision is made to turn a profit after selling 100 airplanes, the cost of amortizing the RDTE money is $2,000,000 per airplane. If the fair market price of the airplane is about $3,000,000 it becomes virtually impossible to have a profitable program since the cost of engines and avionics is around $600,000 and the cost of manufacturing is around $500,000. All of this in 1988 US dollars.

To have a profitable program over 100 airplanes built would require the price of the airplane to be $3,400,000 if a margin of 10% is considered acceptable. The airplane is not competitively placed in the market and probably will not sell well. If a longer term view is taken and the RDTE is to be amortized over a 200 airplane program, the price of the airplane would drop to $2,400,000 which would make it attractive.

9.10.2 Lesson

In any new commercial airplane program a decision has to be made over how many production airplanes the non-recurring costs are to be amortized. The number of airplanes that can be sold depend on the price offered, the service and support level that can be offered to the customer, the production rate that can be sustained and the market share that is envisioned.

Note that management actually has to make a number of guesses. As seen before (Sections 9.1 – 9.8), one can never be sure of the ultimate market for a new airplane. In starting a new airplane program all these factors have to be carefully weighed.

For a variety of reasons not related to the P-180 program, Piaggio ran into financial difficulties until rescued by a re-organization resulting in the Ferrari family taking control of the company. At the time of publication of this book the P-180 is reasonably priced and competing very well in the corporate and air-taxi market.

9.11 VisionAire Vantage

9.11.1 Problem

The VisionAire Vantage airplane (Figure 9.15) was envisioned as a single pilot, single engine, low cost business jet that would take advantage of the well-established reliability of the P&W Canada JT-15D turbofan engine.

Figure 9.15 Wind Tunnel Model of VisionAire Vantage

By using one engine the airplane was to have a considerably lower cost. By using carbon-fiber composite construction, the exterior surfaces would be very smooth for low drag. By using a forward swept wing, advantage would be taken of the better low speed flying qualities because of the tendency of such a wing to stall at its root and because of the ability to maintain positive aileron control at the stall. The latter characteristics had been well demonstrated on the Grumman X-29 program.

Scaled Composites of Mojave, CA was commissioned to build a proof-of-concept airplane (called the POC at VisionAire) without the benefit of a wind tunnel test program.

When the POC was flown by DER (Designated Engineering Representative) test pilots several undesirable flight characteristics were found to exist which would make certification highly improbable without modifications.

One of several faults of the POC was the inability to "pick-up a wing" with rudder alone, a characteristic which is specifically required for FAR 23 airplanes. For an airplane to exhibit this ability requires the rolling moment due to sideslip derivative, C_{ℓ_β}, to be negative. A well-known characteristic of a forward swept wing (without geometric dihedral) is a positive value of C_{ℓ_β}. To generate the correct, negative magnitude of this derivative requires a fairly large, positive geometric dihedral. This was already known in 1943 when Junkers developed the Model 287 (See Figure 8.46).

In 1998 DARcorporation was given a contract to conduct a series of wind tunnel tests on a model of the Vantage airplane to determine the cause of the various short-comings in flying qualities. The tunnel test clearly showed that the test pilots were correct in their assessment. The test results were summarized in a memo written to VisionAire management outlining which modifications should be made to the airplane.

It is hard to believe, but management had already spent large sums of money on production design and tooling. By the time it was recognized that the POC was in fact not certifiable, the company ran out of money and the project was halted. Sometime later VisionAire filed for bankruptcy and ceased operations.

9.11.2 Lesson

When starting a high performance airplane project, do not start production and tooling design until assured of a certifiable configuration. Despite all best CFD that can be brought to bear on a prototype, it is low cost insurance to verify critical flight characteristics in the wind tunnel. Had VisionAire management done this before committing to building the POC the airplane might have become successful in the market place.

9.12 Eclipse 500

9.12.1 Problem

Eclipse Aviation of Albuquerque, New Mexico was formed in 1998 to develop the Eclipse 500 (Figure 9.16) with Williams EJ-22 engines. These engines were not certified nor had they been flight tested when detail design of the Eclipse 500 began.

Figure 9.16 Eclipse 500 with Williams EJ-22 Engines (Courtesy Eclipse Aviation)

The first flight was a disappointment in that it showed that the engines did not meet expectations. Eclipse then cancelled its contract with Williams and switched to Pratt & Whitney PF610F engines flat-rated at 900 lbs of thrust. These engines were undergoing flight certification testing in 2004. A variant of that engine was also selected by Cessna to power the Mustang jet.

Switching the engines represented a fairly major change since the P&W engines are significantly larger which has an effect on longitudinal stability and deep stall recovery characteristics. As a result the empennage had to be adjusted and the weight of the airplane increased.

Also, significant program slippage was caused. The first P&W equipped airplane flew in early 2005 and was certificated in 2006. Finally, the price of the airplane, first set below 1 million dollars had to be increased significantly to $1.52 Million (June 2006 USD). This was not only due to engine change, but also due to increased cost of raw materials and underestimation of supply costs.

9.12.2 Lesson

The consequences of the re-engining problem of the Eclipse 500 on overall program cost, schedule slippage and program profitability has not been published. However, a two year delay in any program is very costly. Unless management can exercise “hiring and firing” at will, a large number of skilled people have to be maintained in the workforce to remain a viable company. It is generally not a good idea to base a new airplane program on an entirely new, uncertified engine.

9.13 Safire S-26

9.13.1 Problem

The Safire Aircraft Company was formed in 1998 to develop the Safire Jet shown in Figure 9.17. Financing was available only for basic engineering development.

Figure 9.17 Safire S-26 Jet (Courtesy Safire Aircraft)

The airplane was originally designed to be powered by the Agilis TF-800 turbofan engine, a new, uncertified engine being developed by a new company. A wind tunnel test was run which identified a number of minor problems which needed to be solved.

In due course it became evident that no assurances about the viability of the Agilis engine could be given and a switch was made to Williams FJ-33 engines which were certified in 2004. The change in engines caused a change in the rear fuselage as well as the empennage and another wind tunnel test was accomplished to make sure that the airplane could be certified. By that time the company had run out of money, a new financing package could not be put together and the future of this program is in doubt.

9.13.2 Lesson

Starting a new airplane program with inadequate financing in place is a recipe for failure. Starting a new airplane program with a non-certified engine is also a recipe for program and financial setbacks.

Chapter 10

Summary of Lessons Learned

"Seriously: The devil is in the details"

Dr. Jan Roskam, 1990

10.1 Introduction

A summary of design lessons learned from aircraft accidents and incidents (described in more detail in Chapters 2-8) is presented in this chapter.

10.2 Lessons Learned from Operational Experience (Chapter 2)

1. Design gust lock systems so that take-off with the locks engaged cannot be made	Section 2.1 Douglas C-54
2. Design the airplane so that it will un-stick even if maximum rotation is commanded at the start of the take-off roll.	Section 2.3, 2.13 DeHavilland Comet
3. Design the gear retraction system such that it cannot be retracted during ground operations.	Section 2.4 Martin 404
4. Develop robust methods and/or procedures to eliminate take-offs and landings with the center of gravity too far aft or too far forward.	Section 2.2, 2.9, 2.15, AVRO Tudor, Douglas DC-7, Beech 1900
5. Design the rudder control system so that in multi-engine airplanes OEI at take-off is a non-event.	Section 2.6, 2.12 Boeing 707
6. Design wing airfoils so that the sensitivity to minor icing on take-off is minimized.	Section 2.18, 2.21 Douglas DC-9
7. Design high lift systems so that take-off without flaps is inhibited	Section 2.19 Douglas DC-9

10.3 Lessons Learned from Structural Design (Chapter 3)

1. Avoid stress concentrations in wing fittings and design bolt connections so that bolts are loaded in shear and not in tension.	Section 3.2 Martin 202
2. Pressure loads on canopies can be very large and result in distortion and failure of the locking fittings.	Section 3.3 Fokker S-14, Cessna T-37
3. Fuselage pressurization cycles cause fatigue cracks in areas of stress concentrations such as window frames and fuselage joints. Water-tank fatigue testing is required before certification of new transports can be obtained.	Section 3.5 DeHavilland Comet
4. Bird strikes can cause serious (sometimes fatal) damage to aircraft structures, wind-screens, engines and systems	Section 3.7 Vickers Viscount
5. Structural failure due to corrosion fatigue should be prevented at the design level by the proper choice of materials and manufacturing tolerances.	Section 3.9 DH-104 Dove STC
6. Failure of a pressure bulkhead should not lead to such damage that an airplane becomes uncontrollable. Also, all primary structure should be designed such that it is easily inspectable.	Section 3.10, 3.14 Airbus A310, Boeing 747
7. Crack propagation in primary structural components should be detectable.	Section 3.11, 3.12, 3.15 Lockheed L-382, Boeing 720, 737
8. Mechanical control systems should be designed so that when both pilots exert cockpit control forces with their maximum strength no structural failure occurs in the control system.	Section 3.13 Embraer EMB-110P1
9. No matter how careful cargo door latching mechanisms are designed and inspected failures will occur. When a door opens in flight the door-hinges should not fail in a manner to “peel” the passenger cabin skin open. Design of door hinge attachments should account for this.	Section 3.16 Boeing 747

10.4 Lessons Learned from Flight Control System Design (Chapter 4)

1. Route flight controls away from heat sources.	Section 4.1 Westland Whirlwind
2. Design internal flight control surface gust locks so they cannot be engaged in flight.	Section 4.3 Douglas DC-4
3. Route engine, flight control cables and flight crucial electrical wires to be protected from propeller blade failures.	Sections 4.4, 4.7 Lockheed L-649 Douglas DC-6A
4. Design all flight crucial components for a one-way fit. If this is not done, Murphy's Law will strike.	Sections 4.5, 4.6 Convair 340, Douglas DC-3
5. Design flight controls such that critical bolts cannot back out due to maintenance errors.	Section 4.8 Lockheed L-049
6. Design flight control systems so that a run-away trim scenario can be controlled within limits of human capability	Section 4.9 Lockheed L-18
7. If an airplane is certified to be dispatchable with a failed pitch-trim (or Mach trim) system its stick-force-speed gradients should not reverse following such a failure.	Section 4.10 Douglas DC-8
8. Design pitch and yaw dampers so that reversal of polarity is not possible when repairs or sensor replacements are carried out.	Sections 4.11, 4.18 Lockheed C-141, Learjet 36 (prototype)
9. Do not use external gust locks: crews forget to take them off	Section 4.12 Douglas DC-3
10. Elastic stop-nuts should not be used in primary or secondary flight control systems. All nuts should be of such a type that they can be wired to their bolts to prevent rotation. Castellated nuts are a preferred way of achieving that.	Section 4.13 Cessna 340
11. Design a flight control system so that it can be checked for functionality before take-off. Also, in a servo-tab system in particular, elevator position indicators should be included.	Section 4.14 Douglas DC-8
12. Structural failure of a control surface actuator should not result in loss of control.	Section 4.15 Boeing 707/720

13. Locate flight control system cables so that no local structural or systems failure can result in loss of control.	Section 4.16 Douglas DC-10
14. Means to verify whether or not autopilots keep an airplane in trim should be part of each autopilot installation	Section 4.17 Boeing 707

10.5 Lessons Learned from Engine Installation Design (Chapter 5)

1. Design landing gears so that propeller-ground clearance is adequate even with larger diameter propellers	Section 5.1 SAAB Scandia
2. Design propeller reversing systems so that in-flight failure is extremely improbable	Sections 5.2, 5.3, 5.4, 5.6, 5.14Convair 240, Martin, 202, Douglas DC-6, Lockheed Constellation
3. Jet engine exhaust fairings should be carefully shaped to avoid excessive drag.	Sections 5.5, 5.13, 5.14 Douglas A4D, Vickers VC-10, NAA XB-70
4. Design propeller reversal controls in the cockpit so that inadvertent actuation by the crew is not possible.	Sections 5.7, 5.18, 5.20, 5.21 Vickers Viscount, Embraer EMB-120, SAAB 340, Fokker F-27
5. Design buried engine installations so that engine removal can be accomplished without removing primary structure.	Section 5.8 Cessna T-37
6. Design mechanical throttle controls so that engine mounting compliance cannot cause a flame-out.	Section 5.9 Cessna T-37
7. Design primary structure and flight-crucial controls so that propeller blade separation cannot destroy structural integrity or system integrity.	Sections 5.16, 5.17, 5.22 Convair 340, DH-114, Embraer EMB-120
8. Design jet engine installations such that uncontained failures do not cause loss of life, loss of structural integrity or loss of flight crucial systems integrity.	Sections 5.21, 5.23 Douglas DC-9, Douglas MD-88
9. Design jet engine thrust reverser controls so that they cannot be deployed during flight.	Sections 5.19, 5.24 Boeing 767, Fokker 100
10. Make sure that jet engine inlets cannot ingest fragments from failed tires.	Section 5.25 BAC-111

11. Make sure that fuel lines cannot be chafed through by adjacent cables, clips or other items that move during wing flexure or flight/flap control cable movement.	Section 5.26 Boeing 747
12. Manufacture solder joints in fuel control boards so that joint cracking can’t result in fuel shut-down	Section 5.27 Boeing 717

10.6 Lessons Learned from Systems Design (Chapter 6)

1. Design electrical power leads so that unintentional contact with other conductors (such as metal skins) is impossible.	Section 6.1 Lockheed Constellation
2. Design electrical systems so that any sparks that may be generated cannot ignite flammable materials.	Section 6.2 NAA Navion
3. Design fuel vents so that any fuel released from the vents will not enter other airplane systems.	Section 6.3 Douglas DC-6
4. Design fire extinguishing systems so that their use cannot result in crew incapacitation.	Section 6.4 Douglas DC-6
5. Design hydraulic systems such that a single failure does not cause brake system failure.	Section 6.5 Martin 202
6. Design landing gear actuation controls so that inadvertent gear retraction is highly improbable.	Sections 6.6, 6.8 Boeing Stratocruiser, Douglas DC-4
7. Design service door fasteners so that doors cannot open in flight because of improper fastener closing.	Section 6.7 Lockheed Constellation
8. Do not put fuel shut-off valves in landing gear wells where rotating wheels can inadvertently actuate them.	Section 6.9 Curtiss C-46
9. In a 4-engine airplane, design hydraulic systems so that with two engines failed on one side the flaps and gear can still be operated.	Section 6.10 Lockheed Constellation
10. Design hydraulic systems so that they can cope with simultaneous retraction of landing gear and flaps.	Section 6.11 Lockheed Constellation
11. Design fuel systems so that fuel can be pumped from any tank with the airplane in any reasonably expected attitude.	Section 6.12 Cessna 310
12. Design fuel vent systems so that asymmetry between left and right systems cannot cause unwanted fuel transfer.	Section 6.13 CessnaT-37

13. Design cabin doors so that accidental opening in flight is extremely improbable.	Section 6.14 Douglas DC-3
14. Design fuel systems so that high forward accelerations during take-off cannot transfer large quantities of fuel aft and cause longitudinal instability.	Sections 6.15, 6.38 Boeing B-52, Bombardier CL-604
15. Design cargo system lights so that flammable materials in the cargo hold cannot ignite when the light(s) are left on.	Section 6.16 Curtiss C-46
16. Design cabin doors such that positive closing is easy from the outside and so that the door cannot open in flight.	Section 6.17 Convair 340/440
17. Design integral fuel tanks such that lightning strikes are extremely unlikely to cause an explosion.	Section 6.18 Boeing 707
19. Design landing gear attachment braces such that in a hard landing no fuel lines can be penetrated.	Section 6.19 Boeing 727
20. Design APU system installations such that failure of an air valve cannot cause afire which then disables the longitudinal control system.	Section 6.20 BAC-111
21. In twin engine airplanes avoid powering the electrical system from one engine and the hydraulic system from the other.	Section 6.21 Piper Apache
22. Design for extra redundancy in flight crucial systems in large passenger transports.	Sections 6.22, 6.27 Boeing 747, Lockheed L-1011
23. Design the brake system in transport airplanes so that a single engine failure cannot cause a brake failure.	Section 6.23 Douglas DC-10
24. Engine pylon attachment structure should be assumed to fail. Stall warning systems should be redundant. Slats should have slate brakes to minimize asymmetry.	Section 6.24 Douglas DC-10
25. Design service lifts in airplanes so that they cannot be operated from one end if the other end is being worked on.	Section 6.25 Douglas DC-10
26. Design slat systems with brakes to avoid asymmetry unless the airplane can be controlled in roll at the stall speed.	Section 6.26 Douglas DC-10

27. Design flap systems to avoid asymmetry unless the airplane can be controlled in roll at the stall speed.	Section 6.28 Piper PA-31
28. Design wind shield washer systems that use flammable liquids so that fluid leakage cannot cause a fire.	Section 6.29 DeHavilland DHC-6
29. Do not place all hydraulic lines in the leading edge.	Section 6.30 Rockwell B-1B
30. Do not place flight crucial electrical or avionics systems so that liquids (water or other) can leak into them.	Sections 6.31, 6.37 Fokker 100, Boeing 717
31. Design fuel filter covers for one way fit. Design nacelle cowls so that when they pop open in flight no flight crucial systems will also be damaged.	Section 6.32 DeHavilland DHC-8
32. Design ground spoiler systems so they cannot deploy in flight.	Section 6.33 Douglas DC-9
33. Design hydraulic systems without the need for the crew to ensure that the pressure level is properly selected.	Section 6.34 Douglas DC-9
34. Design cabin door latching systems without cables. Instead use push-rods.	Section 6.35 Beechcraft 1900
35. Design landing gear actuation systems so that corrosion cannot incapacitate them.	Section 6.36 Boeing 737
36. Design landing gear doors so that if they cannot retract no ground contact can occur upon landing.	Section 6.39 Boeing 717
37. Design galley chiller fans so they cannot chafe and cause an in-flight fire.	Section 6.45 Boeing 747

10.7 Lessons Learned from Maintenance and Manufacturing (Chapter 7)

1. Avoid the introduction of undetectable gouging in propeller blade manufacturing.	Section 7.1 Douglas DC-6
2. In the design of flight crucial systems do not rely on maintenance to prevent control bolts from backing out.	Sections 7.2, 7.3, 7.7 Curtiss C-46, Convair 340
3. Ensure adequate firewalls exist to prevent fire from getting to primary structure.	Section 7.4 Convair240
4. Design propeller reversing systems so that uncommanded in-flight reversals are extremely improbable.	Section 7.5 Lockheed Constellation
5. Design landing gear snubber systems so that contact with the main gear truck is extremely improbable.	Section 7.6 Boeing 707
6. Design flight control systems so that improper installation during maintenance is impossible.	Section 7.8 Lockheed L-188
7. In manufacturing operations which utilize a quenching process check the chemical composition of the quenching liquids	Section 7.9 Lockheed SR-71
8. When using hand-lay-up procedures in composite structures consider the effect this has on empty weight control	Section 7.10 Windecker Eagle
9. When using metal bonding in areas with small radii consider the effect of mis-bonding on structural strength	Section 7.11 Cessna 425
10. Drain holes are often essential for the airworthiness of an airplane. Make sure they are put in and kept open.	Section 7.12 Cessna 441
11. Place components which require frequent service/replacement so that no other components have to be removed for access	Section 7.13 Mc-Donnell F-4
12. Placards on access panels should make common sense	Section 7.14 Boeing 737-300
13. Design inspection covers so that maintenance personnel can open them while wearing gloves	Section 7.15 Boeing 747-200
14. Do not use materials with different electrolytic potential in a landing gear environment without corrosion protection	Section 7.16 Piper Aztec

15. Landing gears are subject to severe stress cycling. When using grit blasting techniques care should be taken to avoid local stress concentrations.	Section 7.17 Boeing MD-83
16. If the size or shape of a fuel pump is changed design the installation so that the wrong type cannot fit.	Section 7.18 Airbus A-340
17. Faulty structural repairs have been responsible for mishaps. Since aviation is a global affair, global standards and Processes should be considered for primary structural repairs.	Section 7.19 Boeing 747-200
18. It should not be possible for an airplane to be operated with fuel tank purge doors left open due to faulty maintenance	Section 7.20 Boeing 777

10.8 Lessons Learned from Aerodynamic Design (Chapter 8)

1. Tail sizes should be well established and verified before first flight.	Section 8.1 AVRO Tudor
2. In airplanes with reversible flight control systems rudder lock is a dangerous possibility. Consider a dorsal fin to prevent rudder lock.	Section 8.2 Bristol Freighter
3. When claiming commonality between two designs try to be realistic.	Section 8.3 Percival Jet Provost
4. It should be possible to recover from an upset into a deep stall condition. There are aerodynamic as well as systems based solutions. Each have their pros and cons.	Cass 8.4, 8.11 Gloster Javelin, Douglas A4D-1, BAC-111
5. Selecting the fuselage diameter of a new design is critical to its success.	Section 8.5 707/DC-8
6. Sizing an airplane to the requirements of one small customer is counter-productive in the world market place	Section 8.6 Vickers VC-10
7. In trainer aircraft spin recovery is a must. Spin strips can work wonders.	Section 8.7 Cessna T-37
8. Transonic aerodynamic center shifts must be accounted for in early design. The effect on trim and drag is considerable.	Section 8.8 Convair B-58, NAA XB-70A
9. Good looks in airplanes can really matter. Swept vertical tails have been very synergistic in this regard.	Section 8.9 Cessna 172

10. Transonic drag rise and flutter should be carefully looked at in early design.	Section 8.10 Convair 990
11. In planning a wind tunnel test do not skimp on the number of sideslip or angle-of-attack test points.	Section 8.12 NAA XB-70A
12. When considering a novel control configuration consider the effect on the total configuration.	Section 8.13 Boeing design study
13. A revolutionary configuration can yield large benefits in empty weight and in performance.	Section 8.14 Piaggio P-180
14. Verify that the horizontal tail is properly sized for take-off	Sections 8.15, 8.16 Boeing F-18, Bombardier CL-600
15. Do not forget momentum drag.	Section 8.17 NAA XFV-12A
16. The forward swept wing is a legitimate configuration choice in many new airplane types.	Section 8.18 Grumman X-29
17. Extreme performance requirements sometimes call for a novel design approach.	Section 8.19 Rutan Voyager
18. Local area ruling can be beneficial even at modest Mach numbers.	Section 8.20 Piaggio P-180
19. Closely coupled canard-tractor-propellers should be checked for approach to stall followed by rapid power application.	Section 8.21 PAT-1
20. Directional stability should be a requirement for all manned vehicles, including experimental types.	Section 8.23 Quad City Challenger

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Appendix A

It was stated in the Introduction that the assignment of a particular accident, incident or event to one of the seven categories detailed in Chapters 2 – 8 is sometimes arbitrary. The purpose of this appendix is to provide a key to how accidents, incidents and events relate with other categories. This is done in the form of the following table.

Chapter 2 Lessons Drawn from Operational Experience	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
2.2 Gust lock on during take-off	●		●		●		
2.3 Center of gravity too far aft I	●		●		●		●
2.4 Minimum un-stick speed I	●		●				●
2.5 Accidental retraction of landing gear during landing roll	●				●		
2.6 Cowl flaps left open during flap retraction	●		●		●		
2.7 Flight characteristics with one engine inoperative I	●		●				●
2.8 Tail stall in icing conditions	●		●		●		●
2.9 Unsweeping wings with flaps down	●				●		
2.10 Center of gravity too far forward	●		●				●
2.11 Loss of electrical power leading to loss of attitude instrumentation	●				●		
2.12 Flight characteristics with one engine inoperative II	●		●				●
2.13 Rudder system too complicated	●		●		●		
2.14 Minimum un-stick speed II	●		●				●
2.15 Electrical system failed during take-off emergency	●				●		
2.16 Aft center of gravity and stabilizer mistrim during take-off	●		●		●		
2.17 Reverse propeller mode in flight	●		●		●		●
2.18 Center of gravity too far aft and seat design	●	●					

Chapter 2 Lessons Drawn from Operational Experience	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
2.19 Icing on take-off I	●						●
2.20 Take-off without flaps	●				●		
2.21 Sterile cockpit and trim switch location	●		●		●		
2.22 Icing on take-off II	●		●				●
2.23 High speed descent to avoid icing	●				●		●
2.24 Center of gravity too far aft II	●		●		●		●
2.25 Center of gravity too far aft, over-loading and mis-rigged controls	●		●		●	●	●
2.26 Center of gravity too far aft and over-loading	●		●		●		●
2.27 Old habits return in emergencies I	●				●		●
2.28 Old habits return in emergencies II	●				●		

Chapter 3 Lessons Drawn from Structural Design	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
3.2 Fatigue failure of wing fitting I		●					
3.3 Fatigue failure of wing fitting II		●					
3.4 Canopy loads must be watched		●			●		●
3.5 Verification in structural design		●					
3.6 Fatigue failure due to pressurization cycles		●					
3.7 Vertical tail flutter		●					
3.8 Whistling swan downs Viscount	●	●					
3.9 A new flutter mode	●	●					
3.10 Corrosion fatigue		●				●	
3.11 Rear pressure bulkhead failure I		●			●	●	
3.12 Crack propagation I		●				●	
3.13 Horizontal stabilizer failure		●				●	
3.14 Elevator structural failure		●	●		●		●
3.15 Rear pressure bulkhead failure II		●	●		●	●	
3.16 Crack propagation II		●				●	
3.17 Cargo door hinge design		●					
3.18 Vertical tail fatigue due to vortex shedding		●					●
3.19 Design instructions ignored		●				●	

Chapter 4 Lessons Drawn from Flight Control System Design	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
4.2 Heat source close to flight controls			●		●		
4.3 Ailerons reversed I	●		●			●	
4.4 Gust lock engaged in flight	●		●				
4.5 Propeller blade severs controls I	●		●	●	●		
4.6 Design for one-way fit	●		●			●	
4.7 Ailerons reversed II	●		●			●	
4.8 Propeller blade severs controls II	●		●		●	●	
4.9 Elevator boost system bolt backs out	●		●			●	
4.10 Elevator control forces to overcome electric trim tab failure become too high	●		●		●		
4.11 Pitch trim failure reverses elevator speed gradient	●		●		●		●
4.12 Reversing polarity in a pitch damper	●		●		●	●	
4.13 Take-off with locked elevator I	●		●				
4.14 Elastic stop-nuts in flight control systems	●		●				
4.15 Controls jammed by foreign object	●		●				
4.16 Rudder fitting failure	●	●	●		●		
4.17 Locating flight control system cables	●	●	●				
4.18 Pilot induced oscillations	●		●				
4.19 Reversing polarity in a yaw damper	●		●		●		
4.20 Loss of control due to unwanted extension of ground and flight spoilers	●		●		●		
4.21 Take-off with locked elevator II	●		●				
4.22 Take-off with rudder and aileron controls locked	●		●				
4.23 Take-off with mis-trimmed stabilizer	●		●			●	

Chapter 4 Lessons Drawn from Flight Control System Design	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
4.24 Defunct elevator hard-stop	●		●				
4.25 Control system compliance	●		●				
4.26 One engine out control problem	●		●				●
4.27 Redundant system not redundant	●		●	●	●		
4.28 Uncommanded elevator travel	●		●			●	
4.29 Uncommanded roll at take-off	●		●			●	
4.30 Elevator trim tab failure	●		●			●	
4.31 The hard-stop which was not a hard-stop	●		●			●	
4.32 Jammed servo tab	●		●			●	
4.33 Unnecessary loss of control	●		●				
4.34 Frozen ailerons	●		●			●	
4.35 Mis-routing of control cables	●		●			●	
4.36 Water leaks do it again	●		●		●	●	
4.37 Uncommanded yaw	●		●				
4.38 Routing control cables past engine burst plane			●				

Chapter 5 Lessons Drawn from Engine Installation Design	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
5.2 Propeller too large or landing gear too short	●			●			●
5.3 Propeller reversal in flight I	●			●	●		
5.4 Propeller reversal in flight II	●			●	●		
5.5 Propeller reversal in flight III	●			●	●		
5.6 Exhaust fairing I				●			●
5.7 Propeller reversal in flight IV	●			●	●		
5.8 Propeller to fine pitch during approach	●			●	●		
5.9 Design for engine removal		●		●			
5.10 Flame-out due to engine mount compliance	●			●	●		
5.11 Engine bearing failure followed by propeller blade separation	●			●	●	●	
5.12 Whirl mode flutter	●	●		●			●
5.13 Adjacent engine installations				●			●
5.14 Exhaust fairing II				●			●
5.15 Propeller reversal in flight V	●			●	●		
5.16 Exhaust fairing III				●			●
5.17 Propeller blade separation I	●	●		●			
5.18 Propeller blade separation II	●		●	●	●		
5.19 Uncommanded propeller blade pitch reduction	●		●	●		●	
5.20 Uncommanded thrust reverser deployment	●		●	●	●		●
5.21 Power levers moved to beta range in flight I	●		●	●	●		
5.22 Uncontained engine failure I	●			●	●		
5.23 Propeller blade separation III	●		●	●	●		
5.24 Uncontained engine failure II	●			●			●
5.25 Uncommanded thrust reverser deployment II	●		●	●			●
5.26 Tire tread ingested into engine	●			●			●
5.27 Fuel line chafed through	●			●	●		
5.28 Involuntary engine shutdown	●			●	●		
5.29 Power levers moved to beta range in flight II	●			●	●		

Chapter 6 Lessons Drawn from Systems Design	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
6.2 Electrical system design I	●				●		
6.3 Fuel system and electrical system design	●				●		
6.4 Fuel vent design I	●				●		
6.5 Fire extinguishing system design	●				●		
6.6 Hydraulic system design I	●				●		
6.7 Design induced mistake I	●		●		●		
6.8 Service door fasteners	●		●		●		●
6.9 Design induced mistake II	●				●		
6.10 Firewall fuel shut-off valve cables in wheel well	●				●		
6.11 Hydraulic system design II	●		●		●		
6.12 Hydraulic system design III	●				●		
6.13 Fuel system design I	●				●		
6.14 Fuel vent design II	●		●		●		
6.15 Cabin door design I	●				●		
6.16 Fuel system design II	●				●		●
6.17 Cargo compartment light causes fire	●				●		
6.18 Cabin door design II	●				●	●	
6.19 Design for lightning strikes	●				●		
6.20 Fuel lines close to landing gear brace	●	●			●		
6.21 Loss of pitch control due to fire	●		●	●	●		
6.22 Confusing systems design	●				●		
6.23 System redundancy saves the day I	●				●		
6.24 Engine failure precipitates brake failure	●				●		
6.25 Systems design, flight crew training and improper maintenance procedures	●		●	●	●	●	●
6.26 Service lift design	●				●	●	
6.27 Leading edge slat asymmetry	●		●		●		
6.28 System redundancy saves the day II	●		●	●	●		

Chapter 6 Lessons Drawn from Systems Design	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
6.29 Flap asymmetry	●		●		●		●
6.30 Design of windshield washer system	●				●	●	
6.31 Three hydraulic system lines in the leading edge	●		●		●		
6.32 Leaks into the avionics bay I	●				●		
6.33 Nacelle cowl design and fuel filter cover design	●		●	●	●	●	
6.34 Ground spoilers deploy in flight	●		●		●	●	
6.35 Hydraulic system design problem	●		●		●		
6.36 Cabin door design III	●				●	●	
6.37 Landing gear actuator corrosion	●		●		●	●	
6.38 Leaks into the avionics bay II	●				●		
6.39 Fuel system design II	●				●		●
6.40 Landing gear door design	●				●		
6.41 Moisture ingress I	●				●		
6.42 Electrical system design II	●				●		
6.43 Icing of stall warning system	●		●		●		●
6.44 Moisture ingress II	●				●		
6.45 Flap/slat system design	●		●		●		
6.46 Galley chiller fan blade and wiring failure causes in-flight fire	●		●		●		

Chapter 7 Lessons Drawn from maintenance and Manufacturing	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
7.2 Propeller blade separation in flight	●			●	●	●	
7.3 Elevator control bolt backed out I	●		●			●	
7.4 Elevator servo tab bolt backed out	●		●		●	●	
7.5 Engine maintenance error	●			●		●	
7.6 Propeller reversal in flight	●			●		●	
7.7 Landing gear truck beam failure	●				●	●	
7.8 Elevator control bolt backed out II	●		●			●	
7.9 Loss of roll control	●		●			●	
7.10 Quenching		●				●	
7.11 Weight control		●				●	
7.12 Incomplete skin bonding		●				●	
7.13 Drain holes forgotten	●		●			●	
7.14 Maintenance manhours per flight hour	●				●	●	
7.15 Placards on inspection covers	●				●	●	
7.16 Inspection covers not large enough	●				●	●	
7.17 Landing gear corrosion		●			●	●	
7.18 Grit blasting	●	●			●	●	
7.19 The wrong hydraulic pump	●		●	●	●	●	
7.20 Faulty structural repair	●	●				●	
7.21 Fuel tank purge door left open	●				●	●	

Chapter 8 Lessons Drawn from Aerodynamic Design, Configuration Design and Aircraft Sizing	2. Operational	3. Structural	4. Flight Controls	5. Engine Install.	6. Systems	7. Manuf. & Maint.	8. Aerod. & Config.
8.2 Empennage changed due to insufficient longitudinal and directional stability	●		●				●
8.3 Dorsal fin suppresses rudder lock	●		●				●
8.4 Commonality lost						●	●
8.5 Deep stall I	●		●				●
8.6 Fuselage width	●						●
8.7 Sizing an airplane for one customer	●						●
8.8 Spin strips	●		●				●
8.9 Transonic aerodynamic center shift	●		●				●
8.10 Swept vertical tails on a propeller driven airplane		●	●				●
8.11 Transonic drag I	●						●
8.12 Deep stall II	●		●				●
8.13 Snaking oscillation due to local directional instability	●						●
8.14 Aileron reversal due to tail interference							●
8.15 From VATLIT to Avanti							●
8.16 Horizontal tail sizing I	●		●				●
8.17 Horizontal tail sizing II			●				●
8.18 The XFV-12							●
8.19 Do forward swept wings make sense?	●						●
8.20 Unique solution to an extreme range requirement							●
8.21 More examples of area ruling	●						●
8.22 Canard with close coupled propeller	●		●				●
8.23 Insufficient geometric dihedral	●		●				●
8.24 Directional stability should be required	●		●				●

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