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Auxiliary Power Systems

NORTH ATLANTIC TREATY ORGANIZATION

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NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.352

AUXILIARY POWER SYSTEMS

Papers presented at the Propulsion and Energetics Panel 61st (B) Specialists' Meeting, held in Copenhagen, Denmark, 30 31 May 1983.

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THEME

One of the characteristics of modern high-performance aircraft is their high demand for power for electrical, hydraulical or pneumatic subsystems. On board auxiliary power systems are installed to fulfil this demand. In past years, the design of these systems was primarily determined by the increasing power levels required. For the future, new requirements concerning the economics of auxiliary power generation and the continuous availability of auxiliary power have grown out of the general fuel situation as well as of the advent of new technologies, like fly-by-wire and active control technology. The latter also underline the need for provision of emergency power. The increasing use of electronics and avionics on board aircraft gives rise also to increasing cooling requirements, which must be taken care of by auxiliary power systems.

For the provision of auxiliary power various technical systems have been developed and new solutions discussed. It was the purpose of the meeting to review the current state-of-the-art, to exchange experiences, and to discuss future problems of auxiliary power generation.

Les avions modernes à hautes performances sont caractérisés, entre autres, par un besoin considérable en énergie pour les sous-systèmes électriques, hydrauliques ou pneumatiques. Des groupes moteurs auxiliaires sont installés à bord pour répondre à ce besoin. Jusqu'ici, ces groupes moteurs étaient essentiellement conçus en fonction des niveaux de puissance accrus qui étaient requis. Cependant, des impératifs nouveaux en ce qui concerne l'économie de la production de puissance auxiliaire et la disponibilité constante de cette puissance sont apparus; ils résultent de la situation générale au plan des carburants ainsi qu l'apparition de technologies nouvelles telles que le pilotage par fil et le pilotage actif. Ces dernières technologies font également ressortir la nécessité de la fourniture d'une énergie de secours. L'emploi de plus en plus étendu d'équipements électroniques à bord des avions se traduit également par un besoin accru en systèmes de refroidissement, auquel doivent pourvoir des groupes moteurs auxiliaires.

Divers systèmes techniques ont été mis au point pour la fourniture d'énergie auxiliaire, et des solutions nouvelles étudiées. Le but de cette réunion était de présenter l'état de l'art dans ce domaine, de susciter des échanges touchant l'expérience acquise, et d'étudier les problèmes que posera, à l'avenir, la production de puissance auxiliare.

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CONTENTS

	Page
THEME	iii
PROPULSION AND ENERGETICS PANEL	iv

	Reference

SESSION I – REQUIREMENTS FOR AUXILIARY POWER SYSTEMS

AUXILIARY POWER REQUIREMENTS. THEIR ROLE IN AIRCRAFT PERFORMANCE by E.Eckert	1
GROUPES AUXILIAIRES A TURBINE A GAZ par A.Romier et F.Calmels	2
SECONDARY POWER SYSTEMS FOR FIGHTER AIRCRAFT. EXPERIENCES TODAY AND REQUIREMENTS FOR A NEXT GENERATION by W.Hausmann, M.Pucher and T.Weber	3
SMALL AUXILIARY POWER UNIT DESIGN CONSTRAINTS by C.Rodgers	4

SESSION II – PROVISION OF AUXILIARY POWER

APU IN COMMERCIAL AIRLINE OPERATION by H.W.C.L.Schoevers and A.Booy	5
AUXILIARY POWER UNITS FOR WIDE-BODY AIRCRAFT by S.Riazuelo, M.Cenival and M.Eglem	6
SECONDARY POWER SUPPLIES FOR SMALL SINGLE ENGINED COMBAT AIRCRAFT by G.Jones	7
GENERATION HYDRAULIQUE DE SECOURS DU MIRAGE 2000 par P.Atlan	8
OPTIMISATION OF ENGINE POWER OFFTAKE BY SECONDARY POWER SYSTEM DESIGN by J.R.Hopkins	9
HOT GAS APU STARTER FOR ADVANCED AIRCRAFT APPLICATIONS by B.W.Schmitz, L.D.Galbraith, W.E.Jorgenson and D.A.Pahl	10
A JET FUEL STARTER FOR LOWEST SYSTEM LIFE CYCLE COST by R.Smith	11
COST EFFICIENT ON-BOARD POWER FOR AIRLINE OPERATION by K.Mose	12
APU OPERATIONAL EFFICIENCY by W.Hoose	13
GROUND AND INFLIGHT OPERATIONAL EFFECTS OF APU's by J.H.Renken	14

SESSION III – ELECTRICAL/HYDRAULI[illegible]AL SYSTEMS/COOLING OF EQUIPMENT

SUPER INTEGRATED POWER UNIT FOR FIGHTER AIRCRAFT by A.D.Lucci, J.A.Williams, E.C.Beder and B.L.McFadden	15

	Reference
'ALL ELECTRIC' ACCESSORY DRIVE SYSTEMS: IMPLICATIONS ON ENGINE DESIGN AND PERFORMANCE by G.B.Toyne and G.S.Hodges	16
IMPLEMENTING MICROPROCESSOR TECHNOLOGY IN AIRCRAFT ELECTRIC POWER GENERATING SYSTEM CONTROL by G.Ruffner, B.Mehl and S.Lorenz	17
400 Hz GENERATORS EVOLUTION, EFFECTS ON INSTALLATION ARRANGEMENTS by C.Jacques	18
CHAIRMAN'S SUMMARY	CS

AUXILIARY POWER REQUIREMENTS
THEIR ROLE IN AIRCRAFT PERFORMANCE

AD P002391

by

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Summary

Based on the historical development and the actual status of requirements, the trends of essential parameters are analyzed, physical and technical possibilities are assessed and a prognosis is offered.

Increasing demands for secondary power for subsystems and more powerful engines have led to stronger secondary power components, thus creating a mass problem. Energy losses add a thermal problem. Complexity rises rapidly with the number of inputs and outputs. All of this, together with such aspects as redundancy, reliability, safety, maintainability and life cycle cost, composes today's auxiliary power system.

The future system, in response to high performance requirements and budgetary and personnel restrictions, should aim at less complexity and more efficiency.

1. ENERGY IN AIRCRAFT

If we regard all sources of stored energy aboard a contemporary combat aircraft, we can identify about two dozens of different types of them. The largest by far, of course, is the main fuel system destined to feed large and powerful engines in order to achieve good performance characteristics, which usually means a high thrust/weight ratio. The energy converted for the latter purpose is by magnitudes higher than that required for the different subsystems which, on their part, are powered by the secondary power system drawing its energy in flight from the main engines, whose operation as a propulsive force for ground operations involves a tremendous waste.

A modern military aircraft is expected to have a total lifetime of 150 000 to 200 000 hours; of this time, some few percent are airborne, while another percentage in the order below 10 refers to ground operation.

For some thousand hours, energy in different forms must be provided. If there is a requirement for autonomous operation, the most reasonable energy source is main fuel, in terms of logistics as well as of cost.

Today's tool for conversion of main fuel energy into usable energy forms, e.g. shaft power to produce electrical current or pressure air for cooling and other purposes, is a small gas turbine, increasingly found in advanced aircraft systems since about the sixties, as the operation of a main engine, equivalent to about 6 000 kW - even in "idle" status - means very poor efficiency in the magnitude of 2 percent.

2. HISTORICAL DEVELOPMENT

A glance into history shows the advance of flight vehicles under different aspects, such as

- speed (note the need for power-aided and non-linearily connected controls), Fig.1
- dimensions and masses (requiring "aids" for the human sensors),
- altitude (note the number of changing parameters up to detrimental/perilous states),
- adverse weather operation (requiring a controlled environment for crew and instrumentation),
- capability for aerobatic operations,
- reliability (flights over sea or polar regions).

The effects of this development on equipment and, consequently, on its power demand have been tremendous:

2.1 Fuel System

Increasing engine power and flight duration and, especially, the jump to jet engines have led to considerable fuel loads. Today's combat aircraft are rated to carry internal fuel of about a quarter of their T.O. mass. Even when the stowage problem is solved, changes in weight/moments of inertia of this magnitude result in problems of C/G, of ground and airborne stability. Inevitably, partition of the fuel mass follows, resulting in a complex fuel system with a number of redundant pumping, fuel gauging, level sensing, switching, supervising, inerting, pressurizing, draining, dumping etc. equipment, a lot of which requires energy. Modern engines need some kg/s of fuel; accordingly, the power demand of the diverse fuel pumps is in the range of some 10 kW per engine, the load being about constant (except for afterburner operation) and connected to the electrical and mechanical drive subsystem.

The mass of the fuel system relative to the empty weight is about 2 ... 3 % both for combat aircraft and helicopters. The tendency is growing in proportion to engine demands.

2.2 Hydraulic System

Hydraulic systems made a fairly late appearance in aviation. Their increase and sophistication is quite obviously connected to the advance of aircraft in the areas of speed, dimension, mass (see above), automatic flight control and critical systems design as Controlled Configured Vehicles (CCV), with the need for high redundancies. The power requested is about 15 ... 20 % of a total SPS-gearbox power, i.e. somewhere between 8 ... 40 kW, showing a rapidly growing tendency, as this subsystem is directly connected to the operational performances of the aircraft. The load during a mission is greatly varying. All of the hydraulic power is finally converted into waste heat, distributed by the system fluid and dissipated by the components and their installation. Fig.2 Fig. 3

2.3 Electrical System

Electrical energy has been aboard since powered flight began, starting with negligible power levels for ignition and followed by ever more diverse applications. Thus in the years before World War II, electrical systems resembled those of motor vehicles that were fed by DC generators, the latter, in turn, being driven by air turbines or from the engine(s). Ever higher, faster and farther flying aircraft made it necessary to install ever more equipment to grant functionality, comfort and safety for man; to provide for regulation and control, navigation, communication, supervision, lighting and deicing, in addition to all sorts of sensors and weapons. There is hardly anything without electrical connection. Fig.4

The necessary power could no longer be generated and distributed as direct current. Today's heavy combat aircraft in general have alternators, three phase, 400 Hz, with integrated or preceding constant speed drive (usually a hydraulically controlled planetary gear), distributing their power via a bus system into a widely ramified network. Bus systems have been used in this context for years. The electrical system - as the other systems - must be designed for peak loads, which occur only sporadically or never during missions (e.g. deicing). It is an interesting question, whether the weight necessary for this design case can be diminished when a continuously operating APU is a constituent part of the secondary power concept so that the total weight will benefit. Fig.5

The electrical power generation rating is a good indicator of the complexity and all-weather capability of an aircraft and usually a clue for any significant secondary power installation. The power required in military aircraft is about 30 % of a total secondary power system gearbox, with generators rated somewhere between 20 kVA ... 60 kVA each; however, this is by far exceeded in special aircraft or large transporters (e.g. Advanced Airborne Command Post total generator rating 8 x 150 = 1,200 kVA), and usually there are at least two generators. Fig.6

Helicopters (small) provide DC generation of about 2 x 3 kW up to 2 x 40 kVA (large) and more. A special function is engine starting, which is found essentially in light attack A/C and helicopters. If the electrical system is a DC system, a combined starter/generator will be used most probably.

Almost all of the electrical power is finally converted inside the aircraft into waste heat and must be carried overboard

- with the aid of new energy,
- within fairly narrow temperature and other limits, at least as far as crew and electronics are concerned; but even for general equipment, Arrhenius' Law points to the benefit of moderate temperatures.

Another interesting relationship with thermal housekeeping and the mass problem should be indicated: The electrical system can be considerably overloaded for short periods, the main parameter in that case being temperature build-up/cooling.

Related to the empty weight, the electrical systems proportion is between 1.8 ... 3.7 %, whilst in a medium transport helicopter this rises to about 7 %. The tendency is growing.

2.4 Pressure Air and Cooling Air

Pressure air for the air supply and environmental control system is usually tapped from one or more of the engines compressor stages in quantities of about .1 ... 1 kg/s and some 100 degrees C. This bleed air is cooled by a ram air or ejector operated cooler and then further processed, e.g. in cooling air turbine, additional ram air/ejector assisted cooler(s), water extractors, and then distributed with the aid of a variety of valves, chokes, regulators, sensors for pressure, temperature etc. It is used, according to its degree of preparation, for

- "air cooling" for the crew (e.g. foot or body spray),
- operation of pneumatic drives and devices,
- supply for anti-g-suit,
- canopy sealing or other movable parts/slot sealing,

- rain removal system (blow upon wind shield/front screen),
- canopy pressurizing and demisting,
- cooling/air conditioning of avionic, equipment gun and RADAR bays,
- supply of compressed air to external loads (pods, tanks).

As this is an enormously complex task, the system must be equally complex. Hardly anything is constant, neither the input air (varying in temperature, humidity, composition, pressure), nor the other environmental conditions (heat rejection or reception of the environment, depending on speed, altitude and other parameters), heat rejection of the equipment (variety of operating condition and times), different temperature gradients, heat conductance and much more. For illustration:
At 12 km altitude, the cooling air requirement is about 4, at 18 km about 10 times as high as at S.L.

Some basic figures for demonstration of the requirements:
Electronics with less than 6 mW/cm3 heat rejection and without "hot spots" need no special cooling, those with more than 18 mW/cm3 need forced air cooling; when more than 120 mW/cm3 are expected, air cooling is no longer sufficient.

During ground operation, a modern fighter's need for cooling air amounts to 10 kg/min; if that is supplied by small inlets, a noise problem for the ground crew will occur.

The power required for the air supply/environmental control system is considerable, ranging from about 70 kW in small combat aircraft to some 100 kW.

As this air is bled from the engine(s), the energy by-passes the mechanical part of the secondary power system. Relative to the aircraft empty weight, the air systems proportion is between 1.3 ... 1.7 %, in medium transport helicopter about 1 %.

Another remarkable feature is that the airstream can carry all sorts of contamination to every corner in the aircraft, cockpit included, be it oil dust or things related to A, B, C warfare.

The air and cooling subsystem is very closely related to equipment and performance characteristics and, therefore, a design-dominating system, which cannot be stealthily eluded. The trend is: Fairly rapidly growing system as the last resort for removal of waste heat from all other systems.

2.5 Power off-take

The requirements for mechanical/pneumatic power must be satisfied Fig.7

- by the engine: during all mission phases,
- from ground sources; precondition: aircraft at adequately equipped and functioning sites,
- from aircraft internal power source other than main engines.

Re dash No. 1: Fig.8

Power off-take, even if only in the percentage region of an engine's power, can lead to or favour engine operating problems in some areas of the flight envelope and during critical flight manoeuvers (engine surge/stall with possible flame out).

Re dash No. 3:

The only eligible source could be a battery. Batteries for military aircraft including helicopters range between 15 Ah ... 40 Ah, energetically equivalent to a kerosene mass in the magnitude of 100 g. Formerly sufficient for checks of moderate duration, they cannot be used in most cases today for various reasons:

- DC most unsuitable to feed a three-phase current network;
- electrical battery power would be consumed in a very short time;
- no cooling air, electronics would be quickly damaged;
- no power/no operational condition/no test practicable for hydraulic, pneumatic, mechanical, oil systems etc.

Thus, several unfavourable trends add together:

A lot of equipment means a lot of (pre-flight) testing, requiring electrical and other forms of energy and implying large generators and other power drives. All these mechanical loads are connected to the shaft of the engine, thus aggravating the already difficult task of starting a large engine, or they are connected via clutches and control mechanisms, thus contributing to complexity. As a rule of thumb, the starting power is about 1 % of the engine compressor drive input at rated rpm or, even more roughly, dry thrust/daN x 2 ... 3 = starter rating/kW.

As the trend in engine power (e.g. expressed by thrust/engine weight) is expected to increase and as this will be accompanied by higher pressures, one may expect starting power to increase as well. Fig.9

3. ACTUAL REQUIREMENTS AND THEIR REALIZATION IN TERMS OF HARDWARE

3.1 Requirements

Actual requirements are governed by the effects of a tremendous increase in equipment, especially in electronic equipment. This is due to the fact that enhanced sensing, regulation and control is a prerequisite to optimally adapting a component or system to all relevant operating conditions, thereby increasing its effectiveness. The rapid advances in electronics, data processing and micro-miniaturization allow almost every imaginable parameter to be sensed, processed and combined with others. Fig.10

Inherent in these overwhelming possibilities, however, is the danger that requirements may be established as an end in themselves.

A further danger is that the well-meant realization of all possible advances, added together, may not unconditionally lead to higher weapon system effectiveness. The latter may be seen as

- a capability to assert oneself against an enemy in a realistically envisioned scenario;
- a deterrent potential realized in the form of a continously available man-machine weapon system, whose performance criteria surpass those of the corresponding system on the opponent's side, and which should be achieved at minimum possible cost.

Given the aforementioned reservations, the list of actual requirements could read as follows:

3.1.1 Sufficient electrical and hydraulic power (possibly cooling supply) for
- subsystem check
- ground servicing.

3.1.2 Sufficient electrical and hydraulic power (possibly cooling and environmental control system supply) for operational stand-by.

3.1.3 Continous APU operation during aircraft and armament turn-around.

3.1.4 Capability to maintain for hours a ready-for-take-off status (within a few minutes).

3.1.5 Capability to serve as an emergency power unit for
- restarting the main engine(s)
- feeding the absolutely necessary subsystems/components with
  - -- hydraulic and/or
  - -- electrical power

  to permit controlled flight or to establish the parameters for ejection-seat operation.

3.1.6 Capability to allow self-contained start of the aircraft.

3.1.7 Capability to allow start with external power (e.g. electrical or other power).

3.1.8 Autonomous oil system.

3.1.9 Use of main engine fuel (primary/alternative/emergency fuels may be required).

3.1.10 Oil type same as for main engine.

3.1.11 A/C can move X sec. after APU starting initiation.

3.1.12 During engine starting:
sufficient electrical power/cooling supply for aircraft operational stand-by.

3.1.13 Capability for some subsequent main engine starting cycles in a given time and environment.

3.1.14 Capability to relieve engine(s) of the burden of electrical, hydraulic etc. drives in very critical flight status (e.g. helicopter hovering at low altitude, at sea etc.).

3.1.15 Vulnerability requirements.

3.1.16 Capability to start the APU manually (e.g. via hydraulic hand pump).

3.1.17 APU/EPU combination for either

- air-breathing mode (jet fuel combustion for ground power delivery inclusive engine start)
- gas generator system for emergency power (hydrazine or LOX with JP-4) to be independent of altitude limitations.

3.2 Past and Present Hardware Solutions

Since more than half a century, there have been requirements and solutions for auxiliary power aboard aircraft. Examples for solutions:

3.2.1 Airships, e.g. Zeppelin LZ 127, in the twenties, carrying a spark ignition engine coupled with an electrical generator of about 1.5 kW, serving as "APU" for three ram air driven electrical generators; LZ 129 in the thirties, equipped with two Diesel engines of 33 kW each, generating AC and DC.

3.2.2 Flying boats/seaplanes as DO 24 in the late thirties, furnished with a spark ignition engine combined with an electrical generator of about 1 kW, supplying auxiliary power for stand-by/lay-days/main engine start.

3.2.4 Jet fighters Me 262 JUMO 004-engine starting facility, provided by a two-stroke engine of lawn-mower type, housed in the front hub and swung manually.

3.2.5 The APU in its contemporary meaning, developed as a small gas turbine for transport aircraft C 130 in the fifties.

3.2.6 The APUs for V/STOL aircraft, such as the HARRIER and VAK 191 B, the latter supplying shaft power for an alternator 15/20 kVA and a hydraulic pump as well as bleed air up to 5 kg/min for electronic equipment cooling and tank pressurization during ground operation and operation throughout the flight envelope as well including in-flight start by battery or windmilling. As the sum of all possible functions exceeds the APU's rating, an automatic load sequence switching is provided.

3.2.7 APU in A-10 as an example for a ground attack aircraft that operates at a critical low altitude.

3.2.8 APU in special aircraft with a large equipment share, such as Maritime Patrol Aircraft (MPA):

- Breguet Atlantic ANG
- Nimrod

  or
- E-3 Sentry as an example for C3 aircraft

  or
- tankers, such as
- VC-10
- KC-135 R (two APUs).

3.2.9 A prototype of an APU/EPU independent of altitude and flight condition according to the SIPU concept.

4. TRENDS OF ESSENTIAL PARAMETERS

It is evident from the preceding paragraphs and slides that the power demand for electrical, hydraulic, air, fuel and other subsystems is rising; so is the demand for removal of equally increasing amounts of waste heat and so is the demand for starting power for engines. Fig.11

Complexity, defined as the number of many "items" n and the maximum of their mutual connections

$$C \approx \frac{n^2}{2}$$

rises.

This, in turn, causes the cost to rise, as it is a function of single parts' cost plus cost for all connections between all "items" (comprising these parts plus all related quantifiable aspects)

$$\text{Total cost} \approx \sum_{i=1}^{n} (\text{parts cost})_i + \sum_{j=1}^{\frac{n^2}{2}} (\text{connections cost})_j$$

where the first term reveals the cost sum of all the parts of a W/S and the second term reveals the cost sum of all interconnected "items".

It is easily deducible that the whole is much more than just the sum of its parts, the second term illustrating

- the lion's share in cost and time,
- the danger of false assessments and loss of control,
- the increase in secondary power demand and waste heat problem, as far as energy flow/energy conversion between "items" in this dense network is concerned.

Complexity, while unavoidable to optimize an aircraft with respect to achievable goals, is contradictory to the wish to develop and operate aircraft on reliably pre-plannable cost basis.

How can we get out of this dilemma?

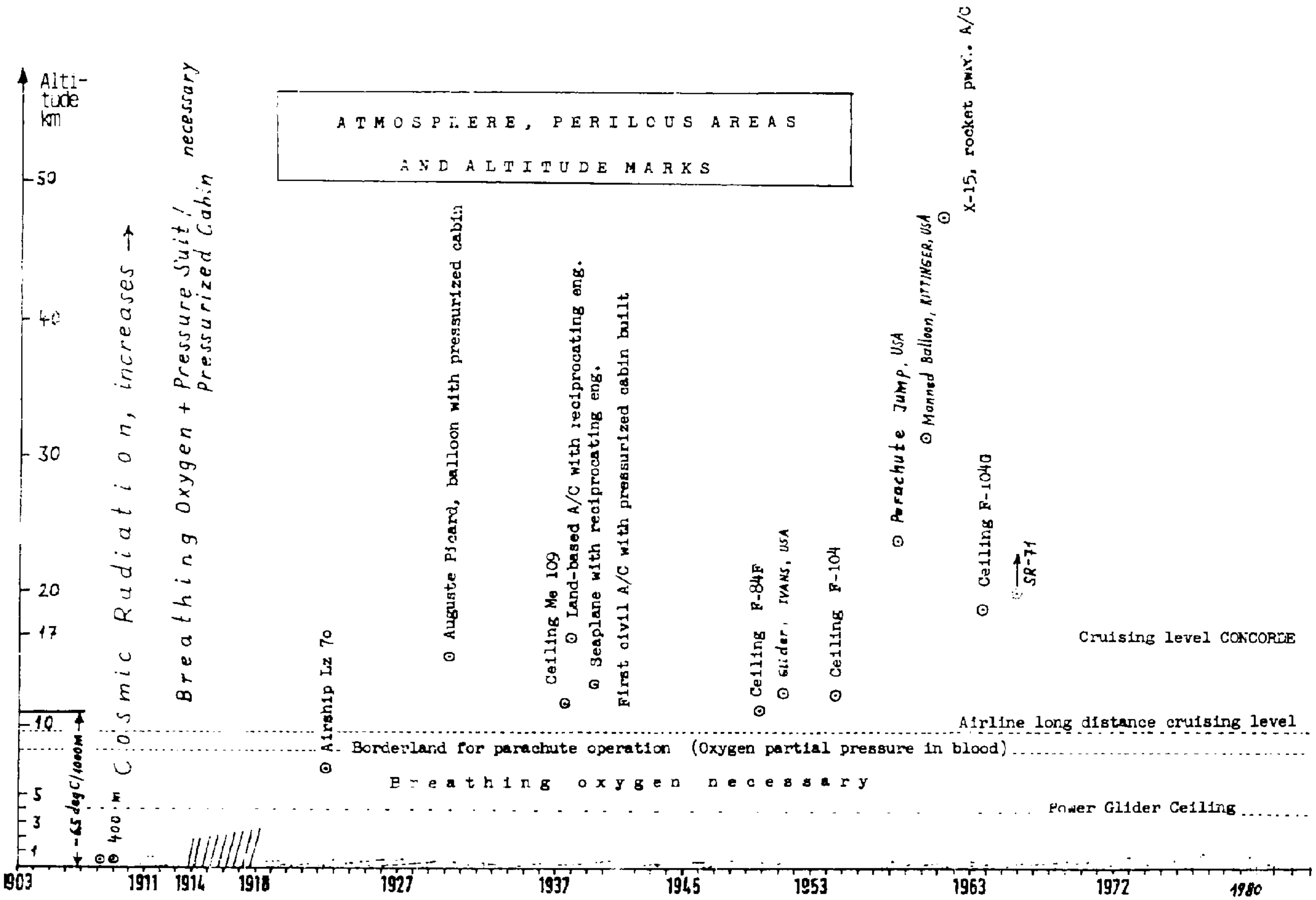
Evaluation of the preceding formula, especially with respect to power system and auxiliary power, shows that future fighter aircraft of

- small,
- light, and consequently
- single seated,
- single engine design

would ease the problem, provided that

- a reliable and modern engine can be chosen,
- this engine is assisted by an adequate airborne operated APU,
- the thermal housekeeping of the aircraft is as thoroughly designed and optimized as that of satellites.
- an emergency power facility (e.g. booster rocket) for short-time emergency thrust is available

consequently, this dictates the renunciation of technical revolutions, as the flippant requirement for the development of completely new, large and super-complex aircraft, but recommends the stepwise evolution of recognized satisfactory systems to even better ones.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Borderland for parachute operation (Oxygen partial pressure in blood)
Breathing oxygen necessary
Power Glider Ceiling

Figure 1

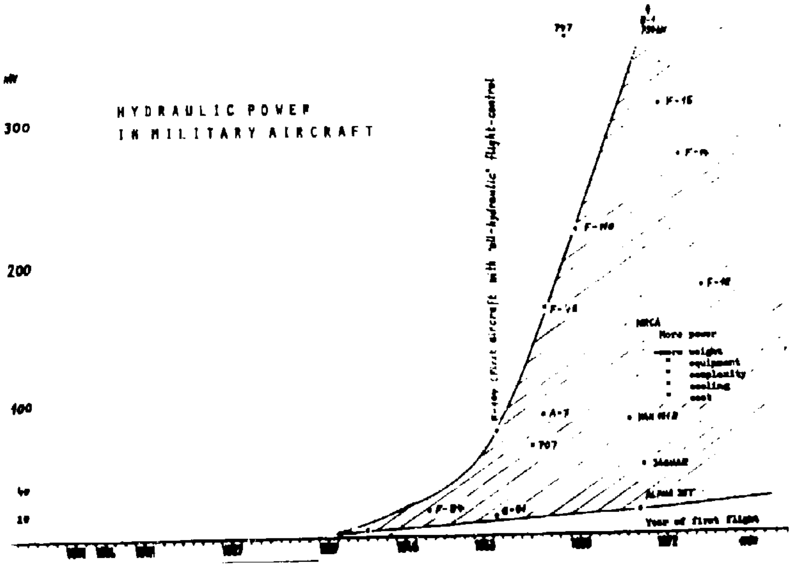


Figure 2

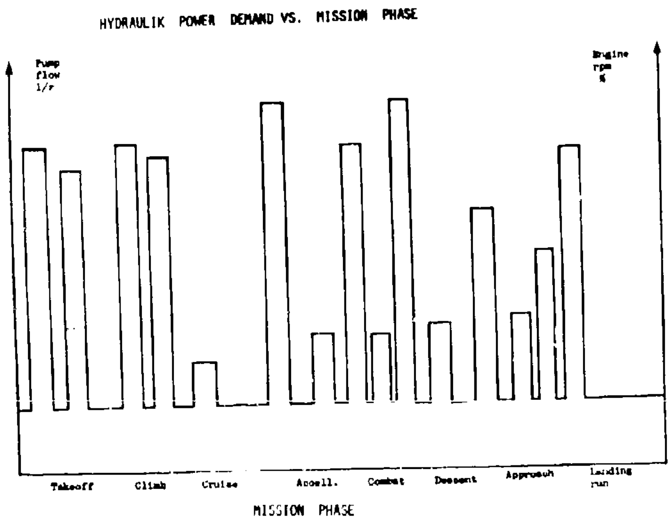
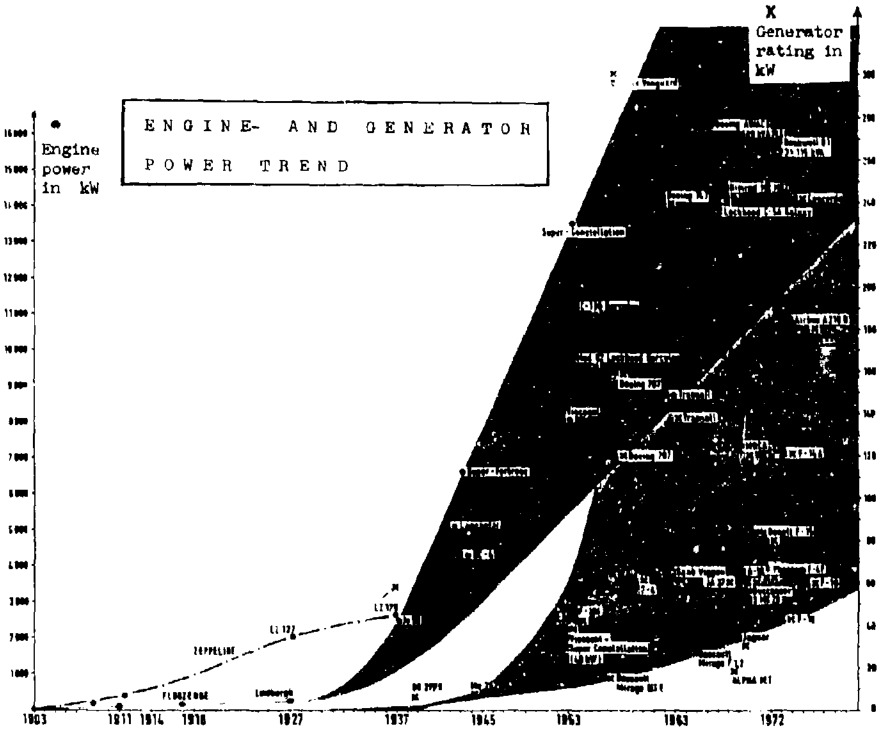


Figure 3



.... AGE OF ELECTRON TUBES ....

.... AGE OF TRANSISTORS ...

.. IC's ....

.. μP's .....

Documentation,Description :	10...20 Data	ca. 6 Pages		some 1000 Pages
Deg. of Integration :				doubles all 2 years
Price per bit :				falls...falls...falls...
Deg.of Integr. x Price p.bit :				about constant
Software/Hardware :		1/10	1/1	100/1
No. of logic funct. per chip :	about linear increase with time			~1 ca. 256k

O. LILIENFELD: Crystal amplifier patents; MOS-principle

BARDEEN + BRATTAIN: Transistor invented

IC's

μP's

Figure 4

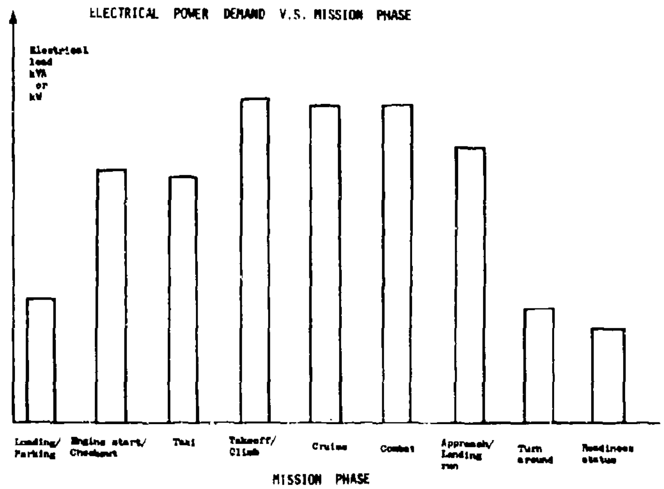


Figure 5

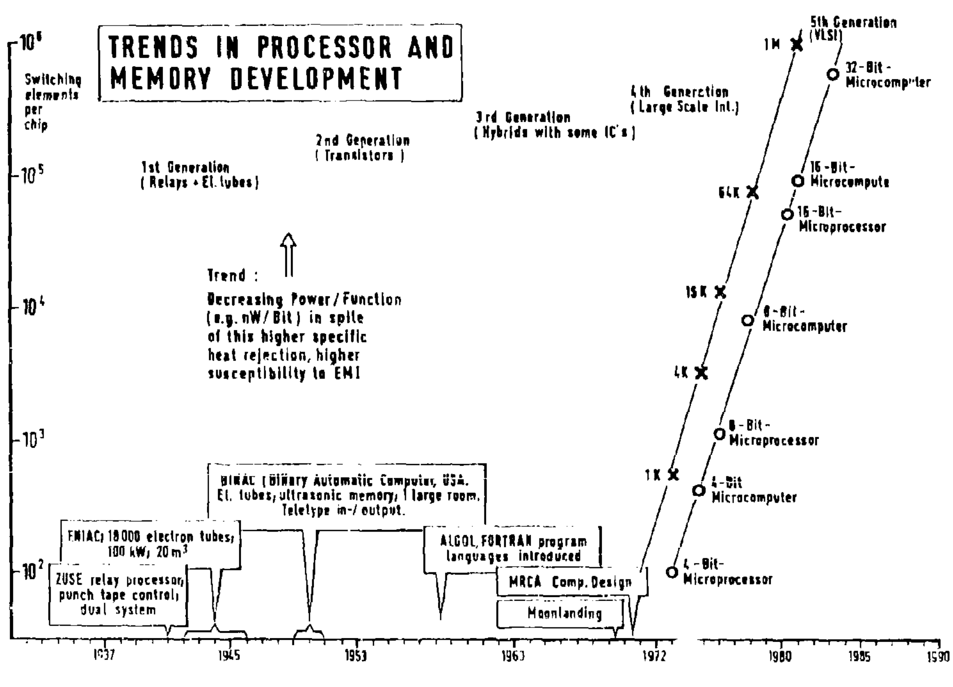


Figure 6

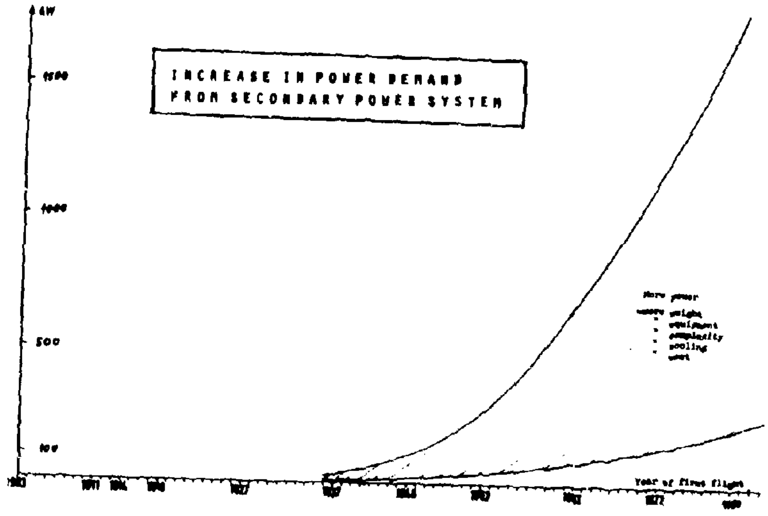


Figure 7

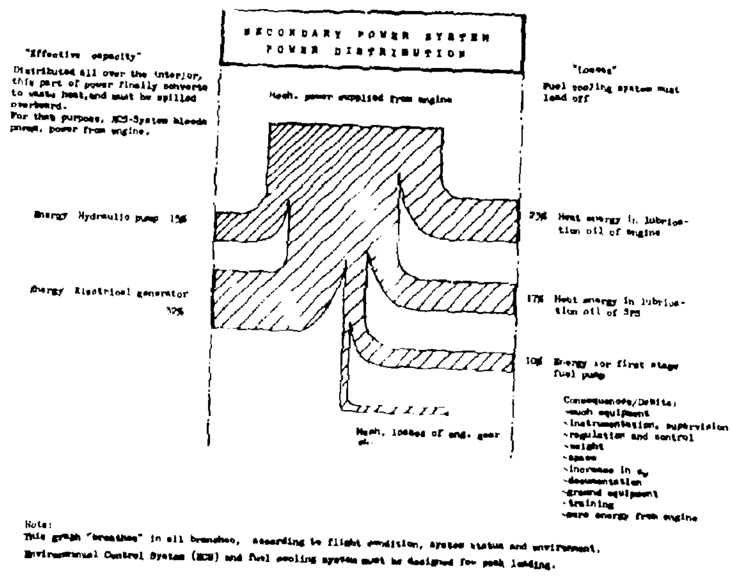


Figure 8

TREND OF COMBAT AIRCRAFT

ENGINES THRUST : WEIGHT-RATIO

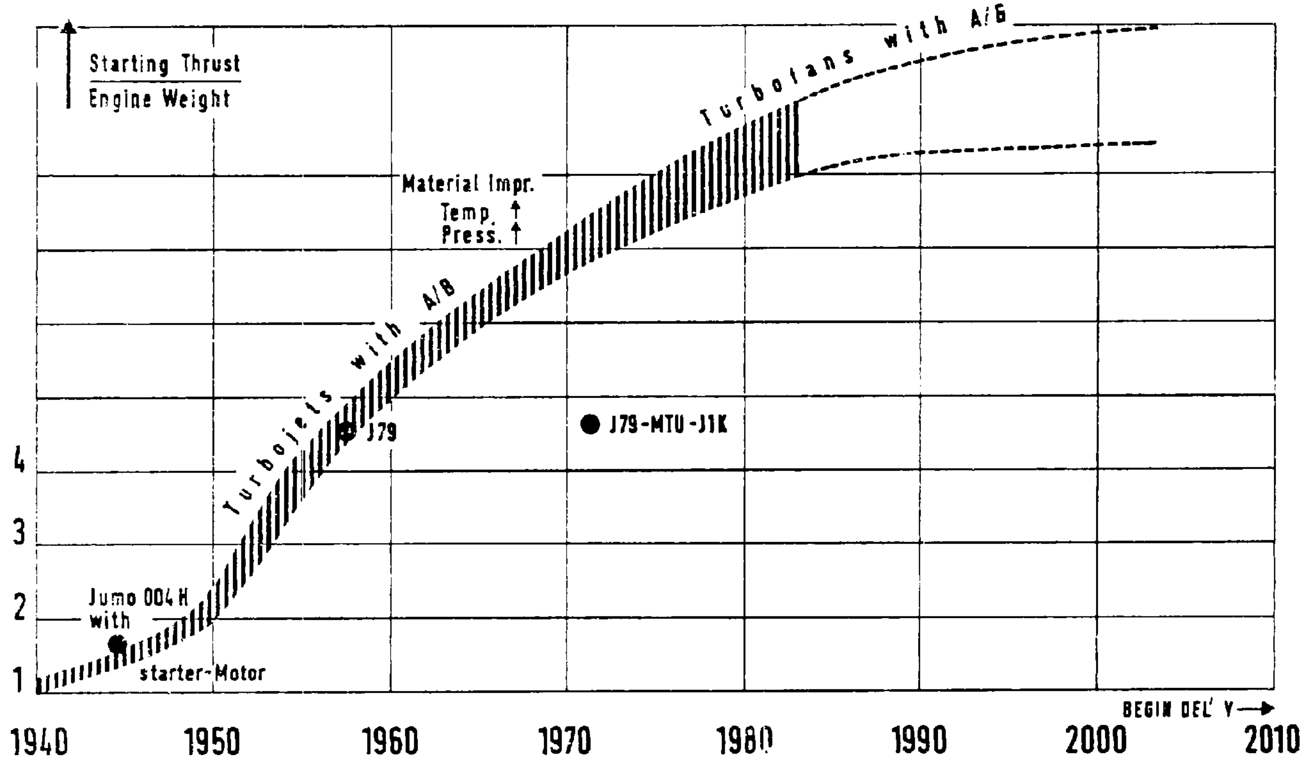
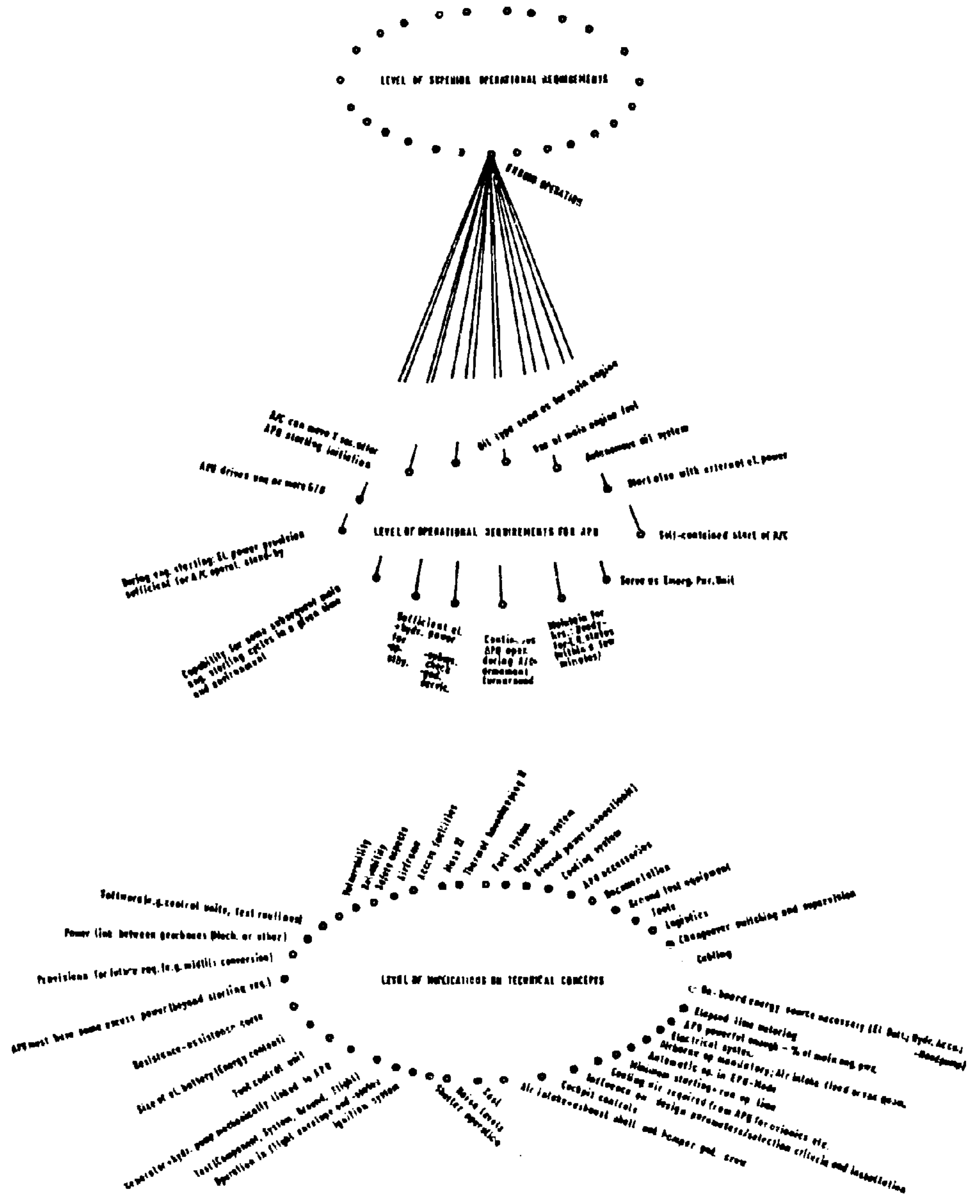


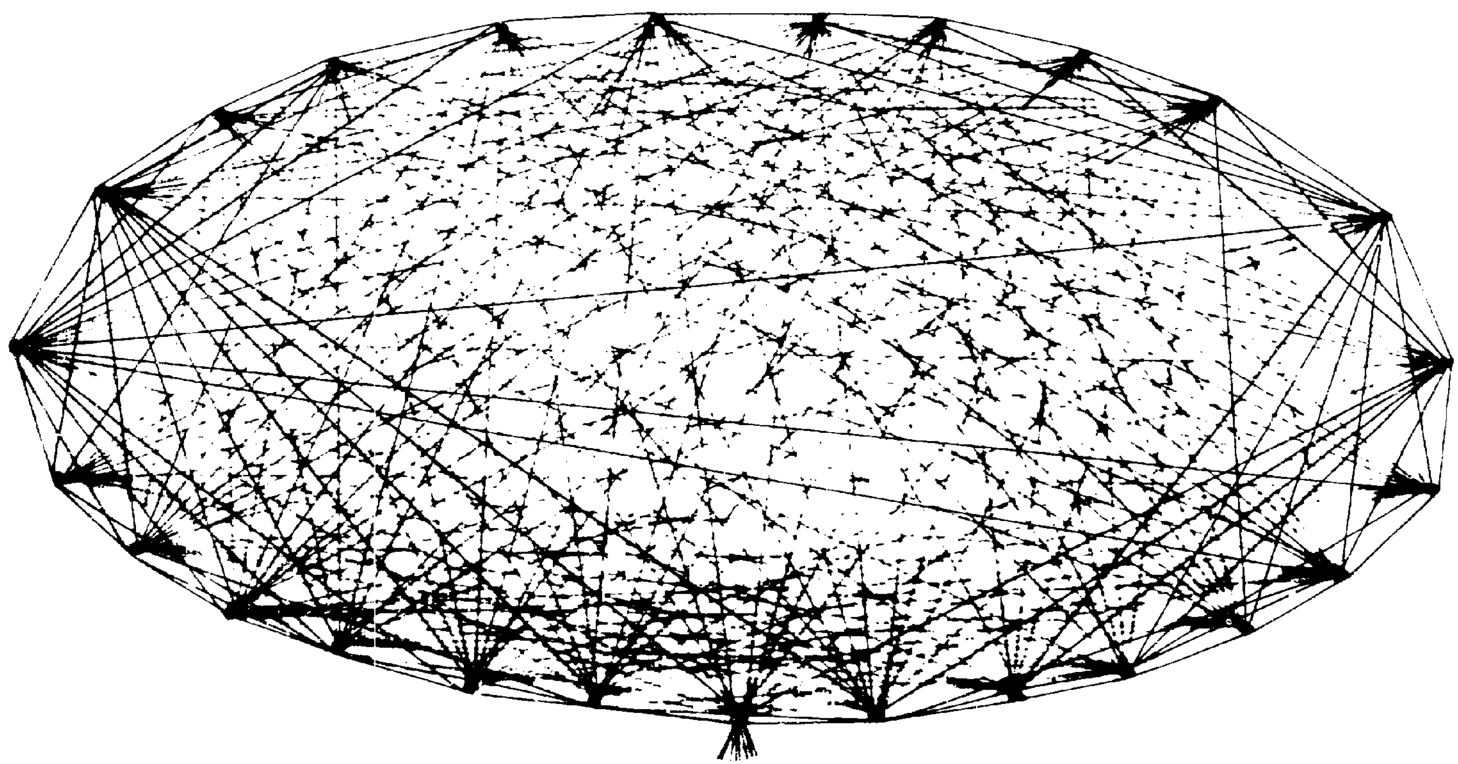
Figure 9



1) An increase in 1 kg payload mass results in an increase of about 5kg in T.O. mass

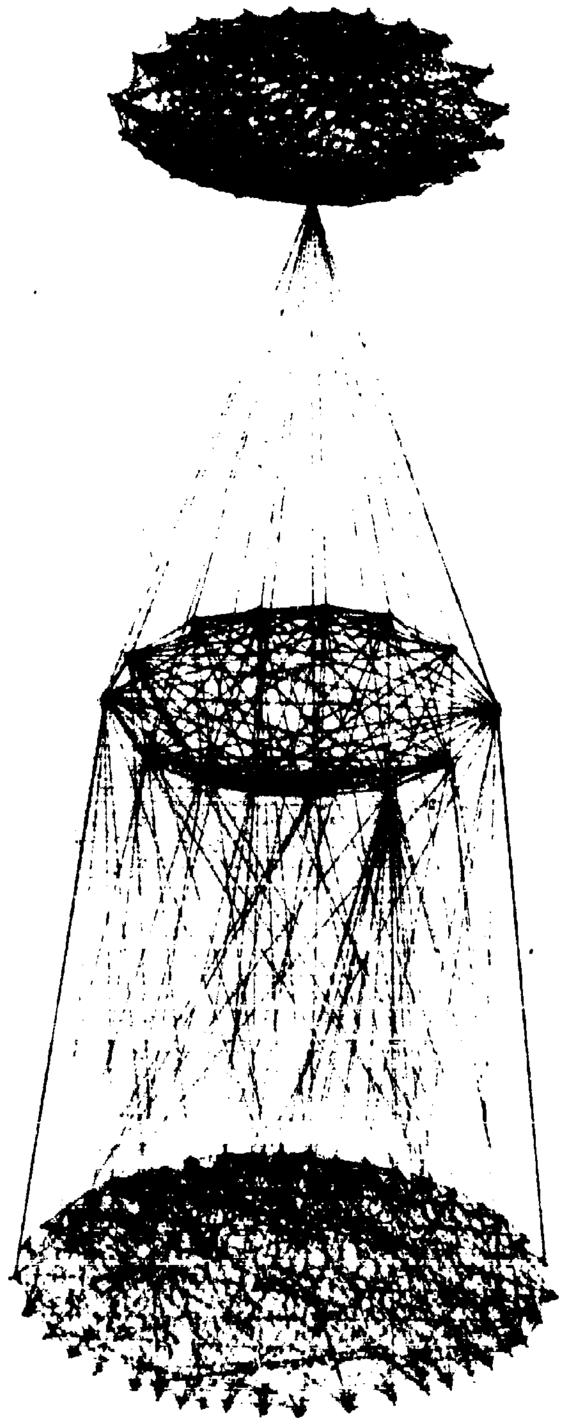
2) 1kW additional waste heat means an increase in A/C mass of about 50 kg

Figure 10



Connexions bet : the individual points in the "LEVEL OF SUPERIOR OPERATIONAL REQUIREMENTS"

Figure 10-1



Connexions between individual points in the same levels and different levels

Figure 10-2

COMPLEXITY

Graphical display of the relation between number of "Items" n and max. connections between them:

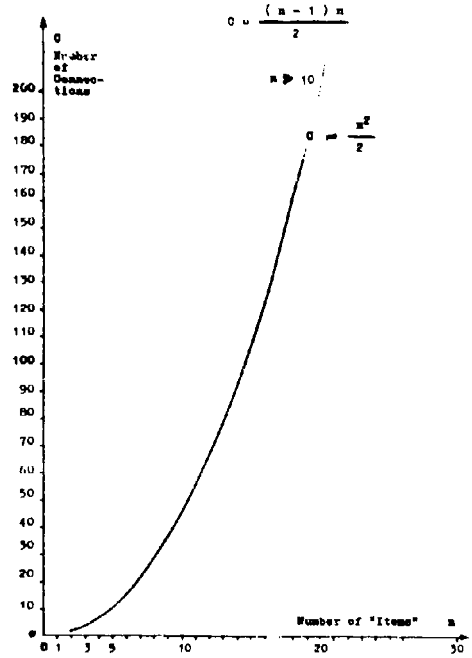


Figure 11

DISCUSSION

E.W.Eckert, Ge
(General comment after end of meeting).

At present there are a lot of discusions in NATO and European organizations and governments on future military helicopters for Army, Navy and Airforce operations, (see footnote) e.g. capability to relieve engine(s) of the burden of Secondary Power System in very critical flight status (e.g. Navy helicopter hovering at low altitude at sea etc.) where every KW is needed for the rotor. Based on first known requirements at least some of them might be candidates for APUs or JFSs.

I wonder why helicopter use has not been mentioned here. The PEP could certainly be helpful in providing support for decisions on auxiliary power in this area.

Footnote In Germany: LTH (Leichter Transport Hubschrauber)
MHS 90 (Marine Hub Schrauber 90)
PAH 2 (I nzer Abwehr Hubschrauber 2)

LTH = Light Transportation Helicopter
MHS 90 = Marine Helicopter 90
PAH 2 = Anti Tank Helicopter 2

GROUPES AUXILIAIRES A TURBINE A GAZ

par
A. ROMIER et P. CALMELS
Société MICROTURBO - Boîte postale 2089 - 31019 TOULOUSE CEDEX (France)

RESUME

Les Groupes Auxiliaires à turbine à gaz embarqués ont été développés pour répondre à la demande d'autonomie des avions lors de leur mise en oeuvre : check list, conditionnement au sol, démarrage des moteurs principaux. On examinera :

1) Les solutions développées dans les domaines militaires et civils tant en système de démarrage pur à fonctionnement court qu'en groupe auxiliaire à fonctionnement continu.

2) Les tendances actuelles sur les avions modernes où le système de démarrage est capable du fonctionnement continu en groupe auxiliaire.

3) Les besoins futurs que l'on peut prévoir en fonction de la complexité croissante des systèmes d'armes embarqués, nécessitant un conditionnement au sol, des développements des réacteurs principaux imposant des puissances de démarrage de plus en plus importantes et un fonctionnement en secours dans l'ensemble du domaine de vol avec des temps de mise en oeuvre extrêmement réduits.

1. SOLUTIONS DEVELOPPEES

1.1. Généralités

Les Groupes Auxiliaires embarqués ont pour objectif principal de conférer à l'avion porteur une autonomie complète par rapport aux moyens sol.

Ce besoin d'autonomie se justifie suivant le type d'utilisation de l'avion ; par la disponibilité logistique pour les avions militaires pouvant être appelés à utiliser des terrains sommaires et temporaires sans accompagnement d'une infrastructure sol nécessitant un transport par avion cargo, vulnérable en opération militaire, par le souci d'économie et d'indépendance pour les avions civils dont la rotation est importante et qui ne peuvent être tributaires d'une infrastructure au sol coûteuse, pour couvrir en nombre suffisant les aéroports et pour un service peu fréquent sur certains terrains.

Ce besoin d'autonomie de l'aéronef peut s'exprimer de plusieurs façons ; essentiellement démarrage du ou des moteurs principaux pour les avions militaires, confort en cabine (climatisation, électricité) avant le vol, moteurs arrêtés, et démarrage des moteurs principaux pour les avions civils.

Chaque cas a donné lieu à des développements spécifiques que nous pouvons brièvement examiner.

1.2. Application militaire

Comme nous l'avons souligné, l'objectif principal est le démarrage du ou des moteurs principaux de l'avion. Le Groupe Auxiliaire embarqué est dans ce cas un simple système de démarrage de fonctionnement très court.

Les critères principaux à satisfaire pour ce type de fonction sont :

- fiabilité de mise en oeuvre au sol dans un large domaine de températures ambiantes à partir d'une source de puissance électrique (batterie) ou hydraulique (accumulateur) relativement faible,

- rapidité de mise en oeuvre : la pleine puissance doit être disponible dans un temps minimum (10 à 15 secondes environ),

- encombrement et masse réduits ; le système de démarrage est un poids mort pour l'avion en vol.

On peut distinguer deux cas principaux de réalisation de système de démarrage suivant le type de transfert de la puissance :

- accouplement mécanique,

- accouplement pneumatique.

Dans l'accouplement mécanique le générateur de ...

Cette solution est bien adaptée à des avions bi ou multi-moteurs et peut satisfaire des impératifs d'installation, le générateur d'air pouvant être installé loin des moteurs, la liaison avec les démarreurs à air, d'encombrement faible, se faisant par de simples conduites (planche 2).

Actuellement, les niveaux de puissance requis pour les moteurs militaires modernes sont de l'ordre de 200 chevaux.

Cette puissance peut être développée en une dizaine de secondes par une turbine à gaz auxiliaire d'une masse de 35 kg (soit 7 Ch/kg).

1.3. Application civile

Dans l'utilisation civile des Groupes Auxiliaires embarqués, les fonctions assurées et le domaine de fonctionnement sont beaucoup plus étendus.

Le groupe assure au sol :

- la génération électrique de bord,
- l'alimentation en air comprimé du circuit de climatisation et de pressurisation de la cabine et des baies électroniques,
- le démarrage des moteurs principaux (planche 3).

Le fonctionnement du Groupe est assuré pendant les phases de décollage et d'atterrissage et peut être utilisé en secours dans un large domaine d'altitude (0 - 9 000 mètres).

Dans ce cas, les critères principaux à satisfaire sont :

- capacité de mise en oeuvre dans un large domaine d'utilisation (0 - 9 000 m, I.S.A. + 30°C à I.S.A. - 40°C),
- encombrement et masse réduits,
- performances élevées (consommation carburant),
- régulation de régime parfaite pour satisfaire l'entraînement d'alternateur à fréquence constante,
- maintenabilité, aptitude à la maintenance élevée par le développement de circuits d'autotests et détecteurs de pannes.

1.4. Conclusion

On constate que les Groupes développés à ce jour satisfont des critères différents et des fonctions différentes. Cependant, les développements récents en aéronautique militaire (Tornado, F18) et les études en cours montrent une tendance de rapprochement entre les deux concepts.

2. TENDANCES ACTUELLES

2.1. Systèmes de démarrage - Groupe Auxiliaire

Les avions d'armes modernes, de par leur complexité, nécessitent de plus en plus de puissance et de tests avant vol (système de navigation, commande de vol électrique). De ce fait, pour conserver le caractère d'autonomie le système de démarrage devient un véritable Groupe Auxiliaire capable d'entraîner les générations électriques et hydrauliques de l'avion monté sur la boîte "structure".

Dans ce cas, le système de démarrage est couplé à la boîte relais d'accessoires structure, elle-même connectée à un réacteur et peut, par le jeu d'embrayages hydrauliques ou électromagnétiques, soit entraîner la boîte relais et, de ce fait, l'ensemble des accessoires, soit démarrer le ou les moteurs. Ce principe est essentiellement retenu dans le cas d'avion bi-moteur, la redondance en vol en cas de panne de l'un des moteurs étant assurée, le système de démarrage ne sera alors utilisé qu'au sol en tant que source de puissance secondaire.

Dans le cas d'un monomoteur, il est généralement demandé au système de démarrage d'être capable d'assurer la relance du moteur principal en vol, l'allumage de ce dernier en auto rotation pouvant être dans certaines configurations de vol délicates et dangereuses (monomoteur école ou d'appui tactique au sol).

2.2. Groupes Auxiliaires - application civile

On ne note pas actuellement dans ce domaine une évolution des fonctions requises.

L'effort le plus important porte sur la fiabilisation et la réduction de la maintenance préventive des Groupes Auxiliaires. La disponibilité étant la qualité principale de ces derniers, il est nécessaire d'assurer le développement des circuits annexes de tests et de contrôle.

Le développement des techniques digitales, électronique numérique, permet de progresser rapidement dans ce domaine.

3. BESOINS ET DEVELOPPEMENTS FUTURS

Quels seront les besoins futurs ?

En plus des fonctions de base citées au sol :

- démarrage des moteurs principaux,
- génération de puissance électrique et/ou hydraulique,
- alimentation des circuits de conditionnement avion,

la conception des Groupes Auxiliaires devra permettre de satisfaire la sécurité en vol, c'est-à-dire, le secours total de l'avion.

Il est nécessaire pour cela d'assurer une redondance complète des sources de puissance. Il n'est donc plus suffisant que le système de démarrage soit capable de prendre en charge, c'est-à-dire d'entraîner, la génération principale de l'avion mais il faut qu'il soit équipé de sa propre boîte relais d'accessoires entraînant ses propres générateurs de puissance capables d'alimenter les circuits vitaux de l'avion dans tout son domaine de vol.

On peut distinguer deux fonctions vol différentes.

Dans le domaine basse altitude 0 à 10 000 mètres, le groupe de puissance peut être en fonctionnement continu à un régime d'attente et capable en 2 ou 3 secondes d'alimenter les circuits avion en cas de défaillance des générateurs entraînés par le réacteur et d'assurer une aide suffisante pour la relance du moteur tout en assurant la génération électrique.

Dans le domaine haute altitude 10 000 à 30 000 mètres, une source annexe doit alimenter en gaz de combustion la turbine de puissance du Groupe Auxiliaire pour assurer la fourniture minimale de puissance nécessaire aux commandes de vol et ce pendant les quelques minutes avant de retrouver le domaine basse altitude d'utilisation normale.

Les principaux obstacles à ce développement sont le poids, l'encombrement et le coût d'un tel système. Ces derniers tombent d'eux-mêmes si, dès la conception de l'avion on accepte de valoriser la fonction du Groupe Auxiliaire et de le concevoir comme une source de puissance embarquée.

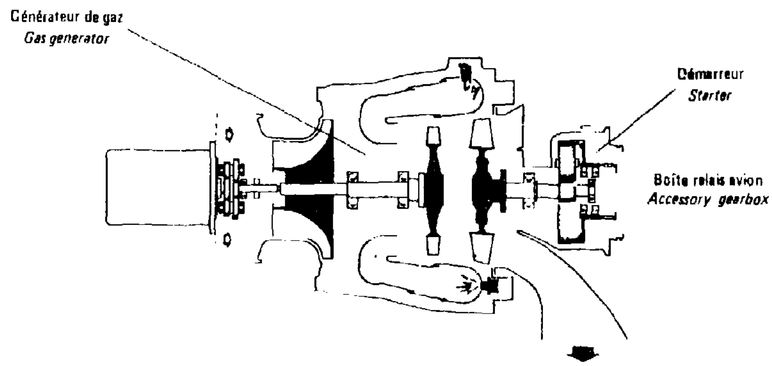
Par exemple, les développements récents, en électrotechnique notamment, peuvent permettre d'étudier des générateurs de puissance adaptées à l'entraînement par turbine à gaz auxiliaire, moins contraignants que par les moteurs principaux puisque les vitesses de rotation sont moins variables (rapport de 1 à 1,2 pour le groupe, contre 1 à 2 pour le moteur). De plus, la vitesse de rotation du groupe est nettement plus élevée que celle du moteur principal et le poids d'un générateur électrique est toujours inversement proportionnel à son régime de rotation. On est capable aujourd'hui de réaliser des alternateurs à grande vitesse (50 000 ou 60 000 tours/minute) de 40 kW ne pesant pas plus de 8 kg et associés à un convertisseur statique de 15 kg environ pour fournir cette puissance sous forme de 400 Hz triphasé et courant continu haute tension ou basse tension.

Les glissements de vitesse, chute de régime du groupe, pendant le démarrage des moteurs principaux, ne sont pas ressentis au niveau de la fréquence du réseau 400 Hz, cette dernière étant obtenue indépendamment du régime alternateur au niveau du convertisseur statique. Ce point est très important pour l'alimentation des systèmes de navigation qui ne peuvent plus être coupés après leur alignement.

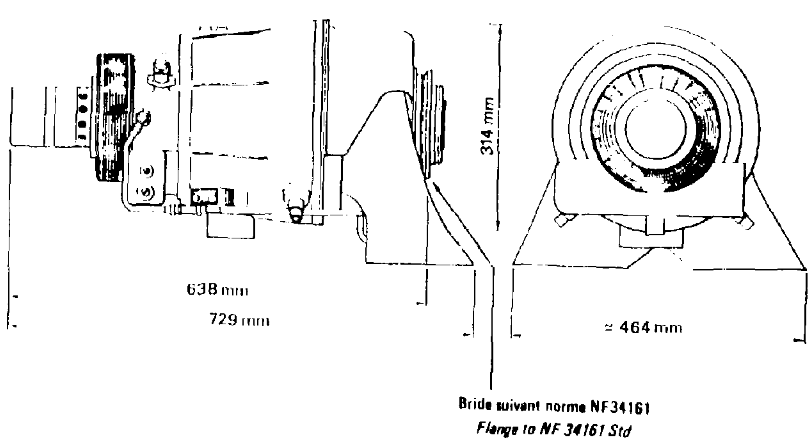
La transmission de puissance de démarrage sera, suivant le cas :

- mécanique, par couplage entre la boîte du groupe et la boîte structurale du réacteur, dans le cas d'un monomoteur et si l'installation le permet,
- pneumatique, par alimentation des démarreurs à air montés sur la boîte relais d'accessoires de chaque moteur dans le cas du biréacteur.

Ces solutions sont résumées sur la planche 4.

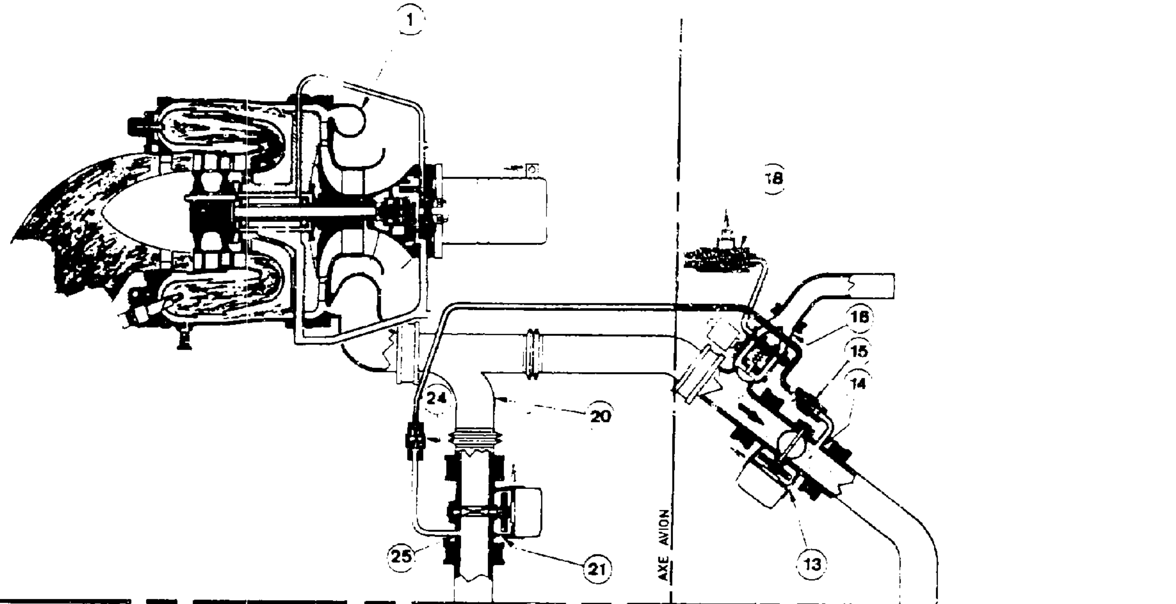


SCHEMA DE FONCTIONNEMENT
FUNCTIONING DIAGRAM



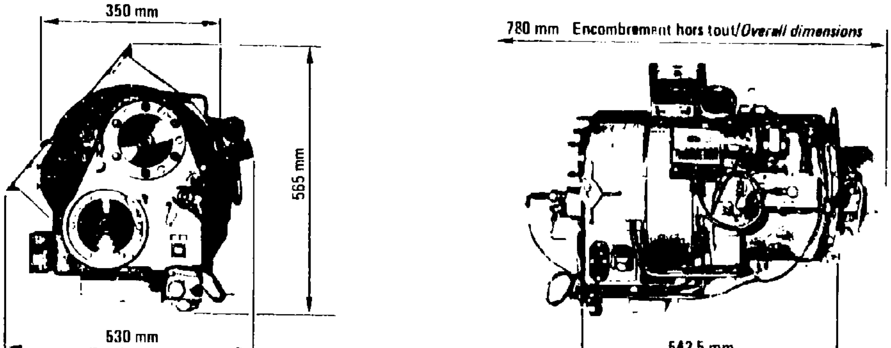
SCHEMA D'ENCOMBREMENT
OVERALL DIMENSIONS

PLANCHE 1

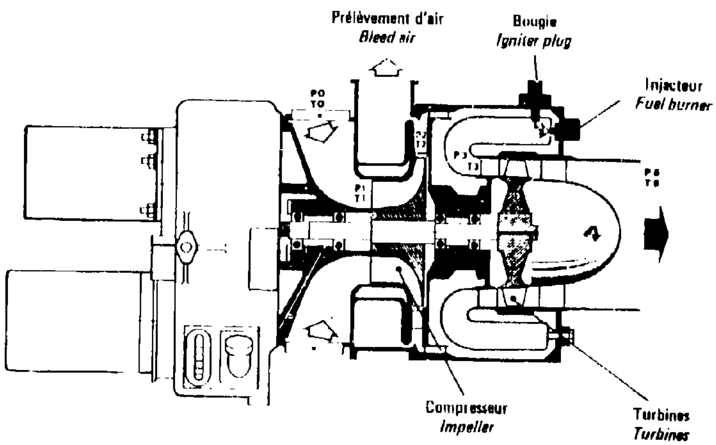


Rep	Désignation	Rep	Désignation
1	Générateur de gaz	13	Vanne d'alimentation
	Boitier électrique	14	Prise de pression air
	Clapet anti-retour carburant	15	Clapet anti-retour (air chaud)
	Bloc de distributeur — Servo-valve	16	Clapet de décharge
	Bloc de distributeur — Filtre régulateur		Filtre retour carburant
	Bloc de distributeur — Filtre à huile	18	Electrovanne de mise à l'air libre
	Bloc de distributeur — Mano-contact		[illegible]pet anti-retour
	Robinet manuel	20	Collecteur de sécurité
	Filtre armée	21	Vanne d'alimentation
	Electrovanne coupe-feu	22	Démarreur
	Electropompe double	23	Arbre cannelé
	Clapet réservoir	24	Clapet anti-retour (air chaud)
	Réservoir huile	25	Prise de pression air
11	Démarreur		Allumeur
12	Arbre cannelé		Câble de bougie
			Rampe de prélèvement

PLANCHE 2



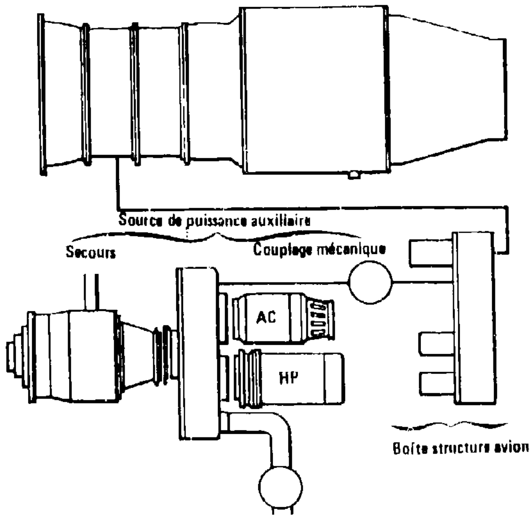
ENCOMBREMENT
OVERALL DIMENSIONS



SCHEMA DE FONCTIONNEMENT
FUNCTIONING DIAGRAM

PLANCHE 3

COUPLAGE MECANIQUE



COUPLAGE PNEUMATIQUE

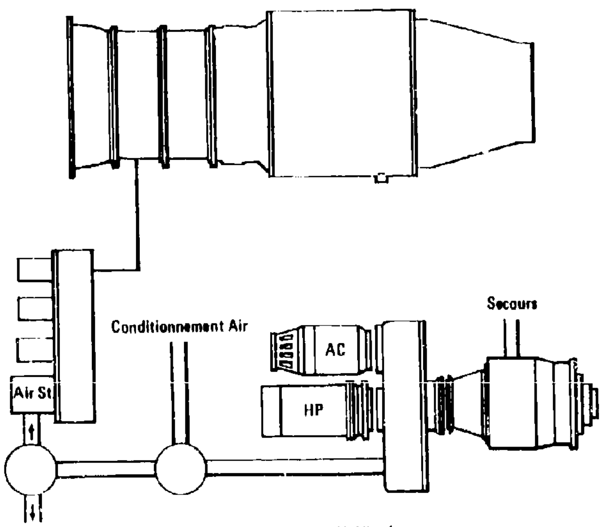


PLANCHE 4

DISCUSSION

R.Smith, US

The statement of start time of approximately 5 seconds. Is this for starting the starter or the main engine?

Author's Reply

Actuellement en utilisant des circuits de démarrage conventionnels (batterie-moteur électrique de lancement) la pleine puissance est disponible pour le lancement du moteur principal en moins de 9 secondes.

Le temps total de la séquence de démarrage du moteur principal, y compris le temps de mise en oeuvre du démarreur est de l'ordre de 25 secondes.

Ceci pour des systèmes développant une puissance de l'ordre de 200 ch.

J.F.Chevalier, Fr

Vous avez parlé d'une variante dans laquelle la turbine de puissance peut être alimentée, à haute altitude, par une chambre séparée à propergols. Avez vous essayé soit par cette chambre spéciale, soit par le générateur de gaz du groupe?

Author's Reply

Non, aucun essai n'a été effectué à ce jour mais cette conception est en cours d'étude.

SECONDARY POWER SYSTEMS FOR FIGHTER AIRCRAFT EXPERIENCES TODAY AND REQUIREMENTS FOR A NEXT GENERATION

by

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SUMMARY

The necessity of a continuous increase of fighting efficiency of weapon systems sets forth a number of new requirements for the next generation of military aircraft in particular regarding the distribution and use of the on board auxiliary power.

Energy conversion methods will have to be applied which are readily adaptable to operation requirements and also favour the thermal balance of the aircraft.

As an example of the pneumatic energy conversion the efficiency of a new auxiliary system is presented together with a listing of those factors which play a role in its optimisation.

secondary power and auxiliary power unit

Design concepts and options available for future ~~SP~~ and ~~APU~~ systems will ~~be~~ are presented. Today's experience, derived from a modern fighter aircraft system, that has successfully entered production, will serve as a basis for discussion of advanced requirements and design features.

SYMBOLS

ATM	Air Turbine Motor
APU	Auxiliary Power Unit
Co	Compressor
C/D	Cross Drive
ECS	Environmental Control System
EM	Electric Motor
FBP	Fuel Backing Pump
FP	Fuel Pump
GB	Gearbox
Ge	Generator
HP	Hydraulic Pump
IDG	Integrated Drive Generator
ME	Main Engine
PTO	Power Take Off
SPS	Secondary Power System
VSCF	Variable Speed Constant Frequency

1. INTRODUCTION

Modern military aircraft have developed into complex flying systems, making increasing use of auxiliary power to enhance both their handling qualities and fighting effectiveness. It is therefore that efficiency, reliability as well as cost of Secondary Power Systems have become of significant design importance.

As next generations of fighter aircraft are in their conceptual stages of design throughout the western world, it is felt appropriate to reflect upon the evolution of Secondary Power Systems and to monitor incipient progress together with future developments against todays experience and tomorrows requirements.

2. PRINCIPALS OF SECONDARY POWER SYSTEMS

Secondary Power Systems (SPS) for military aircraft have to provide and distribute auxiliary energy for airframe and main propulsion engines during flight and on the ground.

This auxiliary energy is needed as shaft output power, pressurized oil, electricity and compressed air.

Two different installation schemes for auxiliary energy systems are known and have been realized today (Fig. 1):

- Auxiliary system integrated into the main engine frame
- Auxiliary systems installed in the aircraft coupled to propulsion engines by means of a shaft.

For aircraft with only a low amount of secondary energy required and accordingly small accessories, these were integrated into the main engine(s). The concept however changed with the introduction of large propulsion engines and multi-engine applications.

Wereas previously the integration of accessories into the engine was possible due to their size, the accessory power now required and hence the dimensional increase of the auxiliaries demands remote arrangement of the SPS in an appropriate location of the aircraft. This also reduces the required cross sectional area of the aircraft. Additionally in many cases SPS's of today are equipped with auxiliary power units (APU), which provide for independence of ground support, availability of auxiliary energy for ground system checks, alert capability at minimum fuel consumption conditions and for main engine starting. However, these advantages have to be weighed against increased mass in case the APU is operationed on the ground only, and cost of installation with respect to air inlet and exhaust.

Fig. 2 summarizes the systems currently in use or in active design.
As already mentioned, accessories were previously installed directly on the main engine. This arrangement is still found in some of today's fighter aircraft.

The next step is the remote arrangement of the SPS. This system is mostly used in the current generation of fighter aircraft. For twin engine installations the gearboxes are mechanically coupled to each other and to the main engines. Also the APU is directly coupled to the gearbox (GB). For the immediate future this coupling will be replaced by pneumatic energy transmission with the exception of gearboxes remaining mechanically connected to the main engines.

3. EXPERIENCE GAINED WITH THE TORNADO-SPS

Aircraft with mechanical SPS have been out in operation for some time now. Their performance standard can therefore be evaluated both by development- and service experience.
Requirements for uninterrupted energy supply during all flight conditions and emergency situations result in very complex systems as they are found for example in F15 and Tornado.

Ten years ago, when these systems were designed, the goal to achieve was high transmission efficiency which automatically required a complicated mechanical system.

Furthermore, in the Tornado case, the start sequence and switching functions during emergency conditions had to be automated to the fullest extent possible which again contributed to the complexity of the system.
Fig. 3 shows the main features of the Tornado-SPS. The arrangement provides the redundancy required in flight, i.e. it has two independent gearboxes (starboard and port) and two generators (IDG-Type), two fuel backing pumps and two hydraulic pumps.

Independence from ground power is achieved by the APU which is mounted on to the starboard GB.
The SPS control unit monitors the function of APU and GB's including control of the clutches.

The system is activated by starting the APU with an electric starter motor. Upon achievement of full APU speed the clutch between APU and GB is automatically actuated. Engagement of the cross drive clutch also permits to drive the port GB so that in the APU running mode all accessories can be checked for proper function. Also the full supply of electrical and hydraulic energy is available.

The actuation of an electrically controlled valve allows to fill either the starboard or the port torque converter which initiates the start sequence of the respective main engine. After engine idle has been achieved the APU is automatically shut down. The second engine is started accordingly. In this case however, the already running engine is now driving the other engine to be started. Finally the cross drive clutch will be opened and both gearboxes are driven individually by their main engine.

The System-Advantages are:

The SPS is independent from ground power supply and can be driven by the APU alone for prolonged periods of time.

The start sequences are automated to a large degree, providing short times for scramble start. The availability of the cross drive clutch allows power input into the second accessory GB in the one engine failure mode as well as the inflight restart of a flamed out engine. This cross drive system yields extremly high mechanical efficiency.

The versatility of the system is payed for by high complexity. This is supported by the choice of integrated drive generators and the integration of their oilsystems into the GB's. The IDG's are susceptible to fast speed changes which necessitated a complex acceleration speed control for the GB mounted clutches. The effort required can also be seen in the oilsystem, where the oilsupply to the IDG's is of prime importance and where variable distribution of the oil under all existing possible flight attitudes has to be safeguarded.
Power losses generated in the accessory GB's and the IDG's are dissipated in fuel/oil heat exchangers.
Substantial effort was required to develop this ambitious oil system.

The wide distribution of the oil users and depots such as cooling system and generators and the interconnecting high volume passages require a large quantity of oil where on the other hand, due to size restrictions only small additional oil volumes for torque converter function and viscosity compensation were possible. These demanded extensive optimisation with respect to econimical use of the oil.

In conjunction with the requirement for multiple functions of the oil system the oilpumps had to be increased in size and number of stages which again adversely influenced optimisation as regards power losses.

Fig. 4 shows the results achieved, they can be considered as a very successful development.

The extensive experience gained with the Tornado SPS will bear fruit for future systems. This especially applies for the following main components:

- Multiple plate / dry disc clutches were selected for their higher temperature capability. These components are wear prone by function, especially in the area of friction surfaces and splines. Required component life could only be achieved by proper choice of material.
  The basic disadvantage of such clutches is their sensitivity to thermal overload; be it due to malfunction of their control system or mishandling. This can be of detrimental influence onto the system reliability.

- For the hydraulic torque converter safe function over a wide temperature range has been demonstrated. This also applies to the low ambient temperatures where oil viscosities approach 10.000 cSt.
  In service use so far has not shown any failure, hence this component can also be recommended for future SPS.

- Overrunning clutches in the Tornado-SPS have shown that tailoring to the individual requirement and test experience is mandatory.
  Parts that have operated successfully under similar conditions will not automatically function with the same reliability in a new design, even though high reliability and life has been finally achieved.

Experience gained during development also suggests that a future system should be equipped with generators that impose less stringent requirments onto the oil system. Furthermore the deletion of the mechanical cross drive between the GB's would significantly reduce complexity.

Fig. 5 shows the SPS of the F15 and F16 fighter aircraft with mechanical power transmission.

4. DEVELOPMENT TRENDS

Future trends in the development of auxiliary systems for fighter aircraft are directed towards a complete decoupling of the SPS's from the main engine. The following reasons dictate this development:

- Further increase of the auxiliary power demand results in accessory size increase which does not allow their installation in close proximity to the engine

- Increase in main engine performance and mission flexibility by the deletion of power extraction from the HP-spool of the propulsion engine

- Quick-change requirement for main engines with the accessories and the SPS remaining in the aircraft

- Electronic components are temperatur sensitive; since their application increases, they have to be removed from the h n temperature engine bay

- Favourable conditions for the integration of SPS and airconditioning system resulting in more economic use of the energy available.

For future generations of fighter aircraft various energy transmission systems are being considered to replace todays mechanical coupling of APU and GB's. For the time being however, the mechanical coupling between SPS and main engine will be maintained.

Technically four energy transmission systems and combination thereof, can be pursued, these being either mechanical, hydraulic, pneumatic or electrical in nature. In parallel with advances in respective technologies, the transition will be made from mechanical to pneumatic and finally to fully electrical system. In spite of significant advances in the past, the latter will however, have to await the full development of the required "high density" electrical generators and motors in order to allow for flying by wire only. This type system has not yet achieved the maturity to make its way into the next generation of fighter aircraft, presently being designed. Incipient progress will be made here by the pneumatic systems, which is therefore described in more detail.

5. THE PNEUMATIC SYSTEM

When changing to a new transmission system not only increased system flexibility but also possible influence onto the aircraft heat balance must be considered.
The dissipation of waste heat which is produced in all energy transmission systems results in considerable problems. The theoretical comparison of airframe waste heat based on the useable energy shows Fig. 6, that out of the systems available today and invisaged for tomorrow (pneumatic, hydraulic, electrical) the pneumatic system is not only most simple but also the system with the lowest waste heat.

Waste heat is generated by the oilsystems of the APU, accessories and intermediate gearboxes. Additional waste heat, resulting from losses when generating compressed air and the subsequent expansion in for example airturbine motors (ATM), is blown overboard.

The advantage of a considerable relief of the aircraft cooling system suffers from the worst total efficiency of all systems discussed in this paper.

An additional advantage of the pneumatic system undoubtetly is that this form of energy is directly available as bleed air from the main engine or the APU and that no further means of energy transformation such as hydraulic pumps etc. are needed.

It is for this reasons, that new fighter aircraft as they are presently under development, are equipped with pneumatic SPS.
The functional features of these systems are

- Flexible and simple energy distribution from various potential sources to many different users.
- Free choice of APU location as regards optimum arrangement for supply of consumers, good maintainability and ease of installation.
- Increased reliability, since rigid coupling is avoided and secondary failures resulting from individual system defects are reduced.

Fig. 7 shows the schematic of a pneumatic system. Not only can the APU, which now mainly generates compressed air, energize the aircraft's pneumatic system but also bleed air from the main engine and compressed air from ground supply can be fed into and distributed by the system.
Pneumatic power will drive the GB and its accessories by means of an ATM.
Air is also available for airconditioning purposes when passing through a cooler and an expansion turbine.

If such an energy distribution system connects all users the previously mentioned flexibility is immediately detectable. It is then for example possible for a main engine already running and supplying bleed air to start the next engine without again utilizing the APU. As a future option the APU of one aircraft takes over the standby-supply of other aircraft by means of umbilical lines.

Less effort is involved in treating emergency situations, such as a main engine failure where the GB could remain in operation by bleed air or monofuel onto the ATM. Accessory gearboxes in turn become much more simple in design and manufacture after the deletion of interconnecting shafts and the mechanical decoupling of the APU.
A factor greatly contributing to lower system life cycle cost.

Fig. 9 compares gearbox losses of mechanical and pneumatic systems.
Even though the losses of the pneumatic system are lower a further improvement in waste heat dissipation methods is mandatory.

Today, heat that cannot be dissipated by engine fuel to ambient has to be stored in the aircraft fuel tanks.
Continuous heat flow into the tanks together with a continous reduction of the amount of fuel will result in high fuel temperatures towards the end of a mission.

Therefore, in order to avoid overhea.ing a minimum residual amount of fuel has to be available at the end of the flight.
This fuel has to be considered as unnecessary ballast which detrimentally influences aircraft design and weight.

An increase in cooler size for the reduction of heat remaining on board does not show any improvement since such size increase will result in higher drag with the consequence of higher main engine power demand and fuel co..umption and last not least increased takeoff weight.

When changing to a pneumatic system it becomes mandatory to emphasize the subject of loss reduction.
It is important that not only the GB but also its oilsystem are simplified. Accessories that utilize the GB oilsystem should be improved to the extent where they do no longer require additional treatment of oil with respect to temperature and air content.

Considerable effort is presently spent to improve the efficiency of the accessor . This is most obvious in the field of electrical generators. Initially the manufacture. changed the Integrated Drive Generator (IDG) used today to the new variable Speed Con stant Frequency (VSCF), which is rigidly coupled to the gearbox, i.e. it can be driven with variable and higher speeds.

The required constant frequency is generated by a power conditioning unit. This electronic device however, poses a new challenge with respect to cooling.
Since there have also been improvements to the IDG, it is not the purpose of this paper to state, which type in the end should be favoured for a given application.

6. NEW AVENUES IN HEAT DISSIPATION

In the following, new avenues in heat dissipation will be described.
The reduction in waste heat together with the application of new cooling techniques allows operation of future pneumatic SPS's making reduced or no use of airframe mounted coolers. The proposals are further in order of their evolution and show increasing improvements as well as a reorientation of cooling techniques.

It can be stated that the max. allowable temperature in the fuel tanks is achieved at the earliest, during the last minutes of descent - most likely however during an increase in mission length caused by an emergency situation.
The amount of heat generated in this segment of the mission is shown in Fig. 9. Once the maximum fuel temperature has been reached, automatic starting of the APU will assist in oilcooling by means of an air/oil cooler located upstream of its inlet. The APU is now operating in a flight segment where idle operation is of no detriment, since fuel consumption is minimal (Fig. 10).

A further reduction of heat flow into the fuel tank can be achieved by heat dissipation directly overboard.
Fig. 11 shows an example for an aircooled gearbox. The finned wall of the gearbox forms part of the outer skin of the fuselage and is cooled by air. Heat transfer is very efficient since the oil flow is forced along the bottom portion of the gearcase.

Fig. 12 shows the typical mission profile of a modern combat aircraft with lines of constant heat disspation over a finned gearbox wall. Heat dissipation is dependent on ram temperature and hence on mach number and altitude. The amount of heat shown corresponds to a given area of finned surface. As would be expected residual heat exists in extreme points of the flight envelope which cannot be fully dissipated. If increased surface for heat transfer cannot be provided, a temporary increase in oil temperature could be acceptable.

If this is not possible or if the mission profile is extended to higher mach number, where ram air cooling looses its efficiency a supplementary expendable coolant system will offer relief (Fig. 13).
A container is filled with water and equipped with a tubular oilcooler. Evaporation of the water and overboard discharge of the generated steam through a valve reduces oil temperature to approx. 100°C. Upon reaching the maximum allowable oil temperature the evaporative cooling is turned on automatically.
A sample calculation resulted in 1,5 kg of water being required for dissipation of 4 kW of residual heat during 15 minutes of suparsonic flight.
In addition it could be contemplated to use an expendable coolant system as the only means for oilcooling. According to Fig. 9, the total heat generated during a mission amounts to 5,14 kWh. 7,5 litres of water would suffice to dissipate this amount of heat. This volume could be stored in a spherical container with a diameter of 250 mm.

7. APU's FOR PNEUMATIC POWER

In instances where aircraft system operation requires independence from ground equipment, this can only be gained by installation of an APU, to provide pneumatic starting power.

The individual application will define APU design criteria, such as

- Split between pneumatic and mechanical power
- Space available
- Air intake and exhaust ducting
- Gearcase and accessories.

Fig. 14 depicts the APU used in the Tornado-SPS. This powerplant supplies mainly shaft power and a small amount of bleed air. It is of single shaft design with axial air intake and a laterally orientated exhaust duct. Fig. 15 shows design schematics for bleed air supply, with a limited amount of mechanical power provided.

Variant "A" is a single shaft turbine driving a compressor. This system offers increased flexibility with respect to customer requirements, it however also results in a long slim unit. If space restriction require more short and compact features, single shaft APU's to configurations B and D are available. In both cases the engine compressor supplies the bleed air. This simplification with respect to design results in a lower efficiency bleed air generation.

Case "B" shows a split-compressor where the pneumatic power is taken from an appropriate location within the compressor flow path where the required pressure has been reached.

Case "D" however, shows an engine where bleed air is taken out at the end of compression i.e. the air is first brought up to pressure required by the engine cycle. Subsequent reduction of pressure to levels compatible with the pneumatic system results in significant penalties for system efficiency.
Where good part load operation and minimum specific fuel consumption are of prime importance todays APU's are designed as two shaft engines with separate load compressor configuration "C".

Fig. 16 finally compare torque and power of single shaft and two shaft designs.
The single shaft engine supplies no positive torque below 50% speed, therefore it cannot be started under load. The speed regime at which such engine can work is limited between approx. 80 and 100%. The shaded area shows the performance map of a single shaft design. It is narrow and drops of rapidly with decreasing speed.
The two shaft engine connects gasgenerator and power turbine pneumatically whereas the load is mechanically coupled to the power turbine. Torque is available from 0 to 100% power turbine speed.

8. CONCLUSION

Together with advances in aircraft systems also SPS's will see improvements in current and future applications. They have become an additional design aspect in its own right and hence will influence system life cycle cost. This challenge has been accepted by manufacturers of SPS's and APU's. They are ready and prepared to make their contribution to improve future generations of fighter aircraft systems.

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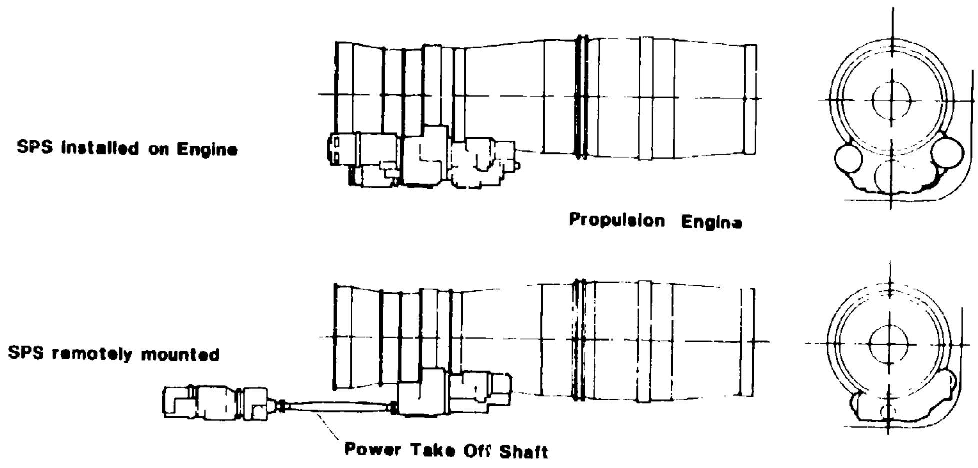

Fig.1 Secondary power system for fighter aircraft

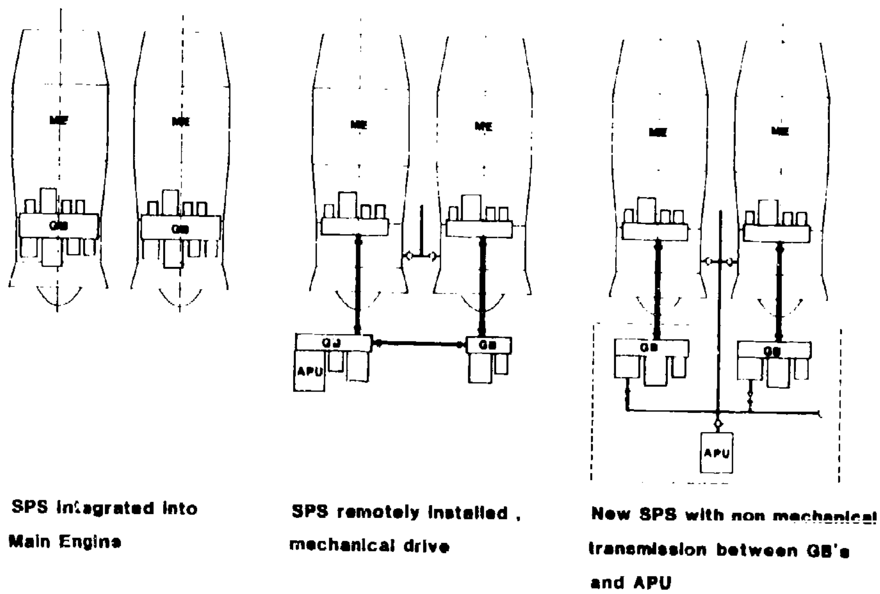

Fig.2 Development trends of secondary power systems

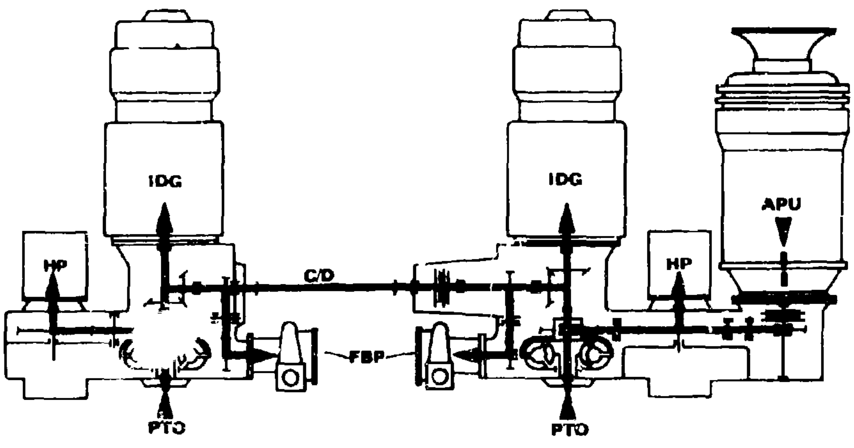

Fig.3 Secondary power system of Tornado aircraft

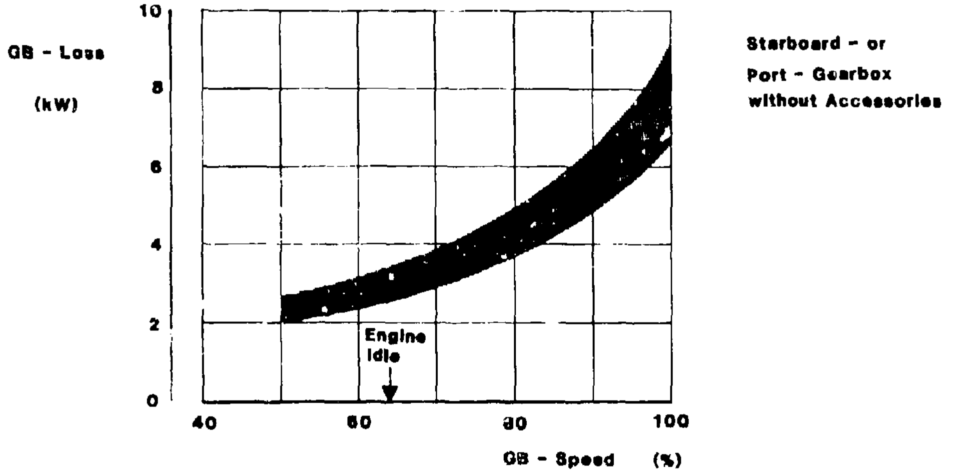

Fig.4 Tornado – secondary power system gearbox – losses

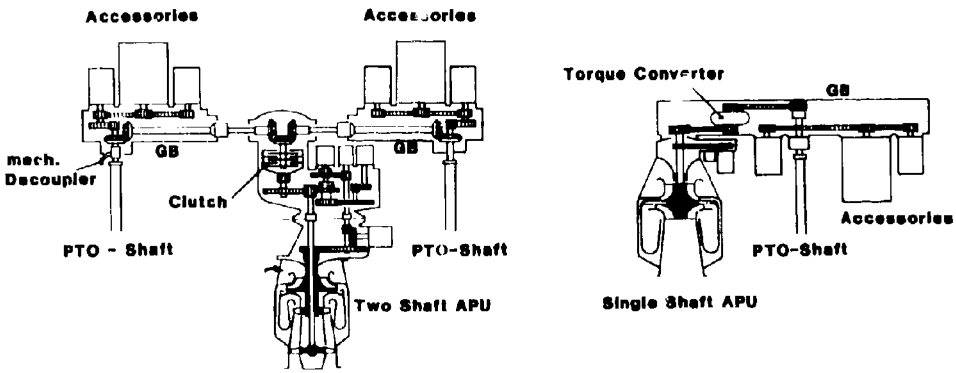

Fig.5 SPS's with mechanical energy transmission

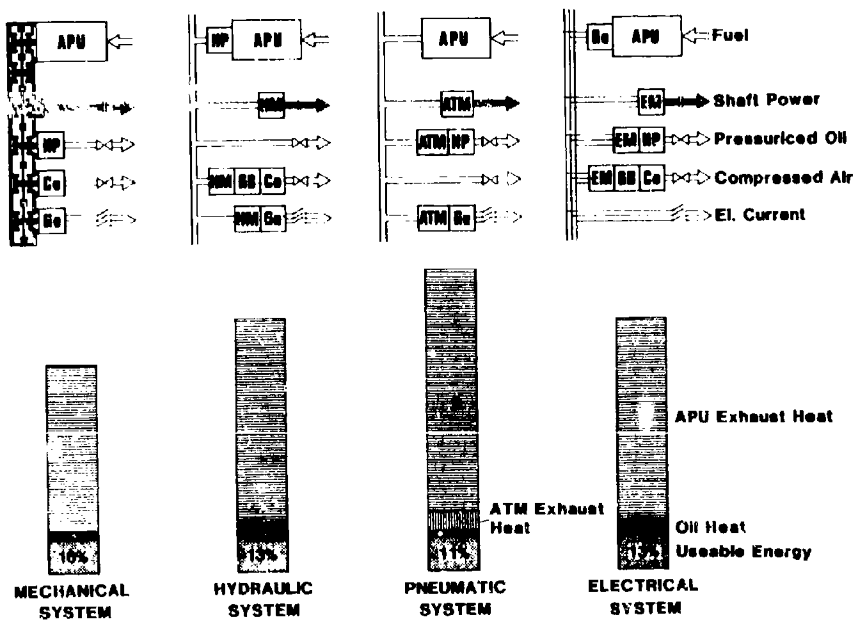

Fig.6 Comparison of different energy transmission systems
estimated values of heat loss

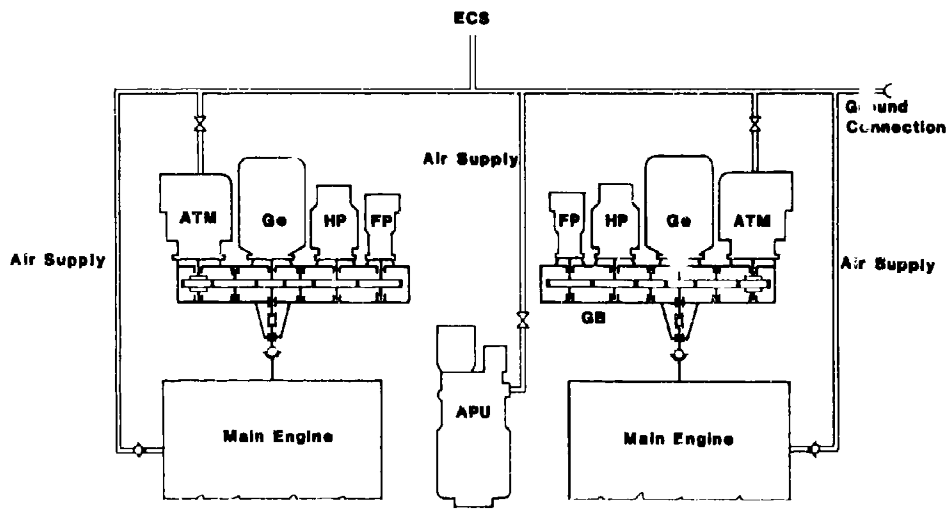

Fig.7 Secondary power system with pneumatic energy transmission

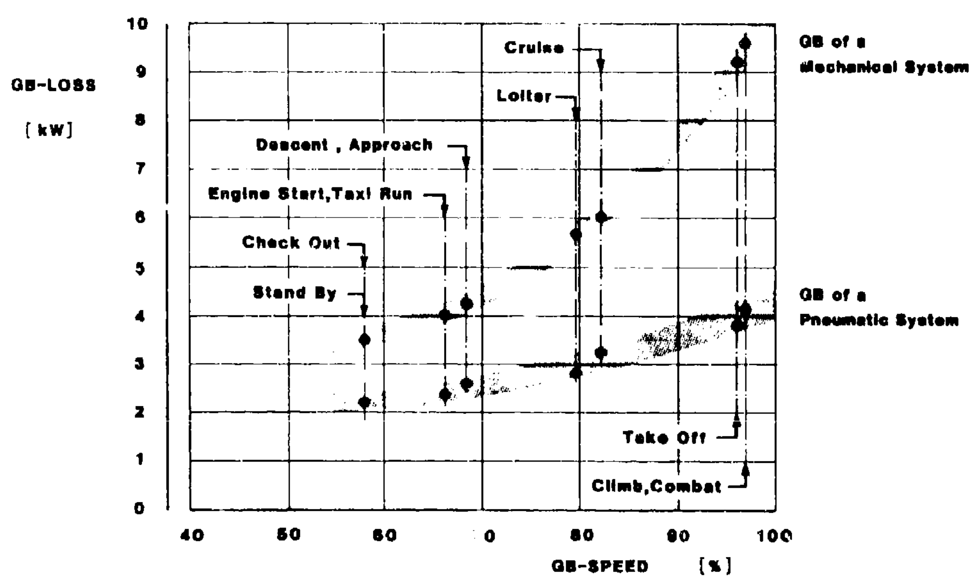

Fig.8 Power loss in different gearboxes

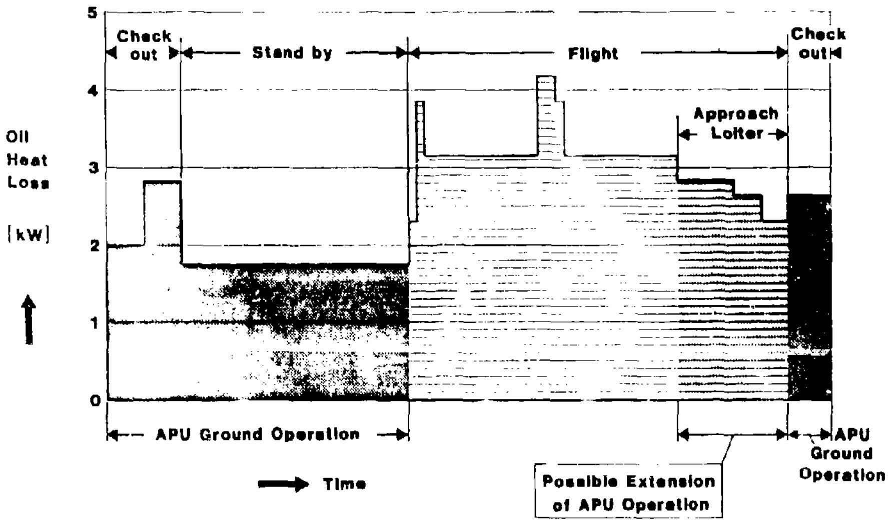

Fig.9 Oil heat losses of a pneumatic system-GB generated during a mission

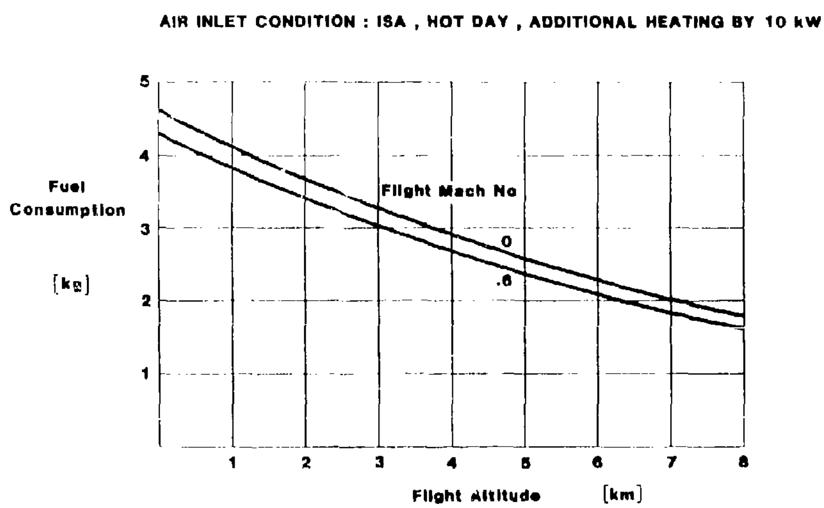

Fig.10 Fuel consumption during 10 minutes of APU operation at full speed, without power extraction

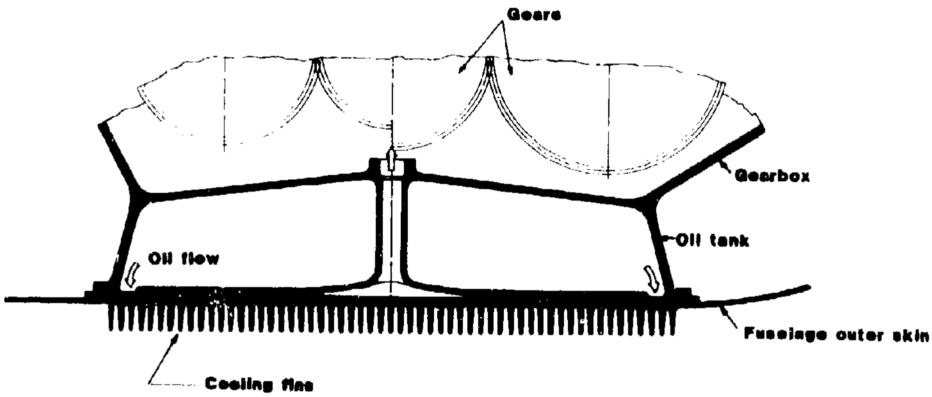

Fig.11 Gearbox with outer skin cooling

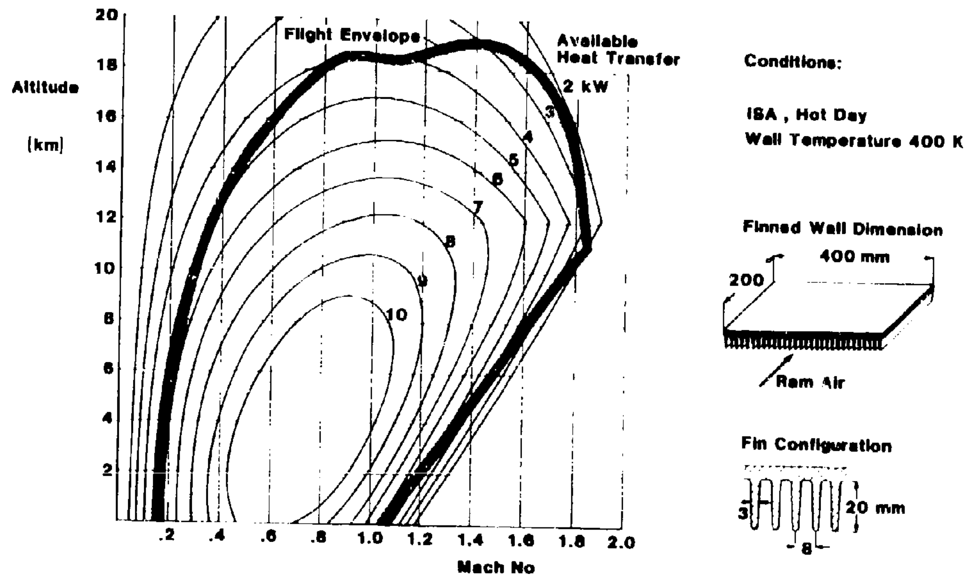

Fig.12 Cooling through a finned surface when subjected to ram air

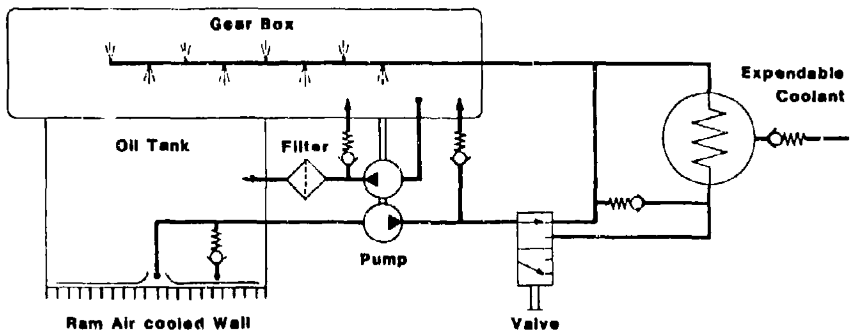

Fig.13 Oil schematic with a new cooling system

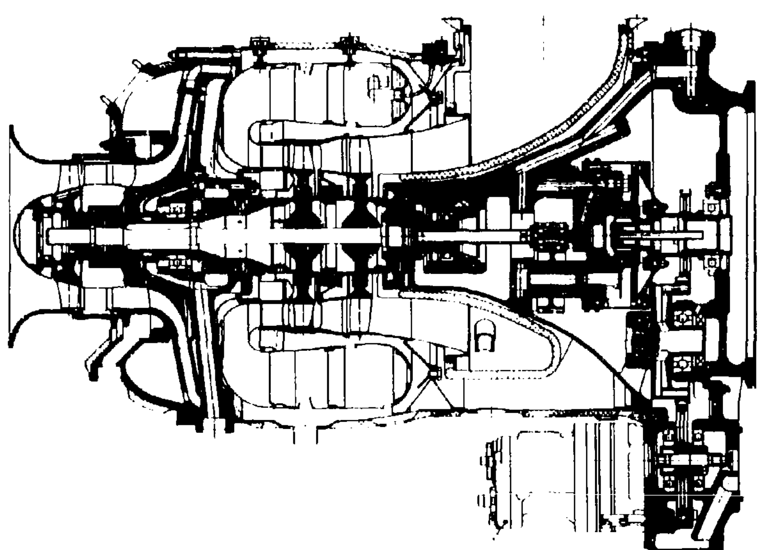

Fig.14 APU KHD T 312 for Tornado SPS

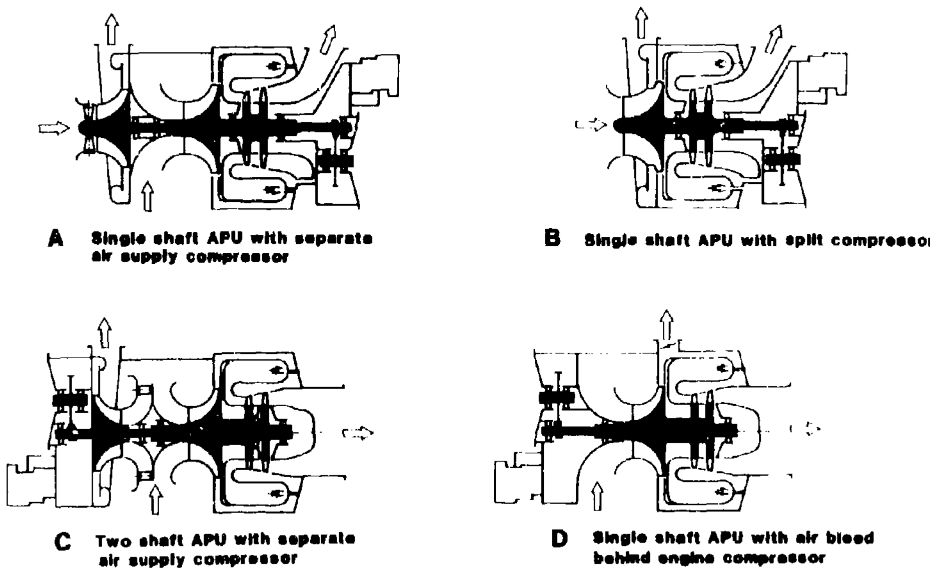

Fig.15 Different configurations of air delivering APU's

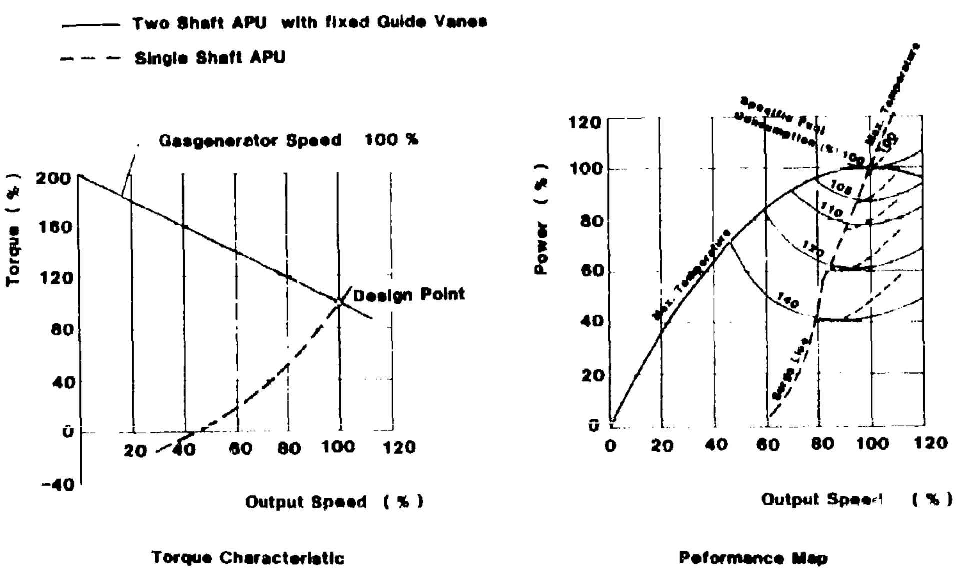

Fig.16 Comparison of torque characteristic and performance between a single shaft and a two shaft APU

DISCUSSION

G.B.Toyne, UK

Fig.6 of the paper shows how the heat is dissipated for each of the following systems; mechanical, hydraulic, pneumatic, electrical. Have the Authors done a comparison of the total weight for each system?

Author's Reply

Not yet, the weight analysis is under evaluation.

A.L.Romanin, US

With retard to the ram air/finned oil cooling design: has this system been compared to other cooling methods regarding weight, resistance, volume and cost?

Author's Reply

As was said before weight analysis is under evaluation, volume and cost will be dealt with later on. The resistance, i.e. flight drag will be approximately 1% less compared with conventional ram air heat exchangers.

E.H.Warne, UK

How do you propose to overcome difficulties due to freezing of the expendable coolant, (Fig.13 of your paper)?

Author's Reply

Freezing of the coolant will be avoided by means of additives.

K.Mose, Ge

Amongst APU configurations presented in Fig.15, is there the APU model being offered for the Airbus A 320?

Author's Reply

For Airbus A 320 most probably configuration C will be used.

R.Smith, US

Fig.11 and 12. Are the cooling fins rammed with a recovery duct or are they simply on the external skin?

Author's Reply

The cooling fins are part of the outer skin of the fuselage.

SMALL AUXILIARY POWER UNIT DESIGN CONSTRAINTS

by
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ABSTRACT

Self-sufficiency for military aircraft operating from remote advanced bases can be attained with small on-board air breathing gas turbine auxiliary power units (APUs) supplying main engine start and aircraft secondary power.

The small, fixed shaft, gas turbine configuration comprising the single-stage radial compressor and radial inflow turbine, mounted back-to-back, and overhung from a "cold" bearing capsule has found favor in providing this duty due to inherent attributes of low cost, simplicity and high power-to-weight ratio.

This configuration of APUs first entered service in the early 1950s, and derivatives have been designed, developed and produced to meet aircraft industry demands. Extensive experience with these APUs has led to the formulation of several major design constraints, within the objective of minimum life cycle costs, that enhance development of both modified and derivative versions. This paper highlights some of these design constraints and identifies advantageous areas of research and development for future APUs.

NOMENCLATURE

b	Blade Height
D	Rotor Diameter
g	Gravitational Constant
H	Head
hp	Horsepower
Hz	Frequency
k	Constant
kW	Power
L	Blade Length
n	Exponent
N	Rotational Speed
NS	Specific Speed (Dimensionless)
q	Compressor Work Factor
Q	Volume Flow
Rc	Compressor Pressure Ratio
Re	Reynolds Number
SFC	Specific Fuel Consumption
t	Blade Root Thickness
T	Total Temperature
T.I.T.	Turbine Inlet Temperature
U	Rotor Tip Speed
Vo	Theoretical Spouting Velocity
β	Blade Angle
δ	Altitude Correction Factor
Δ	Difference
ν	Kinematic Viscosity
ω	Angular Velocity

SUBSCRIPTS

1	Compressor Inlet
2	Compressor Exit
3	Turbine Inlet
c	Compressor
t	Turbine
m	Metal
Pol	Polytropic
opt	Optimu

INTRODUCTION

In the United States, competitive bidding on major military, airborne and air breathing APU programs has historically shown that the primary program requirements were (in order of importance):

- Development and Selling Cost
- Reliability/Maintainability
- Volume and Weight
- Fuel Consumption

To minimize costs, the development of small gas turbine APUs has been constrained to the modification, adaptation, or extension of existing engines and components. Only in special circumstances have private sector or government research and development funds been available for the design and development of new APUs.

Self-sufficiency for both military and commercial aircraft operating from remote airfields can be attained with small on-board APUs supplying main engine start and ground checkout power. In spite of increased fuel costs, APU fuel consumption has not been a major aspect because most applications involve only intermittent duty. Fundamental design emphasis has therefore been focused upon a reduced number of components for reliability and maximum power-to-weight and power-to-volume ratios.

Life cycle cost (LCC) analyses may be conducted for comparative design evaluation when detailed information concerning a specific mission profile has been defined by the user. In most instances, development and selling cost constraints have dictated the selection of an engine utilizing a maximum of existing hardware and experience. This is corroborated by the fact that it is not uncommon for a particular APU product line to have 40 or more individual variants for diverse fixed and rotary wing aircraft applications.

Under the constraints of minimum development, selling costs, high reliability, and minimum weight, a design philosophy has evolved which is the major topic of this paper. These constraints can essentially be divided into two categories. For example, if an APU specification is issued which can be satisfied by an existing model or derivative thereof, the major constraint will be system integration for minimum LCC. Alternately, if the specification demands an entirely new APU, LCC must be optimized within the constraints of the engine and component design disciplines.

DESIGN DISCIPLINES

The major engine design disciplines of cost, life, performance and weight must be integrated to optimize the final APU configuration. Cost often mandates a simple turbomachinery configuration with a minimum number of components and simple external impingement cooling of the hot-end module.

The small gas turbine and turbocharger configurations having the largest production in the United States to date are comprised of a single-stage radial compressor mounted back-to-back to a single-stage radial inflow turbine. This configuration, shown in Figure 1, has found wide acceptance for small APUs in the 10 to 200 kW class.

Figure 1. Small Single Shaft Gas Turbine

Although "simplifying design through component reduction" can be construed as an overstatement of probable engine reliability, the establishment of a design goal in the reduction of components and parts is a big step forward in attaining the ultimate goal. Component reduction is also an obvious technique to minimize cost and tolerance stackup restraints. Consequently, the compressor and turbine rotors are often single-piece castings similar to that of the mass-produced turbocharger rotors.

The dominance of the single-stage radial or centrifugal compressor stems from its cost attributes. Incremental improvements in turbine materials and progressive aerodynamic development of centrifugal compressor technology have provided increased temperature ratio, pressure ratio and airflow swallowing capacity (specific speed) to the extent that APU power-to-weight ratios of 4.4 kW/kg, and power-to-volume ratios of 8000 kW/m³ are attainable.

Other small gas turbine component configurations are also manufactured, retaining the single-stage radial compressor, but in combination with a single-or two-stage axial flow turbine.

CYCLE OPTIMIZATION

Design optimization normally begins with a design requirement for an engine of given power output, specific fuel consumption, and possibly weight and size. Within these confines, it is customary to select an optimum combination from the following cycle variables:

- T.I.T.-to-Ambient Temperature Ratio
- Compressor and Turbine Efficiencies
- Pressure Ratio
- Combustor Efficiency and Pressure Drop
- Mechanical Losses

Comprehensive performance analysis for small gas turbines using single-stage radial compressors and turbines have been developed (Reference 1) and extended to include the constraint of turbine rotor stress rupture life.

Current methods of predicting the peak component efficiencies demand lengthy computation procedures and extensive input including complete turbomachinery geometry description and performance requirements. Such prediction methods are too inflexible for use in a cycle optimization procedure. To obtain a practical procedure, it is necessary to define component efficiencies in terms of a reduced number of parameters without significantly sacrificing accuracy. The major parameters which influence component efficiency are rotational speed, Mach number and pressure ratio, component size, operating clearances and state-of-the-art technology level.

The influence of rotational speed and compressibility can be assessed from specific speed charts such as those for typical single-stage radial flow compressors and turbines shown in Figures 2 and 3 respectively. Component state-of-the-art efficiency levels are depicted based upon the defined limitations, and can be digitized for inclusion into cycle analysis routines.

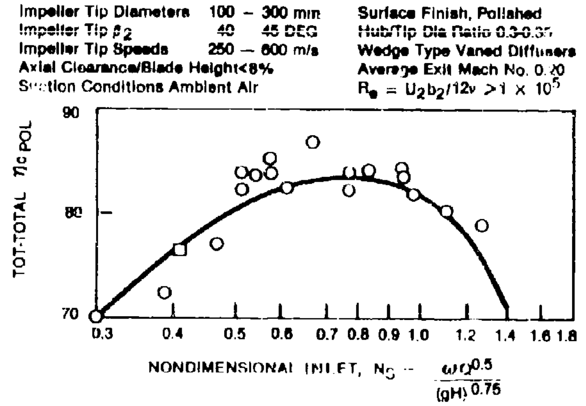


Figure 2. Attainable Peak Efficiencies Single Stage Centrifugal Compressors

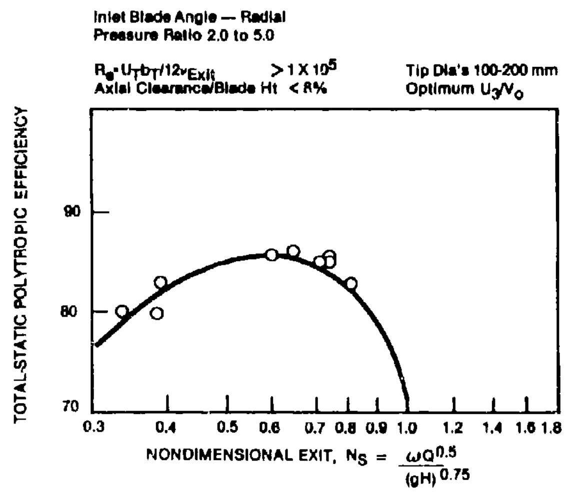


Figure 3. Attainable Peak Efficiencies Single Stage Radial Turbines

For intermittent duty, there are incentives to provide higher specific power with increases in cycle temperature and pressure ratios, commensurate with turbine life and reliability objectives. Higher single-stage compressor pressure ratios are realized by increasing rotor tangential velocities:

$$U_2^2 \propto \Delta H/q \tag{1}$$

Higher Mach numbers at the rotor entry and exit are incurred. The aerodynamic problems associated with diffusion at these conditions are being resolved to attain efficient high pressure ratio compressor operation. These problems involve careful selection of the blading solidity, thickness chord ratios, nose radius, hub and shroud contours, appropriate rotor and diffuser diffusion ratios, and strict control of the design dimensions.

At higher pressure ratios, compressor surge and engine matching do not always allow operation at peak compressor efficiency. Therefore, engine design point compressor efficiency may be one to two percentage points below peak efficiency depending upon compressor characteristics.

Pressure ratio increase by itself does significantly impact turbine performance and life. Efficient turbine expansion demands relatively high rotor tip speeds (650 m/s) at pressure ratios above 5.0. High tip speeds consequently increase rotor centrifugal stresses and compromise turbine life and burst margin.

Optimum efficiency for radially bladed turbines occurs adjacent to a velocity ratio of $U_3/V_0 = 0.7$. Other considerations often restrain the tip speed, U, to values less than the aerodynamic optimum, where turbine efficiency approximately decreases according to the relationship:

$$\eta_t = \eta_{t_{opt}} \left[1 - \left(\frac{U_3/V_0}{(U_3/V_0)_{opt}} - 1 \right)^2 \right] \tag{2}$$

Factors influencing the selection of allowable tip speed are:

- Turbine Life - Stress Rupture, Low Cycle Fatigue, and Burst Margin
- Turbine Efficiency $(U_3/V_0 \neq U_3/V_0 \text{ opt})$ (3)
- Engine Envelope Dimensions
- Engine Starting Characteristics

For a typical non-internally cooled superalloy rotor, the allowable tip speed for a given life can be expressed as:

$$U_3 \propto k_1 (T_m - T_3)^n \quad (4)$$

where: n = an exponent on the order of 0.5

T_m = Metal temperature near zero strength

The implication of this relationship is that the stress (stress rupture and low cycle fatigue) permissible tip speed decreases rapidly as the T.I.T. approaches 50°C of the "zero strength" temperature level, which for current conventional superalloys is around 1400 K.

Engine envelope limitations and starting characteristics result in the selection of a turbine tip diameter normally no larger than 10 percent of the compressor tip diameter: $D_3 = 1.1\ D_2$.

OPTIMIZATION EXAMPLE

The simplicity of the radial gas turbine with single-stage compressor and turbine components permits a computerized performance optimization incorporating all the aforementioned design disciplines plus determination of the approximate APU weight, volume and cost from the following correlations:

$$\text{Weight} \propto k_2(R_c - 1)D_2^{\ n} + k_3D_3^{\ n} + k_4 \quad (5)$$

$$\text{Volume} \propto k_5D^n + k_6 \quad (6)$$

$$\text{Cost} \propto k_7W^n + k_8 \quad (7)$$

Weight and volume include the powerhead and gearbox but exclude accessories, driven equipment, containment, ducting, or external oil cooling. APU cost studies indicate powerhead costs are a function of weight; whereas accessory, control, assembly and test costs are essentially fixed in this size range. These correlations should be interpreted as representative of the weight, volume and original equipment manufacturer's cost trends, rather than indicative of absolute values.

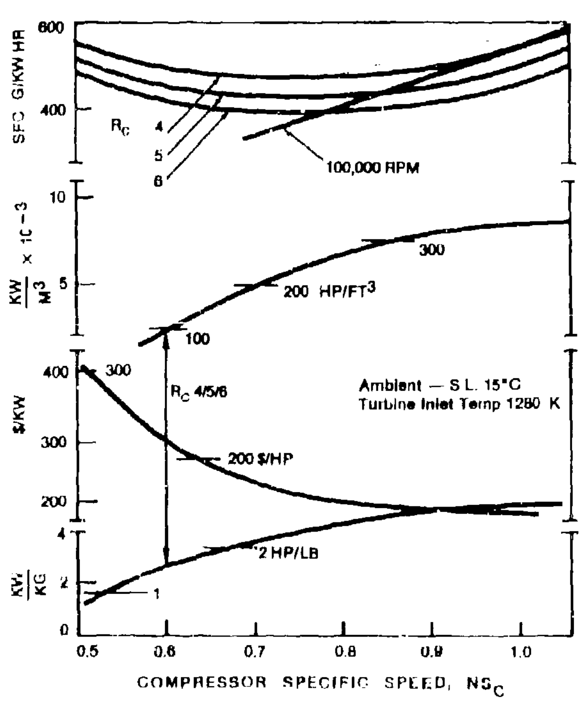


Figure 4. 100 kW APU Optimization

A hypothetical 100 kW APU radial configuration was analyzed using the performance optimization technique to assess the trends of weight, volume, cost, and fuel consumption at sea level, 15°C conditions. A maximum uncooled T.I.T. of 1280 K was selected with compressor pressure ratios of 4.0, 5.0, and 6.0. The results and analysis are shown in Figure 4 and indicate the following trends.

- Optimum cycle SFC and minimum cost/weight occur at different specific speeds.
- Cycle pressure ratio has little effect on specific weight, volume and cost. (The use of titanium compressors at higher pressure ratios can increase cost up to 10 percent.)
- With continuing emphasis upon minimum weight for intermittent duty APUs, component development efforts should be channeled towards specific high speed compressors and turbines, plus high heat release combustors.

The trends only relate to the optimization of an entirely new or hypothetical design. In most instances, higher specific power is obtained by continued uprating in terms of speed, airflow, pressure ratio and T.I.T. for an existing APU. Continued uprating within the same envelope towards maturity in engine development may result in reduced fuel economy in spite of increased power, as precipitated by component specific speed effects and increased duct pressure losses.

It is of further interest to examine the effect of engine power on specific weight, volume and cost at a specific compressor speed of 0.9, as shown in Figure 5. Engine weight departs considerably from the classic $kW^{1.5}$ formulation, since low cost demands utilization of existing accessory and control components.

Specific APU power and volume ratios rapidly approach zero at very low powers, as do most other sources of secondary power. At lower powers (less than 30 kW), rotational speeds in excess of 150,000 rpm are encountered near optimum specific power and volume.

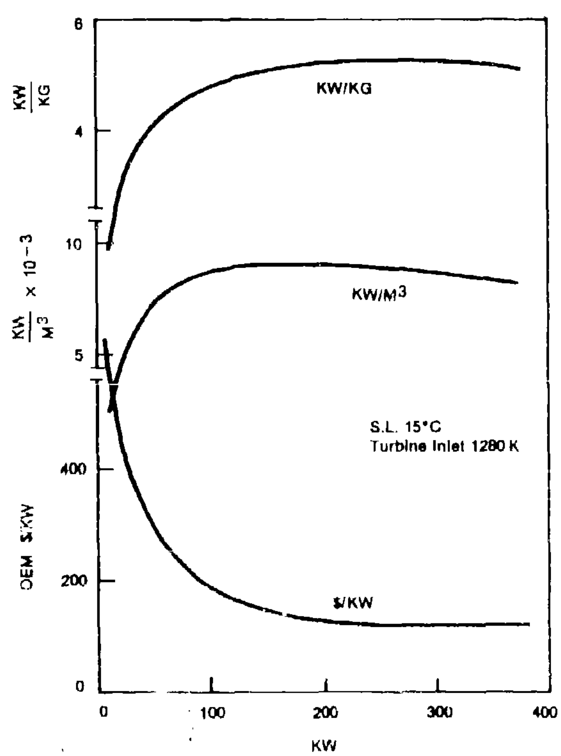


Figure 5. Influence of Power on Cost, Weight and Volume

Small gas turbines (Figure 6) still have higher powe. volume and power-weight ratios than intermittent combustion engines and stored electrical energy (batteries).

The cycle optimization example previously discussed was conducted at sea level, 15°C conditions for comparative evaluation purposes. Most intermittent duty APUs for main engine starting are designed to operate at ambient temperatures of 40° to 50°C where engine starting difficulty is increased.

ENVIRONMENTAL CONSIDERATIONS

Typical trends for the effect of ambient temperature, altitude and flight Mach number for constant speed, single-shaft radial gas turbines operating at constant T.I.T. are shown in Figures 7 and 8. It is observed that output power decreases approximately 30 percent at hot day conditions of 50°C. Thus, if the APU is sized for hot day aircraft starting requirements, the maximum T.I.T. for ground starts will only be experienced at elevated ambient temperatures. Since up to 95 percent of APU operating time is normally spent at ambient temperatures below 32°C, the turbine is rarely exposed to the maximum rating, which is similar to flat rated helicopter gas turbine engines. Emergency inflight starting is occasionally specified up to the altitude limits where consistent APU starts can still be achieved, depending upon starting technique and ignition characteristics. In emergency conditions, time from "power loss" to "power up" is vital, particularly for fly-by-wire aircraft. Air breathing APUs have difficulty in meeting such altitude emergency restart requirements, due to power lapse rate (Figure 7) which invariably occurs with no inlet ram. Consequently, supplementary non-air breathing emergency power units are also installed in many aircraft, with commensurate weight penalties.

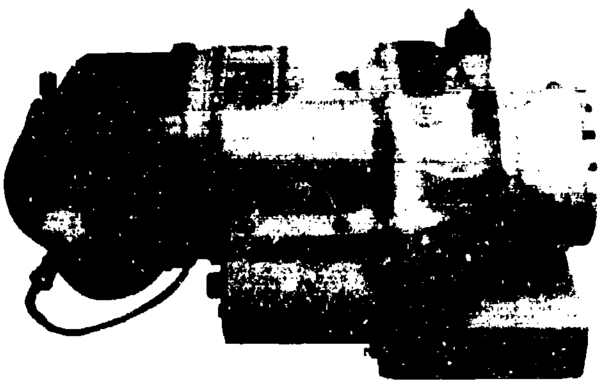


Figure 6. T20G Gas Turbine

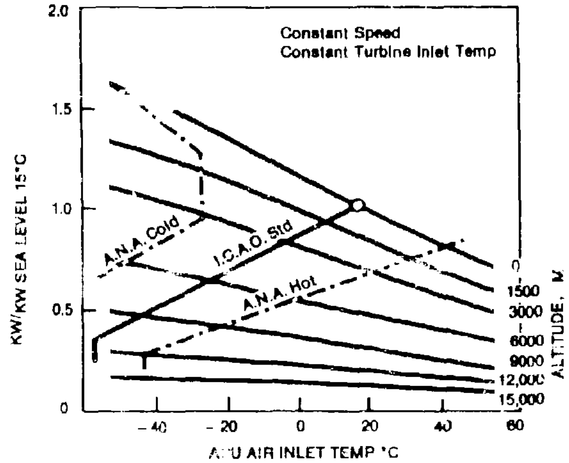


Figure 7. Typical Power Lapse Rate Single Shaft Radial Gas Turbine

Reference 2 indicates that high altitude start and power characteristics of air breathing APUs can be enhanced by improvements in ignition, fuel atomization, and combustion, plus installation to maximize the significant effect of inlet ram (Figure 8). It is now considered feasible with appropriate research and development to attain full APU power in an elapsed time of six to seven seconds at 15,000 meters altitude.

HIGH AND LOW CYCLE FATIGUE CONSTRAINTS

For cost reasons, it is desirable to produce both compressor and turbine rotors from single forgings or castings. A fundamental problem with monolithic rotors is their very high response to excitation (similar to that of a church bell). There are many sources of high frequency energy excitation within small gas turbines which can "ring" a bell-like structure; therefore, cast rotors are susceptible to high cycle fatigue failure. Fortunately, because the small radial gas turbine normally operates at constant speed, it may be possible to detune resonant frequencies out of the operating speed range. The two-shaft gas turbine, often used as a jet fuel starter (JFS), operates at variable speeds and requires additional vibration testing to assess high cycle fatigue durability.

It is a standard and advisable practice to avoid any of the first three or four orders of shaft speed resonance. This is difficult with specific high speed, high Mach number compressors, and it is necessary to accurately control blade frequency between the second and third excitation order levels, or compromise performance in favor of durability. Approximate blade first flap frequencies for radial rotors can be calculated from the data shown in Figure 9, which are useful in preliminary rotor design.

More accurate frequency determination is obtained from three-dimensional, finite element modeling in the detail design phase (Figure 10).

Eventually, holographic testing of actual hardware resolves the static resonant modes, which, in combination with engine dynamic strain gauge testing, defines the resonant frequency (Campbell) diagram.

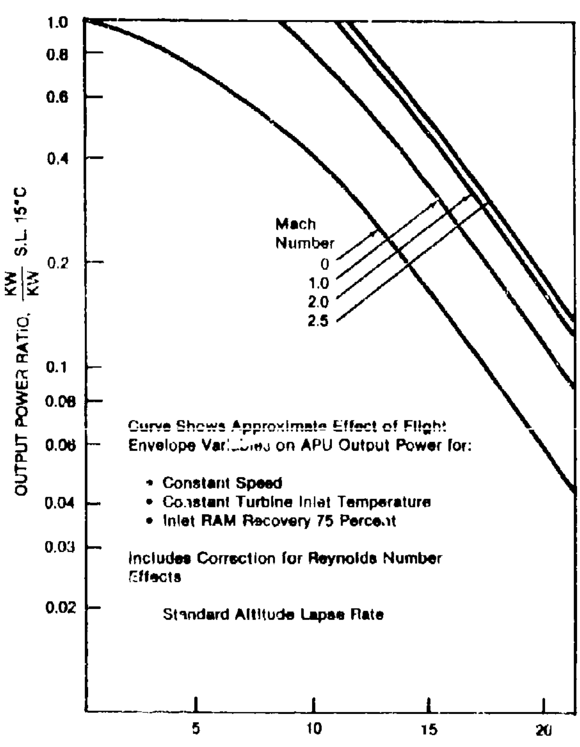
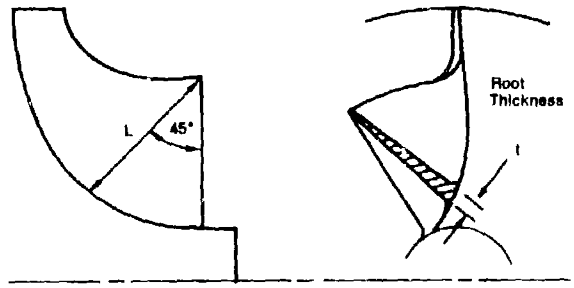


Figure 8. Effect of Altitude and Flight Mach Number on Gas Turbine APU Performance



Static Frequency $H_{ZS} \cong \dfrac{kt}{L^2}$

$k = 55{,}000 - 50{,}000$

$\dfrac{\text{Dynamic Frequency}}{\text{Static Frequency}} \cong \sqrt{1 + 2.0\left(\dfrac{H_{ZS}}{H_{ZR}}\right)^2}$ $\quad H_{ZD}$

H_{ZR} = Rotor Shaft Frequency

Figure 9. Estimated Blade First Flap Frequency

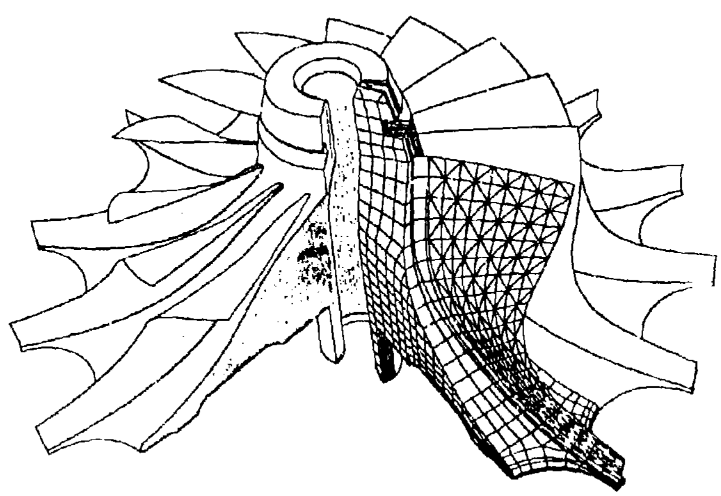


Figure 10. 3D Finite Element Rotor Model

Small gas turbine jet fuel starters are also susceptible to low cycle fatigue (LCF) life limitations. These units are required to make between two to six thousand start cycles, depending upon the installation. A typical start cycle endures up to 10 seconds for the JFS start, followed by 30 seconds of maximum T.I.T. to accomplish a main engine start. Immediate shutdown follows and restarts are often required after a minimum cool down period of two minutes. As most small APUs have minimal external or no turbine rotor cooling, the maximum thermal gradients may be 300°C or more. The turbine rotor accumulates low cycle thermal fatigue damage by repeated operation to full speed and temperature.

Turbine overtemperature and overspeed excursions impact LCF life. In some applications, only one turbine exhaust temperature probe is used for engine control. Calibration of the probe must be maintained, especially in hot climates, and should be accurately positioned at a point in the gas stream representative of the mass-averaged temperature. In general, LCF life is extended by operating at lower temperatures and lower rotor stresses, which are contrary to the attainment of maximum specific power and are another example of demanding design constraints.

In addition to high and low cycle fatigue and stress rupture constraints, turbine rotor burst overspeed margin and mode of burst influence turbine rotor design philosophy. For stress optimized rotor discs, the overspeed burst margin decreases with increasing pressure ratios and turbine inlet temperatures. The mode of burst is a significant factor in determining APU containment weight, if required by specification. The disc can be designed for a high burst margin with a tri-hub burst characteristic, or a somewhat lower burst margin with a more desirable, fragmented, lower energy disc burst.

SHAFT DYNAMICS BEARINGS AND SEALS

The shaft dynamic characteristics of radial gas turbines with overhung back-to-back compressor and turbine rotors (Figure 1) have been extensively studied.

Shaft critical speed is primarily a function of the following variables:

- Combined Rotor Mass
- Bearing Stiffness
- Shaft Diameter
- Bearing Span
- Rotor Overhang

Different vibration modes for a given rotor system with large changes in spring rate are exhibited in Figure 11. The linear portion of the curve, at low bearing spring rates, corresponds to the "rigid body" mode of vibration where little or no shaft bending occurs and all motion is in the bearings and their supports. At high bearing spring rates, the converse is true, and almost all motion is in shaft bending, with little motion in the bearings. In the transition region, the vibration mode shape involves both bearing motion and shaft bending. With bearing spring rates in the 3×10^4 to 4×10^4 kg/mm range, the rotor characteristic lies in the transition region, with a critical speed near 10 percent design speed. This rotor system would demonstrate the very desirable characteristic of having one critical speed in the low speed range and no additional critical speeds in the high speed or operating range.

For a given rotor mass and bearing stiffness, the important variable in the support system is the shaft diameter, preceded by the bearing span/to overhung ratio. Likewise, the shaft diameter is also constrained by the bearing "DN" value limitations. Bearing span to overhang ratios of near unity provide the highest system critical speed.

The significance of seeking the highest critical speed is to provide the stiffest overall system in terms of combined shaft and bearing deflections which are of importance to compressor and turbine shroud clearances.

Important factors controlling bearing life in overhung radial rotor systems are:

1. Selection of optimum bearing size and "DN" value
2. Rotor unbalance, prior to and after repeated dynamic and thermal excursions
3. Bearing environment and lubrication
4. Bearing preload (angular contact bearings)
5. Bearing internal clearance (roller bearings)
6. Rotor static load and end thrust

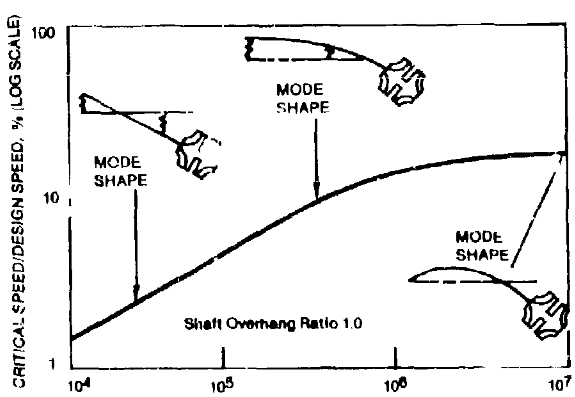


Figure 11. Effect of Bearing Stiffness on Critical Speed

The majority of field service related bearing problems stem from the above factors 2 and 3. Bearing "DN" limitations do not scale directly with engine size. One deviating constraint arises from the larger ball size to withstand centrifugal stresses at high rotational speeds.

Current bearing materials and life requirements limit bearing "DN" values for small radial engines to the order of 1.8 x 10^6 mm/rpm.

The "cold" location of bearings, shown in Figure 1, is an ideal environment where the compressor inlet air bathes the bearing capsule. Sealing is required between the bearings and the subatmospheric pressure at the compressor eye, and is accomplished with either a buffered labyrinth, viscoseal or more expensive carbon face seal. Heat input to the bearing capsule arises from bearing viscous drag, heat conduction from the turbine rotor down the shaft, and possibly the buffer air. Buffer air heat input becomes significant as advanced technology demands higher compressor pressure ratios and higher buffer air temperatures.

Higher compressor pressure ratios also result in elevated Mach numbers and impose a trade-off between compressor hub size - thus bearing external diameter - and minimum inlet relative Mach number.

APU STARTING AND COMBUSTION

The combustor and fuel system are not only vital to APU reliability but may also be the only determining factors of consistent aircraft starting. Fast starting is required over a wide range of ambient temperatures and altitudes for the APU and the main engine to which it supplies starting energy. Starting torque characteristics of small aircraft gas turbines and APUs are discussed at length in Reference 3. The dominant factors controlling start characteristics at sub-zero temperatures are both the lubricant and fuel viscous shear effects. In fact, the stored energy start system (with hydraulic accumulator) for rapid sub-zero starting may weigh as much as the APU itself because of high lubricant shear stresses. Starting duty APUs are further burdened with the incompatibility of torque output dependent upon altitude correction factor "δ". However, main engine initial cranking torque is dependent basically upon lubricant viscosity.

Combustion for the small radial APUs is accomplished in either annular reverse flow or single can scroll burners, which are common to many larger gas turbine engines. The principal combustor requirements are minimum envelope and an efficient short stable combination flame over a wide range of environmental operating conditions. This must be achieved with fuel viscosities of one or more centistokes. Efficient combustion over this viscosity variation permits the use of a compact, lightweight, low energy, ignition system and allows simple, open-loop acceleration fuel scheduling. With restricted volume, the time for fuel evaporation, mixing, and reaction is small. Short residence time is critical to ignition, combustion efficiency, flame stability and flame length. However, mixing and reaction criteria have less influence on fuel evaporation. The principal combustion problem is to provide fine fuel evaporation with viscous fuel, thereby accelerating the fuel evaporation process.

Handling small fuel flows requires extremely small flow passages when conventional pressure atomizing systems are employed. These small passages are susceptible to clogging with dirt or gum. This becomes even more restrictive in considering the importance of good atomizing requirements for lightoff and combustion (during lightoff and APU acceleration) with about one-third of the full speed fuel flow. The problem is further compounded when the total fuel is distributed among several injection points, as required by annular burner designs.

For these reasons, the combustion systems of smaller APUs employ a single can-type combustor, in which all the air and fuel are confined to one location for atomization and burning. A unique rotating cup atomizer, shown in Figure 12, permits fuel addition to the combustor at low pressure through large flow passages, and atomization is accomplished by thin film injection of fuel from the edge of a single, open, rotating cup with little possibility of fouling or plugging. An additional significant advantage of the rotating cup system is that the electric motor-driven cup can be at full speed and full atomization potential when the fuel is first introduced for lightoff during the turbine start-up. The excellence of this system ensures reliable initial lightoff and high combustion efficiency with all fuels, particularly at the extremely low temperatures required in typical military applications.

DEVELOPMENT CONSIDERATIONS

The development history of small radial gas turbines has shown conclusively that it is difficult to duplicate engine component performance in typical individual component test rigs. The major reasons for this difficulty are heat transfer, shaft dynamics, leakage, and flow distribution effects.

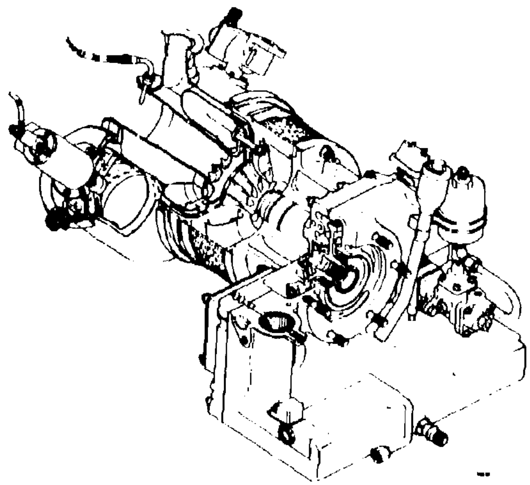


Figure 12. T20G Gas Turbine with Rotating Cup Fuel Atomizer

Clearances

Since clearance gaps must be provided between the rotor and its shrouds, leakages from the rotor blade pressure to suction surfaces occur and decrease the efficiency. The minimum tolerable operating clearance is determined by a transient operating condition, and the efficiency at the desired steady-state operating condition is compromised.

Test results indicate that the effect of clearances on radial compressors and turbine efficiency can be approximated by:

$$\text{Compressor efficiency loss \%} = 0.25 \times \text{axial gap/tip blade height \%} \quad (8)$$

$$\text{Turbine efficiency loss \%} = 0.15 \times \text{axial gap/tip blade height \%} \quad (9)$$

A loss in compressor efficiency due to increased axial clearance is accompanied by a reduction in matched compressor airflow at the rated engine speed and operating temperature. This follows as a consequence of reduced compressor discharge pressure against an essentially constant turbine flow function.

The assignment of clearance settings for the compressor and turbine requires a compromise among the requirements of the aerodynamicist, the mechanical designer, and manufacturing personnel. A thorough appraisal of these requirements is mandatory before clearance gaps can be chosen, especially since the above clearance loss relationships indicate that a gap-to-blade-height ratio of 10 percent could decrease compressor efficiency 2.5 percent and turbine efficiency 1.5 percent. Maintaining gap-to-height ratios may prove impractical to the mechanical designer and manufacturing personnel when blade heights less than 5.0 mm are necessary.

The number of components, individual component tolerances, turbomachinery arrangement, and bearing clearances can magnify the problem of maintaining close gaps. A shimming process is often used to satisfy clearance gap and component tolerance limitations with the widest possible manufacturing tolerances.

Heat Transfer

Heat transfer from the hot section of an engine to the compressor influences both the direct aerodynamic processes and the thermal equilibrium and positioning of the stationary shrouding of the compressor. Assembly clearances which may be acceptable on a rig may result in interferences on an actual engine. Gas turbine package systems often use inlet air for the gearbox and accessory cooling with resulting unmixedness and nonuniform compressor air. Compressor rigs, on the other hand, are noted for their exactly defined and uniform inlet conditions.

Thermal modeling of the gas turbine engine assembly during design can assist in assessing the compatibility of rotor and stationary shroud deflections during steady-state and transient conditions. Solutions of the complex problems raised by these analyses benefit greatly from simulated engine testing early in the compressor development.

The magnitude of heat transfer effects on both the rotating and surrounding stationary components increases with the trend toward higher pressure ratios, higher tip speeds, and higher turbine inlet temperatures. External heat losses also tend to increase with surface area, which is inversely proportional to diameter. The effects are further amplified on small size components, in which potentially high performance losses are associated with large ratios of clearance gap to rotor blade height.

Shaft Dynamics

The customary component development approach is to select a rig design that minimizes potential mechanical problems associated with rotor support and drive-shaft dynamics. At some later stage of development, it becomes necessary to superimpose the effects of the engine and its support and shaft dynamics system. It can be seen that these effects are better measured on an actual system when one examines some of the whirl patterns that characterize engine rotor systems. The minimum rotor to stationary shroud clearances are significant only when due consideration has been given to the rotor system at its critical speed condition on the complete engine rotor. For the back-to-back system with overhung compressor and turbine, the maximum rotor radial excursion occurs at the turbine exducer tip during acceleration through the first critical speed of the shaft system. Here, minimum clearance is primarily contingent upon rotor imbalance and the piloting system retaining the turbine nozzle. These factors are, in turn, dependent upon thermal conditions throughout the engine components. Therefore, it becomes evident that early engine testing is required to fully resolve these problems.

Leakage

The sealing system, required to prevent external and internal engine leakages, may produce performance defects which cannot be simulated on a simple compressor rig. Seal pressure and temperature differentials vary with engine load levels and account for variations in these leakage effects. In actual engines, these effects are rarely known directly, and therefore cannot be duplicated on a compressor rig. The sealing problem is further aggravated by the realities of manufacturing tolerances.

Flow Distribution

The flow distribution in and out of the engine compressor depends upon the final engine design and, in some instances, the final package configuration. The compressor outlet influences the combustor design and vice versa. Performance is critically dependent upon flow symmetry and swirl and cannot be properly considered without correct knowledge of these flow conditions.

In spite of the many pitfalls involved in individual component rig testing, it is advantageous to conduct the initial exploratory component performance evaluation on idealized rigs. These tests confirm the validity of the flow swallowing capacity and provide a more reproducible standard of performance evaluation. Such tests are particularly necessary in instances where departures from established design procedures have been attempted in an effort to obtain significant technological improvements. However, the improvements must be immediately examined in a realistic engine environment when overall engine performance is the ultimate goal.

LIFE CYCLE COSTS

Life cycle costs for airborne small gas turbine APUs are highly mission dependent. For basic start duty life cycle costs consists of:

- Acquisition Cost
- Cost per Overhaul
- Maintenance Cost/Start
- Number of Starts per APU Overhaul
- Start System life
- Development Costs

Costs are fairly well defined with a generic APU overhaul and development, and depend mainly upon labor rate, materials and past experience. Maintenance costs may not be dictated by actual APU running hours or number of starts, since environmental aspects play a dominant role. These aspects are, for example, acceleration (g) forces, vibration amplitude and frequencies, and windmilling.

For combination start and continuous duty APUs, engine operating time and fuel consumption will enter LCC appraisal, depending upon the used life fraction for each mode of operation. For continuous duty periods less than 1000 hours between overhauls, fuel costs may essentially be omitted from LCC calculations.

In a given installation, APU size may be determined by the main engine start time or by the environmental control system for cooling of electronics.

The demand for minimum weight (especially on intermittent duty APUs) necessitates that the APU operates close to its limiting T.I.T. at hot day conditions. The percentage of time spent at this limit controls overhaul life. Typical start cycle requirements between overhauls range from a minimum of 500, to a normal of 2000, and to an upper limit of 8000 starts. For installations where continuous duty consumes the major life fraction, the average time between overhauls is on the order of 2000 hours. For mixed missions, either 2000 hours or 2000 starts may be used as an overhaul criterion.

An option for reduced APU life cycle costs is the selection of a derated APU with better LCF and stress rupture life. This option involves a complete weapon system LCC analysis because this derated APU carries a take-off gross weight penalty.

The diversity of APU applications, modes of operation, geographical locations, T.I.T. and stresses at rated power requires each LCC analysis to be conducted on a case-by-case basis. In general, such analyses indicate that the initial acquisition cost can constitute about one quarter of the total LCC. However, some major APU programs have been and are primarily secured on the basis of acquisition cost.

ACQUISITION COSTS

Much has been written and speculated concerning the potential low cost, small gas turbine. The cost yardstick has been the piston engine of equal power. Factors which have so far prevented the realization of low cost, small gas turbines are:

- Low Production Volume
- Requirement for Specialized Manufacturing Machinery and Tooling
- High Fuel Consumption
- High Rotational Speeds Requiring High Reduction Ratio Gearboxes
- Use of Strategic Materials
- No Low Cost Source for Accessories

Current Department of Energy automobile gas turbine programs (Reference 4) attack the cost standard of superalloy strategic materials with the proposed large scale development of ceramic casting technologies. Nearly all of the large automobile manufacturers have tried the gas turbine venture, without yet entering volume production.

Relative engine component costs for small, simple-cycle radial engines currently in production are listed in Table 1. Improved manufacturing technology has been suggested (Reference 5) as a way to reduce engine manufacturing costs to 60 percent of current standards. However, there is little evidence to support a reduction in small gas turbine engine prices in the long term.

The small radial gas turbine and turbocharger are related in that both use a single-stage radial compressor and turbine. This relationship has often led to the misconception that a small gas turbine is really a sophisticated turbocharger with only a combustor and gearbox added.

A cursory examination of the small gas turbine (Figure 1) and a typical turbocharger shows the commonality of the single-stage compressor and turbine, but any other similarity is purely coincidental.

The relative component costs in Table 1 indicate that the rotating assembly and housings constitute one-third of the engine price. However, even if small gas turbines could be manu-

Table 1. Relative Costs of Small Gas Turbine and Turbocharger Components (1982)

Component	Small Gas Turbine, %	Turbocharger, %
Rotating Assembly	23	2.0
Housings	12	2.0
Combustor	7	
Gearbox	18	-
Controls and Accessories	27	-
Assembly and Test	13	1.0
Total	100	5.0

factured in the volume and at the percentage of cost of turbochargers, the price of the turbine would still be more than three times the price of the turbocharger. Presently, the small gas turbine costs 20 times more than the turbochargers.

Using the monorotor and monostator configuration (Figure 13 and Reference 6) is one possible way of reducing powerhead costs up to 15 percent.

High rotational speeds of small gas turbines, 60,000 to 100,000 rpm, result in expensive high speed reduction gearboxes with precision ground gears to obtain output power at speeds compatible with existing driven equipment generators, engines, converters, hydraulic pumps, and accessories. Engine accessories do not scale in cost or weight. Theoretically, the small gas turbine should score a weight advantage according to the square cube law of power ratio. Specialized accessories are not always available and manufacturing constraints prevent direct scaling. The net result is that small APU weight deviates substantially from the square cube relationship (Figure 5).

The increasing demand for built-in test equipment for engine health monitoring, and closed loop acceleration fuel scheduling is spearheading the advent of full authority electronic digital control, even in small, low cost APUs. As a result, the control system and accessories may constitute the major portion of the acquisition cost. The opposite approach is to provide an APU which has to be removed from the installation and returned to the manufacturer for failure determination.

LCC comparisons between existing and advanced fuel efficient APUs for typical airline service indicate the importance of fuel costs. Maintenance and overhaul costs are dependent upon engine complexity and labor rates which are indirectly linked to fuel costs. More sophisticated fuel efficient APUs must be as reliable as their simpler counterparts in order to provide lower LCC. Emphasis towards design to cost, with early and extensive reliability and maintenance testing of advanced APUs is then mandatory, along with the requisite development expenditures.

CONCLUSION

The small single-shaft radial gas turbine APU was first introduced into aircraft service in the early 1950s. Since that advent, extensive design and operating experience has permitted refinement and optimization of the same basic features to higher levels of performance and

Figure 13. Monorotor and Monostator of Titan Gas Turbine

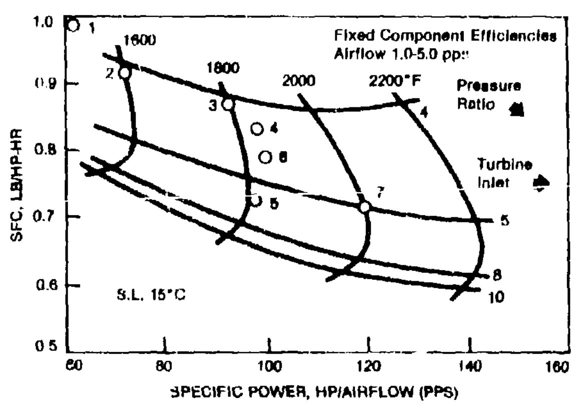
reliability. A direct example of this is the continued development of Solar's Titan T-62 APU product line (Figure 14). The initial T-62T-2 model was rated at 62 kW, sea level (SL), and 15°C conditions. The current Titan T-62T-45 model shown on test in Figure 15, is now rated at 280 kW within the same envelope dimensions, and later prototype engines will be developed up to 370 kW. During this development period, a host of design constraints have been identified and categorized. Dominant in hierarchy are cost, reliability/maintainability, weight, and volume. The bottom line is the LCC for the complete aircraft system of which the APU is only a minor component.

An example of a new hypothetically optimized APU design indicated no apparent weight improvement through the use of higher compressor pressure ratios. On the contrary, however, uprating of an existing APU can be achieved by increasing speed and pressure ratio.

Centrifugal compressor and radial inflow turbine techniques have come a long way since the advent of the Solar's T-41 radial gas turbine in the 1950s. Design techniques now exist to essentially confirm both component performance and life prior to cutting any metal. The progress of technology in the last two decades has reduced potential performance improvements and the rate of return on research and development investment. Performance limitations for centrifugal compressor impellers are discussed in Reference 7 where it is reasoned that the inherent friction limit is being approached for large high performance compressors.

Small radial turbomachinery is more restrained by mechanical and manufacturing limitations; thus, significant improvements are possible. Emphasis upon improving the investment casting technology for thinner blades with better surface finish, plus thermal and dynamic compatibility of the stationary shrouds and rotors is likely to produce higher performance levels.

Performance improvements not only stem from increased component efficiency levels but also from increased component operating ranges, either by decreasing incidence and diffusion effects on both compressors and turbines, or by utilization of variable (low leakage) geometry stators. Increased efficiency of the compressor operating range is particularly important for small radial gas turbines with integral bleed and shaft power output. Compressor range can be increased by designing blade tip sweepback angles of 50 degrees. Higher blade stresses are consequently incurred and better materials must be sought for the task.



	Model	Year	Pressure Ratio	Airflow kg/s	T.I.T., K	kW
1.	T-62T-2	1958	3.5	0.64	1087	62
2.	T-62T-27	1964	4.1	1.00	1142	112
3.	T-62T-40-8	1975	4.1	1.22	1254	186
4.	T-62T-40 L/C	1979	4.4	1.32	1282	215
5.	T-62T-32A	1979	5.4	0.73	1254	112
6.	T-62T-45	198	4.7	1.82	1282	280
7.	T-62T-45 Monorotor	1985	5.0	2.04	1367	370

Engine O/D Constant = 320 mm

Figure 15. T62T-45 Test Installation

As discussed previously, maximum T.I.T. for simple, externally cooled radial turbine rotors bracket the 1350 K mark. Several government sponsored programs (References 4 and 8) are currently being directed towards increasing T.I.T. up to and beyond 1500 K, both with sophisticated, internally cooled, metallic and ceramic, radial inflow rotors. This cooled radial turbine technology could provide higher engine specific power, if the manufacturing costs are not prohibitive.

Continuous duty APUs have not been fully addressed in this discussion, but it is obvious that increased fuel costs will eventually demand higher APU compressor pressure ratios to improve fuel consumption. Acquisition costs of fuel efficient APUs will consequently increase, and adaptations of proven medium-size turboprop and turboshaft engines for APUs will and are beginning to enter the marketplace.

In concert with the increase in fuel efficient APUs, there will be an improvement in the efficiency of secondary power absorption systems and energy conservation. Air conditioning requirements for older commercial aircraft were on the order of 0.5 kg/m per passenger. This could be reduced by 30 percent for future aircraft, which would decrease APU size requirements.

Finally it is suggested that the most profitable development avenues for higher specific power, lower weight, intermittent duty, single-shaft radial gas turbine APUs with low LCC are those focusing upon:

- Higher specific speed, single-stage radial compressors and turbines.
- Higher temperature capability, lower cost hot end materials, such as ceramics.
- High heat release combustors with wider altitude start and operating envelopes.
- Manufacturing techniques to produce near net-shaped components which will minimize machinery costs and save critical materials.
- Computerized design techniques to minimize development risk and time, integrating many of the design constraints discussed herein.

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DISCUSSIONS

R.Smith, US

(1) What is the tip speed of the fuel slinger driven by the electric motor?

(2) Do you run high tip speeds at low rpm and during starting?

Author's Reply

(1) The fuel slinger electric motor operates at a constant speed of 10 000 rpm.

(2) Tip speed is of the order of 25 m/s.

S.S.Stecco, It

According to your fig.5, influence of power on cost, weight and volume, it looks that over 300 KW of power the ratio between power and weight first slows down and then decreases. Is this correct or is it a "drawing effect"? And if it is correct how do you explain it? It seems to me that this ratio should have roughly asymptotic behaviour.

Author's Reply

The effect is correct due to the square cube relationship of power and weight in terms of engine dimensions (size).

K.Mose, Ge

Not knowing the APU's application to the aircraft types, did TURBOMACH ever have problems relative to APU system weight restrictions as asked for by the aircraft manufacturers?

Author's Reply

Most APU-, engine- and aircraft manufacturers have experienced weight problems at one time or another. Our experiences with APUs for large commercial aircraft is limited but nevertheless we have been confronted with excessive weight. The difficulty mainly arises from the inability of accurately predicting complete installation weights, sizing of the cooling system, noise silencing material etc.

A.L.Romanin, US

What material, overall length and diameters of monorotor(s) have been successfully tested? What is the most critical LCF-problem by analysis or test?

Author's Reply

Monorotors of 165 mm and 114 mm have been successfully tested made from UDIMET 700 and hipped IN 792. More specific details can be found in ASME 78-WA/GT-2.

AD P002287

APU IN COMMERCIAL AIRLINE OPERATION

by

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1. Introduction

Early in the sixties, while the jets we operate today were still on the drawing boards, it was apparent that an efficient and rapid ground handling could not be obtained when following the practices common at that time. The demands on ground support become larger than could be handled by most of the existing facilities necessitating modification or, as in most instances, procurement of entirely new equipment.

Therefore and for many other reasons the airlines whole-heartedly welcomed the introduction of the auxiliary power unit (APU), which provides pneumatic and electrical power on the ground for operation of the airconditi ing and electrical systems and for starting the main engines.

Operation and control are fully automatic after start initiation, making a better use of the manpower available.

Since the airplane's air packs are run, the APU can provide a cooler aircraft at boarding time on hot days than ground equipment. Moreover, individual temperature control of the cabin sections is possible.

The APU reduces dependance on ground equipment availability and reliability, thus enhancing on-time performance. It reduces the quantity of required ramp equipment, thus reducing ramp congestion and consequently turn-around time.
Trouble correction on aircraft on the ramp can be performed without the necessity to move mobile units into position, which also means time saving.
It is interesting to note that many systems operate on air pressure as well as electricity each requiring a mobile unit to be moved.
Even a water system may fall in this category.

On some airplanes, for instance the DC9 and the DC10, availability of the APU may be used to replace a failing engine generator during specific flight phases.
Flight flexibility and last but not least safety can be badly hampered by loosing a generator.

For these and not yet envisioned reasons the APU was gladly accepted by the airline as a standard installation of every airplane to come.
The disadvantage of constantly carrying the weight and the costs at that time for maintenance and fuel did not cause special concern.

2. Configurations

At this point it seems appropriate to mention some salient data of the APU's as we use them on our aircraft.

Our first APU was on the DC9. This is built as a single shaft engine with a 2-stage centrifugal type compressor driven by a radial inward-flow turbine. It develops 60 shaft horsepower and 90 lbs of air per minute with a pressure of 47 psi.

The 747 APU contains a 4 stage compressor driven by a 2-stage turbine both axial. The unit drives 2 generators each 90 KVA; it develops 300 shaft horsepower and 500 lbs of air per minute on a standard day.
The entire installation, that is the engine proper, the associate ducting, wiring, monitoring hardware etc. weighs 800 kg.

The DC10 unit consists of a 3 stage axial compressor driven by a 2 stage turbine and a radial compress driven by a one stage turbine. It moves one 90 KVA generator. This installation weighs 500 kg. The unit delivers 142 shaft h.p. and 385 lb/min of air.

In both 747 and DC10 installations the air pressure supplied is 40–42 psi.

The latest APU in our fleet is the one for the A310. This APU is the first in our fleet with a modular construction and consists of a power section, a load compressor and a gearbox, all mounted on a single shaft.

The power section consists of a two stage centrifugal compressor, a reverse flow anular combustor and a 3 stage axial turbine. The single stage centrifugal load compressor is driven directly by the power section and delivers the bleed-air to the aircraft system. The gearbox attached to the load compressor, drives the generator and other accessories.
It delivers 135 shaft h.p., 250 lbs/min air with 43 psi air pressure.

3. Reliability and availability

The APU is not an airworthiness requirement, so it is an airline's choice how to use the APU most economically. Aspects such as passenger comfort, on-time departures and turn-around times as well as possible alternatives play a role in the evaluation against costs and effort to keep the APU in operation.

Based on the airworthiness criterium KLM used to classify the APU as a so-called go-item or at most a consult-item, which meant that a defective unit was not considered prohibitive for departure except for special circumstances like doubtful ground equipment or extreme weather conditions at next station, not to forget the selection of alternate air[illegible]

We also have sufficient examples where airplanes had to be swapped in order to ascertain a punctual flight schedule.
In a case where such an interchange is impossible, dispatch with a fly-away unit is the only alternative. As a matter of fact not a very attractive method, considering the weight of 300 kg. and the dimensions that take a considerable cargo hold space. The unit is kept as simple as possible and is only meant to start an aircraft engine. No electrical power supply or airconditioning is provided.

However, the APU reliability or rather availability is influenced by its status of being a go or a no-go item. We experienced too many crew and passenger complaints due to low comfort.

Consequently we now consider an APU as a no-go item from home base.

A special APU workteam has been instituted, consisting of maintenance and engineering specialists, to overcome the low availability. They monitor the individual APU condition on each aircraft closely, investigate chronic or trend problems in consult also with workshop and operational specialists and try to define possible solutions.
The actual reliability figures are roughly for the DC10 MTBF 3500/MTBUR 1600 (APU)hrs and for the B747 MTBF 5100/MTBUR 2450 (APU) hrs.

Some typical problem areas we experienced, are

1. compressor damage
2. compressor deterioration
3. incorrect troubleshooting

In the spring of 1981 we experienced a rather bad record of unscheduled removals for the DC10. Investigation revealed that deleterious matter enters the APU-air intake and settles in the multiple-bent, long ducting and fouls the compressor blades, thus causing a gradual decrease of performance.

The improvement we achieved in the summer season was attributable to the action, that the ducting is removed after every 1800 flying hours and thoroughly cleaned. Also a performance-trend curve is maintained, using parameters such as N1, N2, EGT and duct pressure, taken after every flight. By accomplishment of the rice-hull cleaning process the APU performance is improved if need so requires.
As can be seen from the MTBF/MTBUR figures something is failing in the troubleshooting or corrective action implementation. I mentioned already our APU workteam.
Further we emphasized the use of testequipment and in particular the FIM (Fault Isolation Module). This is an electronic box connected to the Control Box of the APU, which contains circuits to monitor the protective shutdown functions of the APU. When an automatic shutdown occurs the FIM will display a number or letter, telling which protection was activated.

4. Maintenance and weight costs

For our fleet of 16 Boeings 747 and 6 DC10's yearly expenditure for maintenance is USD 750.000 based on an expense of approx. USD 11 per flying hour being about equal or slightly lower than figures of other major airlines.
In addition we have to take into account approximately USD 125.000 to carry the weight of these installations as well as the extra fuel that has to be lifted.

5. Fuel costs

The costs of a three quarter of a million dollars for maintenance as mentioned before is surmountable, but the situation have changed dramatically since the fuel prices started to escalate.

The 747 APU gorges 600 liters of fuel per hour under normal load conditions e.g. when supplying power for electricity and air for the airplanes pneumatic systems and airconditioning. This consumption will be a little less of course when extracting either electrical power or air. The figures are then 425 and 550 l/hr respectively.

The DC10 APU is comparatively modest with 300 l. for normal load, 140 l. for electrical power and 230 l. for air supply.

The A310 APU needs approximately 200 l. per hour to supply its air and shaft horsepower.
Recently, based on a fuel price of USD 1,20 per USG KLM has to spend 8 million dollars per year to keep these precious machines going in the 747's and DC10's. The costs for our 28 DC9's and yet two Airbusses does not leave much to be imagined anymore.

6. Fuelsaving measures

It stands to reason that these exorbitant expenses have set the airlines about studying ways to decrease this consumption.

Even the removal of the APU all together is contemplated.
The various airline-studies that were recently brought forward, clearly indicate that this matter is complicated as could be expected. The problems are manifold and different for every single airline and for every airport.

There are factors such as:

- the availability and capabilities of the necessary groundequipment of a company along its network;
- the climatological circumstances where hot and cold temper res require reliable and powerfull units.

Until today we do not know of any airline that has decided to remove the APU. To the best of our knowledge only one airline has proposed to commence a test on a limited scale with some freighter aircraft that remain close-to-base.

Since it is extensively cheaper to run groundequipment rather than the APU, airline instructions are issued that primarily aim at a drastic limitation of APU employment and use of the ground support instead, even though the capabilities are at times insufficient to provide the required comfort. Again the inconvenience of apron clutter, noise and pollution associated with multiple pieces of ground support equipment has to be accepted.

These inc pabilities will not only be manifest under extreme conditions. As a matter of fact, stations in the Far, Middle and Near East as well as in Africa and South, Middle and Southern part of North-America are often too warm and in Northern USA too cold to handle with even the maximum possible number of Airco units. Only Europe is considered moderate in our network!
In many transit stations passengers remain on-board requiring full airconditioning.

Dependent on circumstances such as directives by airport authorities and the preference to certain procedures of the airlines a variety of departure and arrival procedures has been noticed,

- Power-back instead of push-back from the boarding gate is done by some airlines in the U.S.A. This is carried out on reverse power of the main engines. It is claimed that this procedure offers a more rapid departure than push-back, that less ground personnel, ground equipment is required and thus the entire operation will be cheaper.
- Other companies like KLM use the push-back method and start the engines while being moved backwards. Time between off-blocks and roll-out is shorter. Shorter usage of APU and engines is claimed.

- Regardless of the selected method some operators taxy out with one or two engines still dead. These are started shortly before reaching the take-off position.
- In many cases the APU is started about 8 to 10 minutes before departure with the sole reason to assist engine starting.
- The APU is on while taxiing to the gate after landing, except at turn-around stations where ground power is used.
  On behalf of passenger comfort and in order to meet our scheduled tight transit-stoptimes we run the APU during the stay-over unless the airport rules prohibit such a procedure. Particularly airports close to a dense population have very stringent rules. Since the APU is sitting high above the ground (3 - 9 meter) its noise is carried out over a considerable distance. While incessantly running at approx 40.000 rpm it can become quite a nuisance.

- Sometimes the aircraft is towed-in to the gate after arrival. APU has to be switched on if the tractor does not carry a ground power unit.

However efficient these and other procedures may be, the exorbitant fuel costs have urged the industry to search for cheaper techniques for airplane groundhandling. The most economical way to meet that requirement will be a fixed ground-installation, which can operate on diesel fuel or on purchased municipal electrical power.

Such a centralized system feeds every boarding gate with electrical power and air for the airplane's airconditioning system and for engine starting. If climatological circumstances permit it is possible to provide electrical power only, leaving the air supply to mobile equipment if needed. The facility to start engines makes the installation complicated and expensive.

Estimates indicate that the costs of an all-in system per hour will be about one fifth of the APU and about equal or a little less for the mobile units providing the same services. Consequential to these savings the building of fixed systems was started a couple of years ago. More than 100 airports in the U.S.A. provide this service already but also in other countries we will find these facilities either available or under construction.

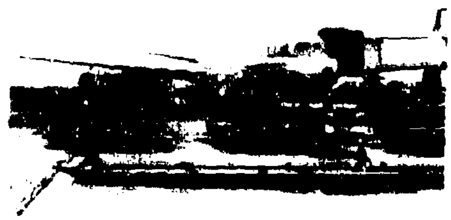
7. APU or not?

Knowing that the APU is the most expensive support device, the temptation is strong to entirely remove it from the commercial transports.
Even though it may be repetitive, a couple of adverse consequences are summarized such as:

- loss of the APU puts the burden on the ground equipment, of which the maintenance at the line stations leaves sometimes much to be desired;
- for some stations the removal of the APU could require a sizeable investment of ground support equipment;
- loss of the ability to start engines during or after push-back;
- increased engine operating time at the gate;
- engines have to be kept running after arrival until ground equipment is hooked up;
- last but not least the risk of deficient comfort for passengers at certain stations in hot or cold weather.

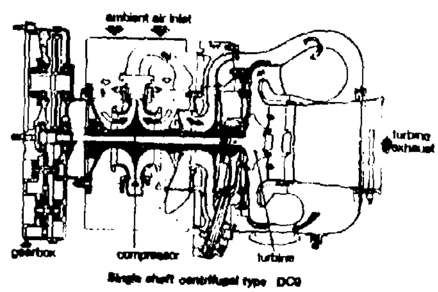
It is our firm opinion that KLM cannot permit the removal of the APU since this is the only device that provides prompt and reliable service, independent of other facilities. Since KLM has a route network to every corner of the world we will have to live with these little giants for many years to come.

A GENERAL OVERVIEW HOW WE, AIRLINES, ESPECIALLY KLM, LOOK AT THE APU.

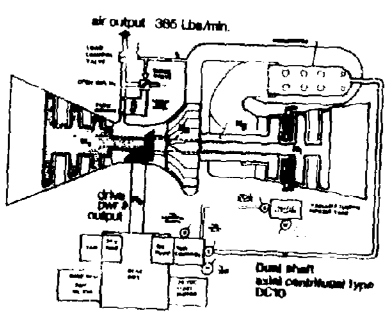


AN EFFICIENT AND RAPID GROUND HANDLING COULD NOT BE OBTAINED WHEN FOLLOWING THE PRACTICES OF THE EARLY SIXTIES.

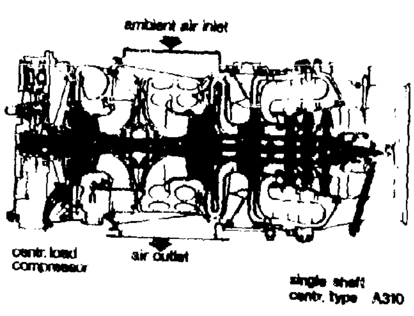
THE AIRLINES WHOLE-HEARTEDLY WELCOMED THE INTRODUCTION OF THE AUXILIARY POWER UNIT.



OUR FIRST APU WAS ON THE DC9.



THE DC10 UNIT.



THE LATEST APU IN OUR FLEET IS THE ONE FOR THE A310.

A/C	type shaft	compr. stages	turb. stages	shp	air outp. lbs/min	psi	weight kg
DC9	single	2 centr.	1 rad. inflow	60	90	47	150
747	single	4 axial	2 axial	300	500	41	800
DC10	dual	3 1 ax/cen	2 1 ax/ax	142	385	41	560
A310	single	1 centr. load / 2 centr.	3 axial	135	250	43	227

APU configurations

SOME TECHNICAL DATA OF OUR APU'S.

SOMETIMES DISPATCH WITH A FLY-AWAY UNIT IS THE ONLY ALTERNATIVE.

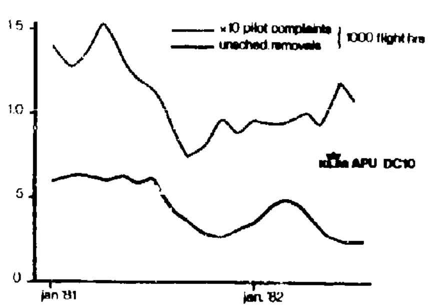
AIRCRAFT OPERATIONS MANUAL 747	2.8. DISPATCH DEFICIENCY GUIDE 2.8.1 APU
- Auxiliary Power Unit	Must be serviceable: . For departures from Schiphol for Middle & Far East and Middle & South Atlantic flights. . For departures from Schiphol for destinations in the North-Eastern region of the United States, Canada and Alaska during the winter period. . When landing(s) can be anticipated at airport(s) without required ground equipment (e.g. LAX and [illegible]). May be unserviceable for other routes.
APU FAULT Light	May be unserviceable provided a qualified operator is in the cockpit during APU operation.
- APU Oil Quantity Indicator	May be unserviceable provided, oil quantity is checked prior to departure.
- APU RPM Indicator	May be unserviceable provided one APU generator frequency indication is serviceable and EGT is monitored during APU acceleration
★ - APU Inlet Door	May be unserviceable in the fixed closed position provided APU is not used. Refer to Maintenance Manual chapter 49. May be unserviceable in the open or partially open position provided the performance limited weight for take-off and landing is reduced by 318 kg and the trip fuel is increased by 1%. If door is unserviceable in the full open [illegible]

WE NOW CONSIDER AN APU AS A NO-GO ITEM FROM HOME BASE TO ALMOST ALL ROUTES EXCEPT NORTH-AMERICA IN SUMMERTIME.

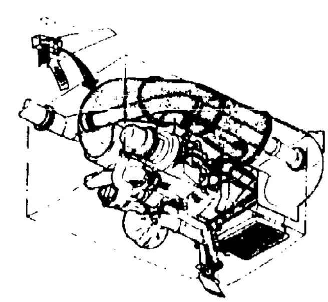
typical problems :

- **compressor damage**
- **" " deterioration**
- **incorrect trouble shooting**

SOME TYPICAL PROBLEM AREAS WE EXPERIENCED.



IN THE SPRING OF 1981 WE EXPERIENCED A RATHER BAD RECORD OF UNSCHEDULED REMOVALS FOR THE DC10.



INVESTIGATION REVEALED THAT DELETERIOUS MATT[illegible] ENTERS THE APU-AIR INTAKE AND SETTLES IN THE MULTIPLE BENT, LONG DUCTING.

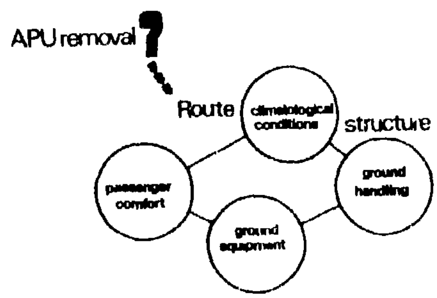
FOR COMPRESSOR DETERIORATION A PERFORMANCE-TREND CURVE IS MAINTAINED.

APU costs	$	maintenance	weight
fleet: (16) 747 + (6) DC10	per year	750,000	125,000
	per fl.hr.	11.00	1.80

MAINTENANCE AND WEIGHT COSTS, STILL SURMOUNTABLE.

	APU fuel consumption S.L. 100°F ltr/hr			annual costs
	norm. load	pneu. only	el.pwr. only	$ 8 million (120/USG)
747	600	550	425	
DC10	300	230	140	
A310	200	160	145	

THE SITUATION HAS CHANGED DRAMATICALLY SINCE THE FUEL PRICES STARTED TO ESCALATE.

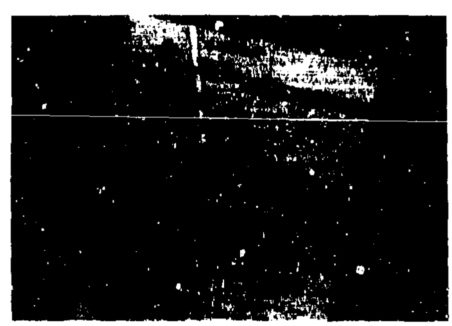


THE REMOVAL OF THE APU, A COMPLICATED MATTER, DIFFERENT FOR EVERY SINGLE AIRLINE AND AIRPORT.

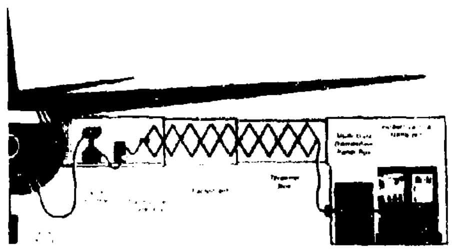
fuel saving measures:

- power-back instead of pushback
- engine start during pushback
- taxi out with dead engine(s)
- limited use of APU:
- shutdown during tow/taxi-in
- engine start only

A VARIETY OF DEPARTURE AND ARRIVAL PROCEDURES HAS BEEN NOTICED.



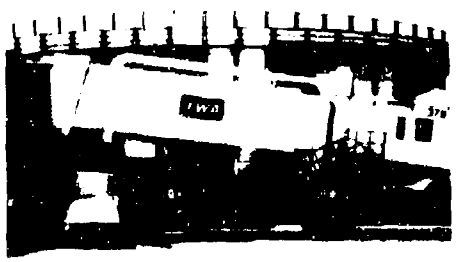
THE KLM POLICY IS AS FOLLOWS: THE APU IS NOT RUNNING WHILE TAXIING TO THE GATE AT SPL AND TURN-AROUND STATIONS, WHERE GROUND POWER IS USED. AT DEPARTURE APU WILL BE STARTED DEPENDING WEATHER CONDITIONS AT TRANSIT-STOPTIMES WE RUN THE APU UNLESS THE AIRPORT RULES PROHIBIT SUCH A PROCEDURE.



Fixed Ground Power system

THE MOST ECONOMICAL WAY WILL BE A FIXED GROUND-INSTALLATION, WHICH CAN OPERATE ON DIESEL FUEL OR ON PURCHASED MUNICIPAL ELECTRICAL POWER.

FIXED ELECTRICAL POWER.



FIXED AIRCONDITIONING SYSTEM.

no APU ...

ground equipment: -maintenance
-investment

increased: -en_ ne operation at gate
-turn aroun time

decreased: -pax comfort

REMOVING THE APU WAS A COUPLE OF ADVERSE CONSEQUENCES MENTIONED HERE IN SUMMARY.

DISCUSSION

E.H.Warne, UK

Do you feel it is justified to taxi out without all engines started since a failure to start at the main runway can affect other flight operations.

Author's Reply

Failure to start during taxi-out will be inconvenient to the own airline, due to extra delay time and gives extra congestion of taxiways.

Starting the engines at the main runway is unacceptable for the engines. They need at least 3 minutes for warming-up before T.O. thrust setting.

So, taxi-out with one or more engines off, only makes sense with long taxi times.

KLM will start all engines before taxying out.

P.Vaquez, Fr

I suppose your APU's are "on condition"? What do you expect from trend monitoring?

Author's Reply

Yes, our APU's are on condition.

Trend monitoring is used for determining the health of the APU. If the trend shows a deterioration, timely maintenance actions (e.g. compressor cleaning, component- or APU-changes) will be initiated to preclude en-route breakdown and/or major damage.

C.Rodgers, US

(1) Are you advocating changing the APU role from auxiliary to "emergency" power generating role?

(2) If the APU is to be used sparingly would you recommend future APU's be designed for higher reliability and lower cost or still for improved fuel economy?

Author's Reply

(1) We do not consider our passengers' comfort an emergency! Like already mentioned in our presentation, the APU will be necessary as long as adequate ground equipment is not available all over the world. Since this will be the case in the foreseeable future, the APU must be able to fulfill the functions like they do today.

(2) Although the use of the APU can be reduced further in future, fuel will still account for some million dollars per year. In commercial airline operation passenger comfort and punctual flight schedule requires high reliability and fuel and low-reliability must be paid for the whole aircraft life. For this reason high reliability and improved fuel economy will both have the highest priority.

AD P002288

AUXILIARY POWER UNITS FOR WIDE-BODY AIRCRAFT

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1. INTRODUCTION

This document is prepared by A DYNE pooling of common economic interests (GIE) grouping Turbomeca and ABG/SEMCA companies for design and manufacture of Auxiliary Power Units (APU).

The auxiliary power units designed for wide-body aircraft shall closely meet the requirements concerning, ~~more particularly~~ the reduction of consumption, weight, dimensions and maintenance cost.

Meeting these imperative requirements is obtained thanks to a highly detailed analysis of the various aircraft power requirements and utilization of a load compressor and a digital-type regulation.

Further improvements in the near future consist in ~~the~~ optimization of the APU characteristics by using a free turbine generator and a load compressor flow regulation which takes the variable air requirements of the aircraft into account.

2. DESCRIPTION OF EXISTING AUXILIARY POWER UNITS TYPE AST

The AST auxiliary power units are designed to be installed on wide-body military and civil aircraft. The range from AST 600 to AST 950 covers a power range from 400 to 700 kilowatts.

These auxil ower units consist of modules and are mainly composed of the following elements :

- Power section
- Accessory gear box
- Load compressor
- Digital electronic regulation.

This modular concept as well as the complete interchangeability of each module without adjustment enable maintenance optimization.

The AST family which comprises power generators Astazou III, Astazou XIV, Astazou XVI, Astazou XX, Arriel, TM 333 and TM 319 includes two types of architecture in order to meet the various installation requirements :

- The in-line assembly formed by the gas turbine engine, the load compressor and the accessory gear box

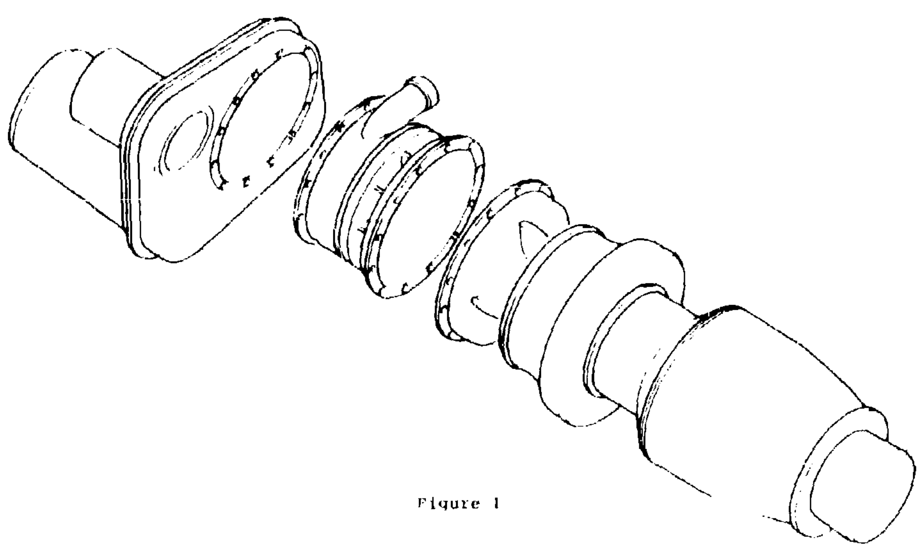


Figure 1

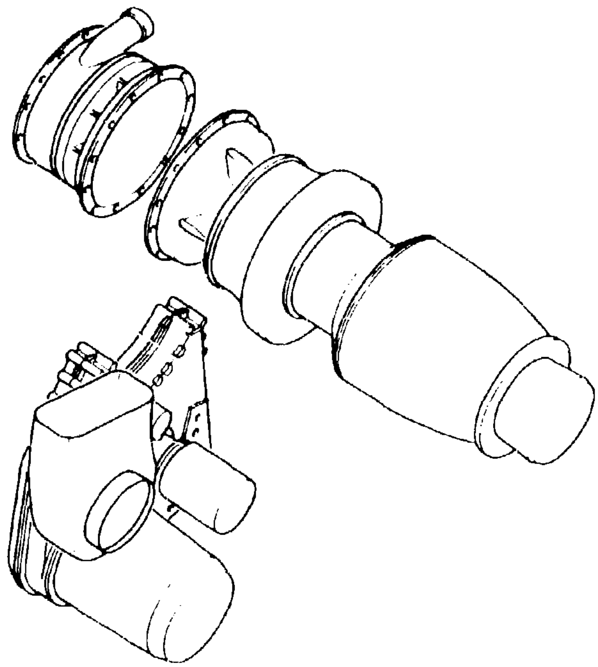


Figure 2

- The assembly formed by the gas turbine engine, the offset accessory gear box and the load compressor.

2.1 Power section

Derived from existing engines used for powering helicopters, the gas generator is composed of :

1 An air inlet.

2 An assembly of one, two or three axial compressors in accordance with the type of generator in use.

3 A centrifugal compressor.

4 An annular or reverse flow combustion chamber fitted with a centre or lateral injection according to the type of engine in use.

5 An assembly of three axial turbines.

6 A shielding ensuring the turbine stage containment.

Especially for the AST 950 built around the Astazou XVI, the internal generator layout is defined by the following section view :

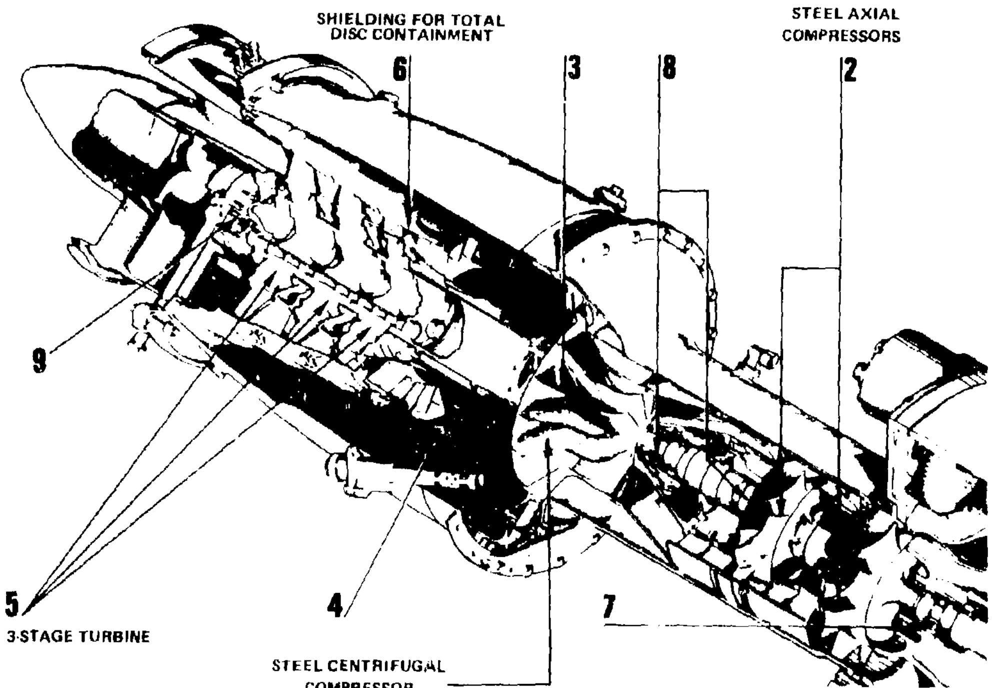


Figure 3

Axial compressors are supported by two bearings 7 and 8 and centrifugal compressor-turbine assembly bearings 8 and 9.

The containment capabilities of all the generation rotary parts, have been proved by full scale tests.

2.2 Load compressor

This module is composed of an advanced-technology centrifugal compressor stage which is directly driven by the generator shaft.

It is supported by two ball bearings.

The supply airflow includes mobile guide vanes which control the compressor performance while ensuring prerotation at the impeller inlet.

The position of these vanes is permanently monitored by the digital regulation module. At constant rotational speed, this regulation provides a quantity of air in conformity with the aircraft requirement in order to minimize the power consumption. Upon the APU starting phase, the vanes are closed to limit the power draw from the generator.

These vanes are driven by a proportional control actuator as well as the air discharge valve connected to the load compressor outlet and avoids surge during extreme operating phases.

The utilization air bleed system consists of a device measuring the delivered air flow and pressure ratio for controlling this module. A shut-off valve isolates this system at the aircraft system interface.

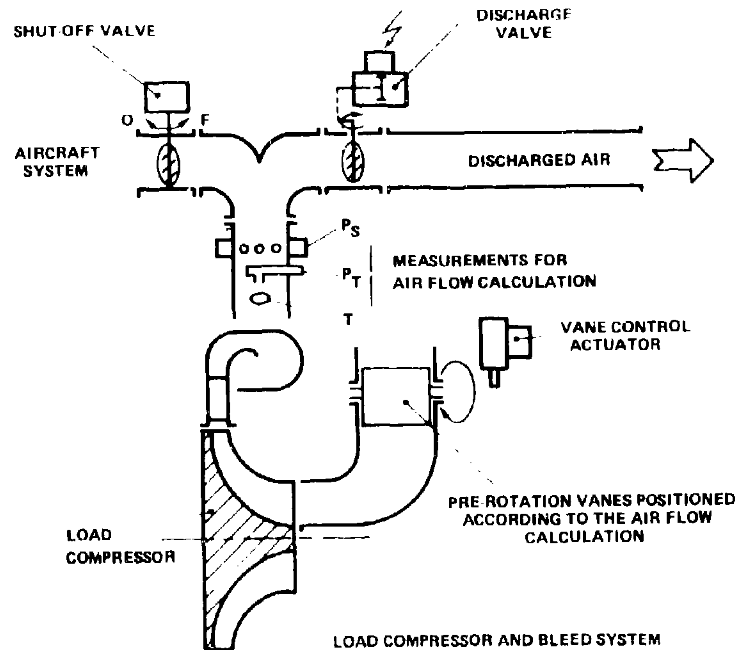


Figure 4

2.3 Accessory gear box

The purpose of accessory gear box is to transmit the power available at the generator to the various connection flanges linked to the various accessories used for aircraft and APU operation ancillary equipment.

The accessory gear box consists of spur-gears stages lubricated with pressurized oil and delivers power to the following equipment :

- aircraft ancillary equipment

  . Generator
  . Hydraulic pump, if required

- APU ancillary equipment

  . Starter
  . Fuel pump
  . Oil pump
  . Oil system cooling fan
  . P.M.G.

When the equipment uses an offset accessory gearbox, the primary reduction gearbox is installed between the generator and the driven modules.

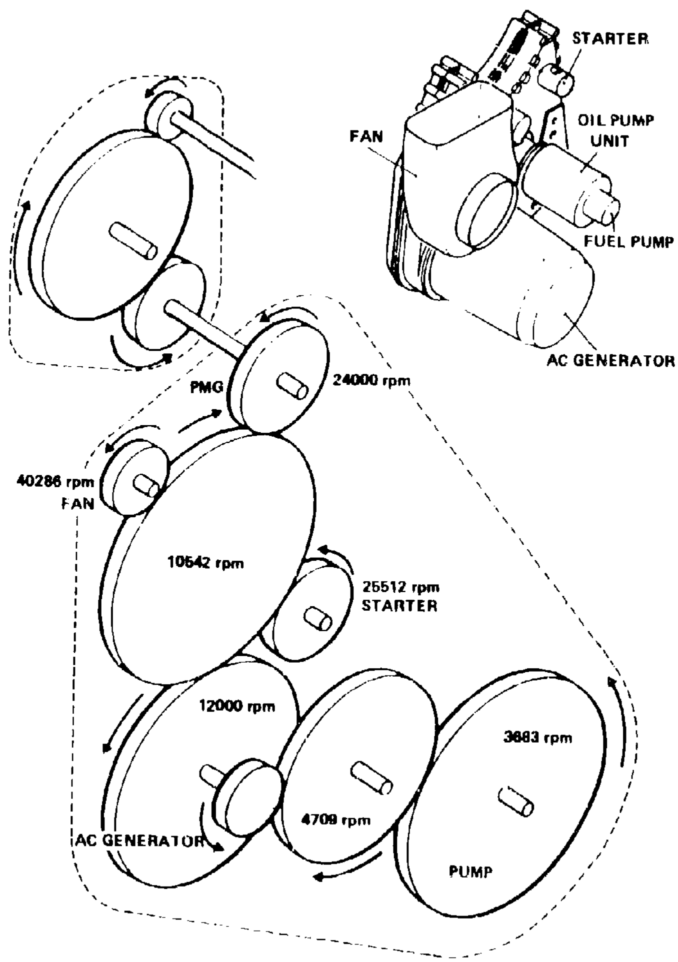

STARTER
OIL PUMP UNIT
FAN
FUEL PUMP
AC GENERATOR
PMG
24000 rpm
40286 rpm
FAN
10642 rpm
25512 rpm
STARTER
12000 rpm
3683 rpm
4709 rpm
AC GENERATOR
PUMP

Figure 5

2.4 Digital electronic control module

The control module, which can be divided into 2 units according to the aircraft installation requirements ensures two main functions :

1 The microprocessor regulation function, the measurement acquisition and processing and the power controls.

2 The parameters monitoring, safeties and maintenance aid. A second microprocessor covers digital processing for these functions and controls the failure store BITE system.
The data relative to failures and their monitoring function can be displayed on the control module front panel or transmitted to the aircraft monitoring system through an ARINC 429 type connection.

3. CHOICES FOR THE FUTURE

3.1 Reduction of the rated power by a detailed analysis of the various power requirements

After describing a modern APU such as that proposed, let us study how it can be improved in the future.

The first user's concern is the gain in weight.

It is worth noting that the weight to be considered includes the weight of the APU itself (power generator, accessory drives, load compressor...), of the driving accessories (starter, fuel and oil pumps, radiators, fans, regulation...) driven accessories (hydraulic pumps, ac and/or dc generators...), of the fuel, of the APU casing.

Consequently, common efforts from the APU supplier, accessory manufacturers and aircraft manufacturer shall result in weight gain.

For information, figure 6 indicates the weight of the various elements in list.

Obviously fuel weight can considerably vary according to the type of operation.

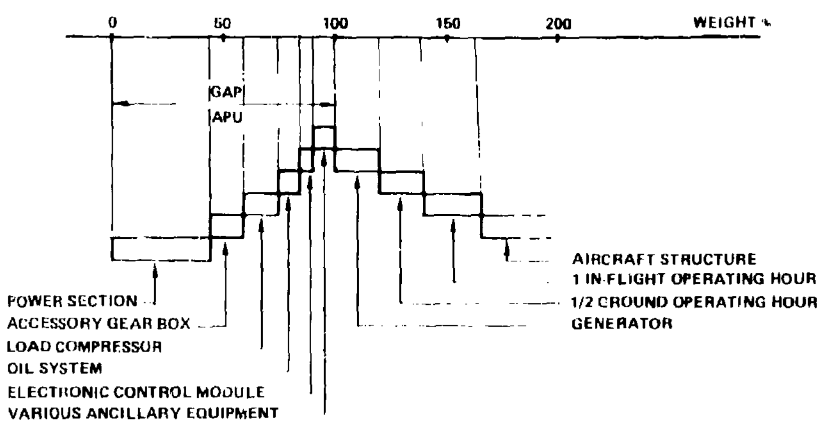


Figure 6

The first solution to gain weight is to install a smaller APU. This is made possible by reducing the power demand in the most critical configuration.

This often occurs with an electrical supply demand, should one of the main power sources fail during high altitude cruise. Power saving can be quite appreciable since every kw saved at 35,000 feet permits to use a generator supplying 1 kw less at peak rate together with a power section providing 3 kw less on ground (air density ratio).

This operation also enables the consumption to be reduced in the most current operational configuration, at low altitude and at a nearly standard temperature : operating with a power value close to the rated power results in specific consumption saving.

Let us take an example, with a single shaft turbine gas generator operating in these conditions 30 % below its rated power. If, while supplying the same power, the rated power is reduced by 1 %, the specific consumption is reduced by 0.5 % approximately. This means that if the extreme requirement (which is used rarely during the aircraft service life) has been overestimated by 10 %, this will result in a consumption increase of 5 % per day of utilization on ground.

It is nevertheless recommended to be careful, since the power consumption of an aircraft generally tends to increase with its evolution.

Another operational case determining the APU dimension is the supply of compressed air to start the main engines in "high and hot" conditions. For this case, the load compressor can be slightly undersized, withstanding an APU overspeed when this case actually occurs : in the operational zone of a compressor used at this moment, the flow increase in relation to the speed is significant (approximately a 2 % flow increase for a 1 % speed increase) to make this system efficient. Once again care must be taken where new dimensions are required for certain generator parts and which could result in a frequency increase of the current supplied by a simple generator.

This method is now being adopted for some APU.

3.2 Utilization of a free turbine auxiliary power unit

The power generator weight and consumption can simultaneously be reduced by using an engine featuring advanced technology. For design leadtimes and costs reasons and to readily obtain a high reliability and a long service life, an APU is often derived from an existing gas turbine engine.

This particular case raises a problem.

Our high efficiency modern engines (MAKILA, TM 333,...) are of the free turbine type whereas, up to now, single shaft turbine engines (ASTAZOU family) were in use.

Such a modification is beneficial at nearly all levels :

- Weight : Gain in weight in the power section due to its advanced concept (5 to 10 %)

Consumption saving at partial power due to the utilization of a free turbine (5 % at 30 % of rated power, at equal technology).

Consumption saving at all speeds due to advanced concept (10, 15 % or more).

NB : These two gains are added.

- Power section dimensions slightly reduced (advanced design).

- Operation : Starting conditions improved since during this phase, the accessory gearbox and ancillary equipment are not driven. This results in a reduction of service-time and a gain in weight on the starter.

Nevertheless, the response to an instantaneous generator loading is not to be disregarded. With a single shaft turbine, the unit rotates at constant speed and the power varies only in accordance with the fuel flow variation. With a perfect regulation, the speed deviation might be null and in reality, the value easily remains within the $\pm$ 4 % tolerance required by operators. With a free turbine, the power varies by increasing the gas flow through turbine, which presently implies a power section acceleration.

Figure 7 gives an example of this type of response.

It represents a turbine of an older type (TURBOMECA TURMO III C4) with a rated power of 1,000 kw driving a 500 kw generator.

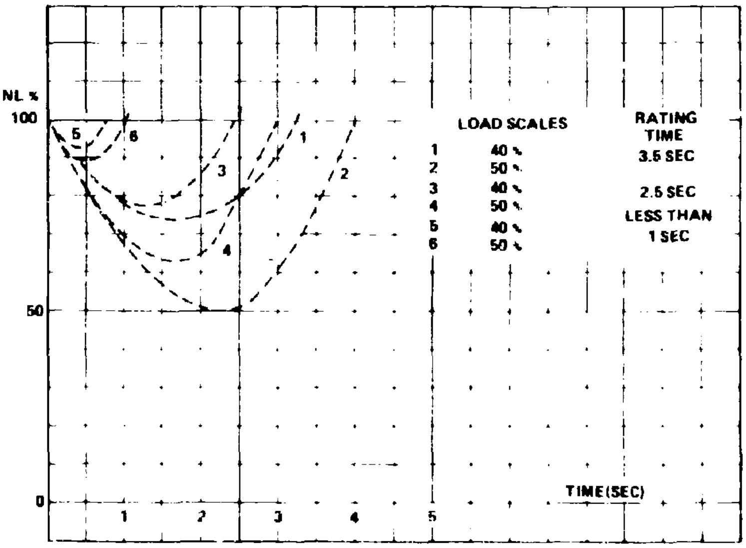


Figure 7

The best acceleration time we can obtain on ground with this type of engine is 3.5 s between 0 and 1,000 kw. When applying a load of 400 to 500 kw to the a.c generator instantaneously, response curves 1 and 2 are obtained. These curves respectively show transitory speed drops of the generator by 26 and 50 %.

When improving the engine with currently used methods (reduction of the rotary assembly inertia), the acceleration time can be reduced to 2.5 s and to obtain curves 3 and 4 with transitory drops by 22 and 37.5 %.

If modern current gas generators can withstand such input speed variations without any excessive additional weight, no further step is being required. If not, it will be necessary to find the way to reduce the power variation time significantly. The relevant studies are now in progress.

The power generated by the free turbine depends on the gas flow, expansion ratio and blade angle. At present, the first two parameters are made to vary while varying the speed of the gas generator ; the third parameter is determined by the distributor geometry.

It is also possible to maintain the free turbine speed constant or nearly constant and vary the air flow with vanes placed in front of the compressor and the blade angle on the free turbine with a new set of vanes.

The power variation is then obtained by simultaneously varying the setting of the two vane sets and fuel flow.

The first tests prove that times for full power initiation can be around 0.3 s. Such a result applied to the previous example gives curves 5 and 6 with transitory speed drops reduced to 7 and 11 %.

An additional advantage is that fluctuations of full power initiation time decrease with altitude.

3.3 Installation and maintenance improvement

A significant improvement of the specific consumption can be obtained by using a turbine outlet/compressor outlet heat exchanger, but with the present technology, the weight and dimensions of such a system are excessive.

As far as the driven equipment are concerned, significant weight gains are presently obtained : the weight of recent generators is less than half the weight of old generators, for the same power output.

New gains will be obtained by increasing the speed of rotation.

Improvements can also be brought up by the aircraft manufacturer.

For example if it facilitates installation, the location usually chosen for the APU (in the fuselage rear cone) is certainly not quite satisfactory for in-flight operation : the pressure is generally higher at the turbine outlet than at the compressor inlet which is in a thick boundary layer with a ram air pressure recovery null or negative. Moreover, it should be reminded that a gain of 1 % on the inlet pressure gives a gain of 2 % of the maximum power and a gain of 1 % of the specific consumption. It is to be added that any favorable difference in pressure results in a considerable cooling improvement.

Simplifications are already in progress. The control panel is simple, it comprises an on/off control, an indicator light and a fault light.

.dditional data can be displayed on one of the flight compartment screens through regulator on crew request.

The control module detects and analyses f tional faults (it particularly monitors the indications provided by each measur; ; sensor) stored for maintenance purposes. It also counts down the service life of the unit components into number of cycles and operating time, alloted with a temperature coefficient if necessary. Obviously, this method can be improved but its principle is establis d.

Scheduled maintenance is limited to a check of the oil level and magnetic plugs and, if required by the user, it can be extended to oil sampling for spectographic analysis.

The APU is modular without excess ; it comprises three modules : the power section the accessory gearbox and the load compressor.

In case of utilization of a free turbine generator, this one can be divided in two modules, one hot and the other cold. Each module is interchangeable without adjustment of the assembly.

It is necessary that the time required for the APU removal be very short (> 30 mn) and that the operations carried out in-situ be limited to the replacement of a measuring sensor or of a small accessory. For a major action like replacement of a module, it is easier and quicker to perform it on workbench.

:.4 Optimisation of the APU air bleed

:.4.1 General

The main advantage of the load compressor mechanically driven by the power generator unit is in that it can be suited to all the aircraft requirements thanks to the regulation covered by the air intake vanes.

Another advantage is the pos. bility for the load compressor regulation to be entirely independent of the aircraft.

This is made possible by a good integration of the APU into the aircraft an accurate knowledge of the aircraft actual requirements stored in the electronic cartrel module.

:.4.2 Utilization system (Airbleed) see fig. 8

The two main systems using pneumatic power in the aircraft are the main engine starting system and the air conditioning system.

* Starting system

The main engine starting system consists of a fixed sonic port through which the air delivery is the greatest in order to obtain the shortest possible engine starting phase. Generally, it is then considered that this requirement determines the maximum capacity of the load compressor. The very short duration of this phase does not require the unit consumption optimization.

* Air conditioning system

Associated with the digital computers, the aircraft air conditioning system can vary its flow requirement in relation to the thermal load of the cabin or flight compartment. The air flow supplied to the air conditioning unit(s) is limited by an adjustable flow regulation valve located upstream the air conditioning system. This valve is controlled by a signal released by zone temperature regulators.

Thus it can be noted that the air requirement of the air conditioning unit(s) varies in flow and pressure in a significant way, according to various aircraft configurations.

The long duration of this phase within the APU operating time results in the optimization of the air supply to the aircraft with minimum losses (the flow regulation valve of the air conditioning unit should be as much open as possible) and a maximum compressor efficiency.

3.4.3 Load compressor characteristics – Operating points

The characteristics of the variable-vane load compressor are such that it is possible to obtain a very wide operating range for flow as well as for pressure, without discharging air for protection of the compressor impeller against surge.

Diagram in figure 9 defines the possible operating range within the unit temperature range.

This diagram also shows the typical pneumatic bleed points on aircraft. It can be noted that the air requirement for starting the engines is much higher than the air conditioning points (except cabin heating during cold weather).

Air bleed points for air conditioning can appreciably vary in flow (from 70 to 100 % of rated flow) and pressure (according to outside temperature).

3.4.4 Load compressor regulation

With this type of regulation, our aim is that unit should remain entirely independent of aircraft controls.

The purpose of the pressure ratio and flow detection system consists in determining the position of the operating point of the compressor within the compressor diagram at any time. When establishing one of the parameters, the pressure is regulated to a reference value and the compressor outlet flow is automatically adapted to the permeability of the downstream system. The inlet vanes are controlled by the regulation module to maintain the outlet pressure at the reference value.

In the air conditioning configuration, this type of regulation has been chosen in order to observe at any moment the permeability variations of the downstream system (position of the flow regulation valve of the air conditioning unit), while maintaining a fixed pressure level.

This reference pressure level can vary in relation to the data received by the regulation module (external pressure, external temperature and it is then possible to optimise the air delivery for air conditioning in the various attitudes of the aircraft.

In the case of a high air flow requirement (aircraft heating or main engine start), the permeability of the system downstream of the compressor is greater than when the flow regulation valve of the air conditioning system is open. Above a certain compressor outlet flow, a reference value of the flow-pressure is ensured at its outlet ; this reference value is such that the compressor corresponds to a higher permeability of the downstream system. Consequently the compressor vanes will fully open thus allowing for rapid heating of the cabin or optimum starting of the main engines.

Figure 10 represents the reference operation line (for a given altitude and temperature) that the load compressor regulation will follow to control the inlet vanes according to the aircraft system requirements.

3.4.5 Machine protections

The APU control module comprises two control systems for protection purposes.

* Protection against the load compressor surge

In case of aircraft system air bleed lower than the compressor possibilities, a discharge valve, simultaneously slaved to the main regulation by the compute protects the compressor against surge. The difference between the required flow and the possible minimum compressor flow at the reference pressure only is discharged.

Figure 11 represents such a functional configuration. The actual operational point of the compressor is in B whereas the requirement is in A.

* Protection against the thermal engine overload

During starting phases of main engines in hot weather conditions using the electrical supply from the APU generator, it may be possible that the hot parts of the gas turbine engine be in thermal overload condition. In order to avoid such a problem, the computer permanently determines the thermal load value and compares it with the limit value ensuring the unit service life. If this difference becomes positive, a signal is sent to the load compressor air inlet vanes to close by a value such that this difference is cancelled or becomes negative as long as the electrical supply from the machine is not reduced.

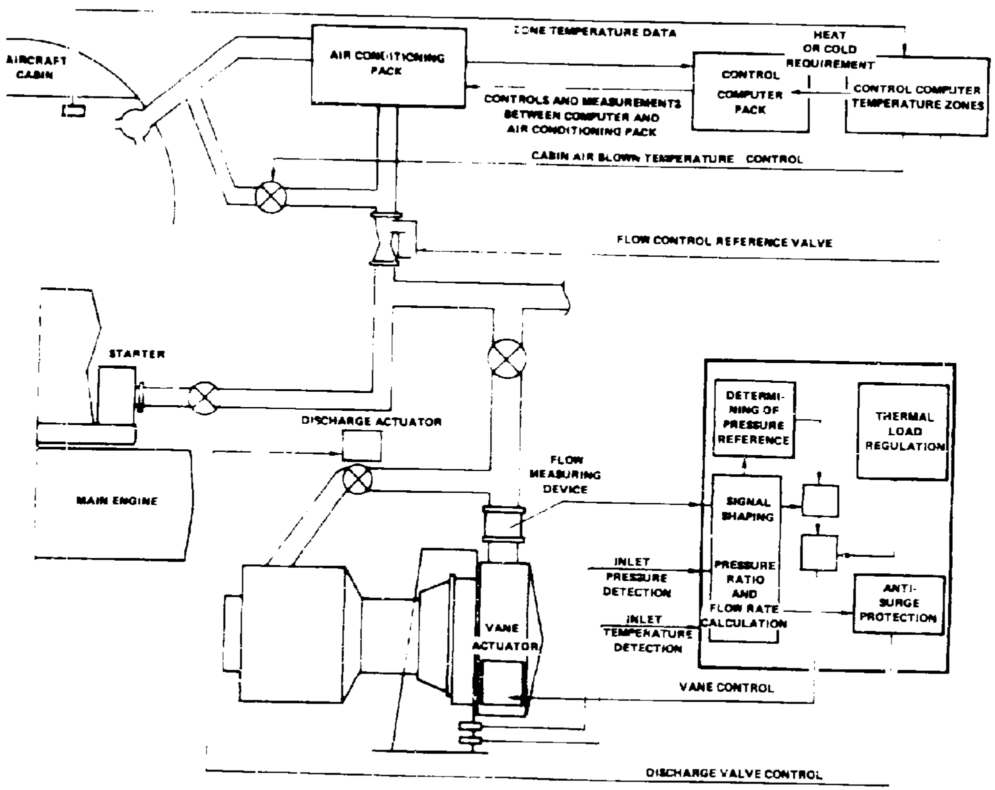
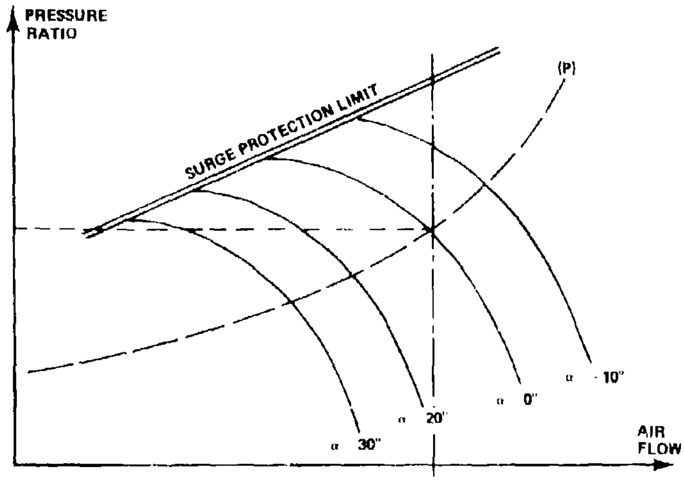

ZONE TEMPERATURE DATA
AIRCRAFT CABIN
AIR CONDITIONING PACK
HEAT OR COLD REQUIREMENT
CONTROL COMPUTER PACK
CONTROL COMPUTER TEMPERATURE ZONES
CONTROLS AND MEASUREMENTS BETWEEN COMPUTER AND AIR CONDITIONING PACK
CABIN AIR BLOWN TEMPERATURE CONTROL
FLOW CONTROL REFERENCE VALVE
STARTER
DISCHARGE ACTUATOR
MAIN ENGINE
FLOW MEASURING DEVICE
DETERMINING OF PRESSURE REFERENCE
THERMAL LOAD REGULATION
SIGNAL SHAPING
PRESSURE RATIO AND FLOW RATE CALCULATION
ANTI-SURGE PROTECTION
INLET PRESSURE DETECTION
INLET TEMPERATURE DETECTION
VANE ACTUATOR
VANE CONTROL
DISCHARGE VALVE CONTROL

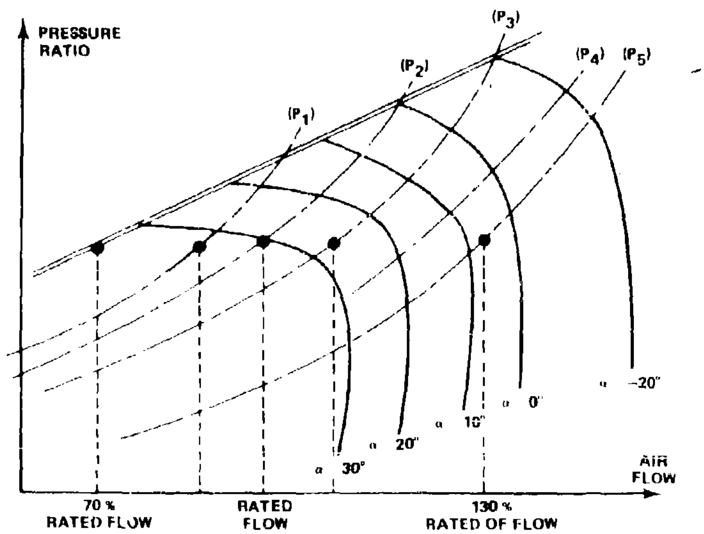
Figure 8



α : ANGLE OF THE LOAD COMPRESSOR VANES

(P) : CURVE DETERMINING DOWN STREAM SYSTEM PERMEABILITY

Figure 9



(P_1), (P_2), (P_3) : DIFFERENT AIR CONDITIONING SYSTEM PERMEABILITIES

(P_4) : CHARACTERISTICS OF THE MAIN ENGINE STARTER

(P_5) : PERMEABILITY OF AIRCRAFT HEATING SYSTEM

(α) : ANGLE OF LOAD COMPRESSOR INLET VANES

Figure 9 bis

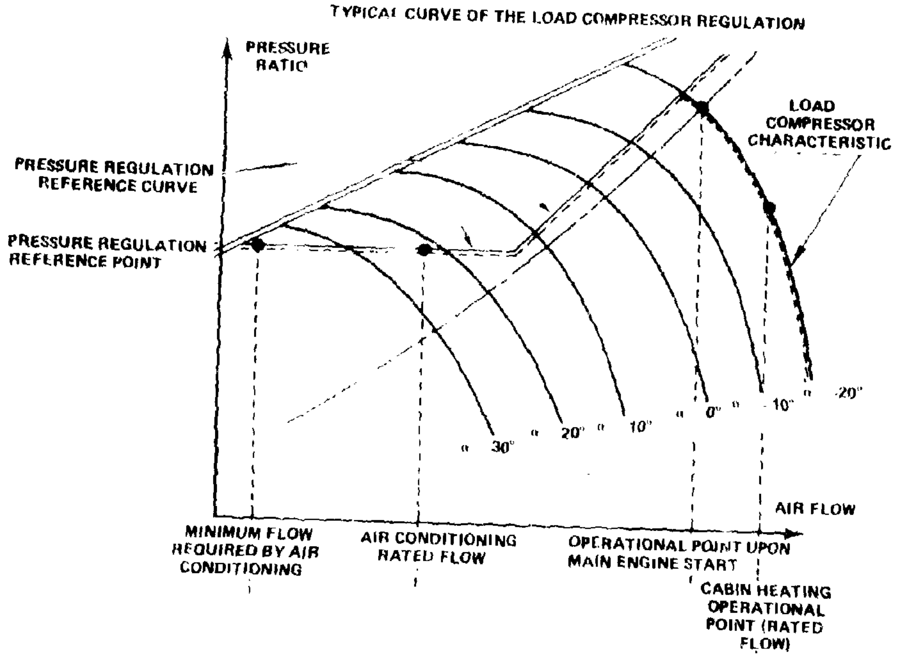


Figure 10

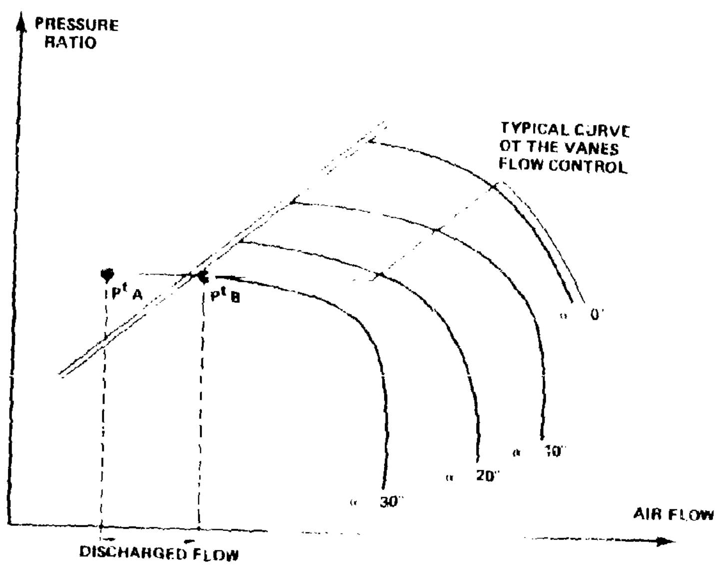


Figure 11

DISCUSSION

C.Rodgers, US

How rapidly can you close the load compressor discharge valve without causing surge?

Author's Reply

The load compressor outlet is connected with two valves located in parallel disposition: a bleed valve and a discharge valve. The discharge valve isnormally closed and opens only when the bleed valve closes or if the bleed flow decreases too much. In these cases, the APU digital control opens the discharge valve befo[illegible] the bleed valve will be completely closed or before the compressor wheel goes across the surge line. When the [illegible] charge valve closes, the bleed valve has begun to open before. This operation sequency does no[illegible] allow any APU surge due to bleed or discharge valve closing. Minimum surge closing time is 3 seconds.

SECONDARY POWER SUPPLIES FOR A SMALL SINGLE ENGINED COMBAT AIRCRAFT

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SUMMARY

This paper describes a possible secondary power supply system for a small single-engined combat aircraft. It describes the way in which reliability, survivability and installation problems influence the type of system which is proposed for such an aircraft. The particular constraints imposed by the use of a Pegasus vectored thrust engine in a Short Take Off Vertical Landing (STOVL) aircraft are also discussed.

The proposed system consists of a conventional engine-driven accessory gearbox to provide hydraulic and electrical power in flight. A remotely mounted APU is used to start the engine by means of a pneumatic link from a load compressor. The APU drives a separate standby hydraulic pump and generator for ground operation and emergency power in flight.

Possible areas of future research and development which would lead to improved efficiency and reduced weight are also discussed.

INTRODUCTION

Over the past few years British Aerospace Brough have been involved in project studies on several small single engined combat aircraft. Some of these have been 'conventional' aircraft but the majority of the studies have been of aircraft with vectored thrust to give a short take off and vertical landing capability. Some of the work has been done in collaboration with other British Aerospace sites such as Kingston and Warton.

All aircraft designs have to be a compromise to meet the conflicting requirements of survivability, reliability and maintainability. In addition, there is a need to minimise weight and cost, both initial purchase cost and subsequent operating costs. A STOVL fighter/attack aircraft probably presents one of the most severe challenges in all these respects. It must not only meet the performance requirements but it must also be capable of operating from dispersed sites in relatively primitive conditions.

PRINCIPAL SYSTEM REQUIREMENTS

The auxiliary power system must provide hydraulic power, electrical power and bleed air to the airframe services, both in flight and on the ground. It must also be capable of starting the main engine and if necessary assisting it to relight in flight.

There has been a steady growth in the size of auxiliary power requirements as aircraft have become more sophisticated. The development of more manoeuvrable aircraft fitted with 'fly by wire' flight control systems has led to increased demands on the hydraulic system. The required flowrates have increased because of the larger number of actuators and faster rates of movement demanded by such systems. As an example, the Buccaneer, which first flew in 1958 has an installed hydraulic system power of about 65KW. Broadly similar aircraft of more recent design have typically got about twice the installed hydraulic system power.

Electrical power demands have also increased as aircraft have become more sophisticated and the amount of essential avionic equipment has increased. It is interesting to note that with each generation of avionic equipment great claims are made for reduced power demands and smaller sizes. However, the smaller size of equipment almost invariably leads to more components being installed in the available space so overall the power demands have increased steadily. This trend is illustrated by the development of the Harrier family of aircraft. This started life with a pair of 4KVA alternators, graduated to a single 12KVA integrated drive generator (IDG) and now in its latest form as the GR 5 has two 15 KVA IDG.

The integrity of electrical power supplies must also be improved since systems such as fly by wire are flight critical and cannot tolerate interruptions to their power supplies. It is quite feasible to fly a Buccaneer for over an hour using its single lead acid battery as the sole source of electrical power. However, on some more recent aircraft the endurance available on the battery can be less than 10 minutes.

The auxiliary power system must also be capable of being used to power systems for ground checkout and servicing. There is increased emphasis nowadays on dispersed site operation to protect the aircraft fleet. This is particularly true for STOVL aircraft which gain much of their operational flexibility from this type of utilisation. Power is also required while aircraft remain on alert, possibly with crew manning the aircraft for quite long periods of time. During such alerts the crew would need to wear suitable clothing to protect against nuclear, chemical or biological weapons. Consequently, there is a need to supply some form of air conditioning to make their life tolerable. Some aircraft operators have specified quite long periods of aircraft readiness, one requirement was for 8 hours in 4 two hour blocks with a crew change at the end of each block.

HYDRAULIC AND ELECTRICAL POWER SUPPLIES

Reliability studies show that the primary hydraulic power supplies need to be duplicated. Two independent systems are required, each powered from an engine driven pump. For the size and configuration of aircraft studied with a typical empty weight around 7000-9000 kg, each hydraulic pump needs to be rated at about 1.7 litres/sec at a nominal delivery pressure of 27600 KPa. Some research has been done on higher pressure systems, up to 55200 KPa, in the USA. However, there does not yet seem to be a clear case for such a large increase in system operating pressures.

Some novel hydraulic pump designs have been tried on recent aircraft. However, the conventional axial, multi-piston pump still appears to offer the most reliable source of hydraulic power. Relatively conventional rotational speeds would be proposed, i.e. a maximum of about 6000 rpm for a pump of the size chosen. Hydraulic accumulators will be used in each system for 'peak lopping' to minimise the required pump flow.

The reliability of the electrical power generation channel is usually better than the hydraulic system and the primary demands can be met by a single generator rated at 40/60 KVA. A typical breakdown in the load is given in the table below.

SYSTEM	TYPICAL MAX. LOAD KVA
Radar/Weapons	9.4
Electronic Warfare	5.0
Heaters/De-icing	5.5
Navigation/Communication	1.5
Flight/Engine Controls	1.7
Computing/Displays	2.0
Miscellaneous Systems	5.5
Reserve for Growth	15.0
TOTAL	59.0

It can be seen from this table that there is no provision for electrically driven fuel booster pumps. The BAe Brough philosophy for aircraft fuel systems is based on the use of a hydraulically driven fuel flow proportioner for engine fuel feed systems and air pressurisation for intertank transfer. The main advantage of this system is that its input power requirements are more demand conscious than electrically driven booster pumps. There is, therefore, less of a power penalty incurred by the need to supply the maximum reheated engine fuel flow.

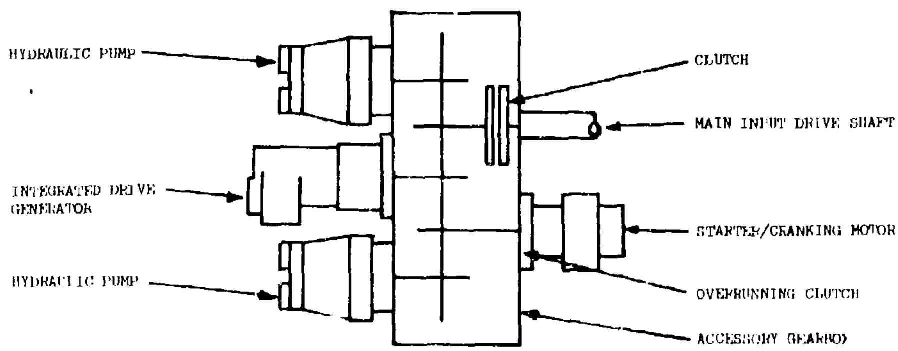
There is a significant electrical load required to safely recover the aircraft after a main generator failure. Provisional estimates suggest that this adds up to about 5.5 KVA. The standby load required while the aircraft is on alert is about the same order. The absolute minimum flight critical loads are estimated to be about 1500 watts to keep the FCS, engine and basic instruments powered.

Various types of AC generator have been considered. When relatively small power levels are required, i.e. below 30 KVA, it appears that Variable Speed Constant Frequency (VSCF) systems are beginning to be competitive. However, in the mid range the most appropriate system seems to be an integrated drive generator using an axial gear differential constant speed drive. Very high speed (25,000 rpm) systems such as the Lucas Compact Constant Frequency Generator have yet to be proven at this sort of power supplies level but look promising for high power systems, i.e. 60 KVA and above.

Some studies have been carried out into the possibility of an 'all electric aircraft' in which hydraulic systems could be replaced by high power electric motors. However, we are yet to be convinced that this is a feasible option for a small military aircraft.

STANDBY AND EMERGENCY POWER SUPPLIES

Power is required on the ground both for checkout of the aircraft systems and for powering essential equipment while the aircraft is on quick reaction alert or operating in the 'cab-rank' role. In order to check the main sources of power it is necessary to be able to drive the main accessory gearbox from the APU. Although it is unlikely that this power will be needed for very long, or very often it is undesirable to drive the engine as well as the gearbox. However, for engine starting the main power take off shaft must be connected to the accessory gearbox if the same source of power is to be used. Various methods have been used to overcome the problem including overrunning clutches, torque converters and friction clutches. However, all these methods add to the complication and weight of the system. The proposed solution is a simple mechanical clutch on the power take off shaft from the engine to isolate it when the gearbox, hydraulic pumps or main generator need to be checked. The system schematically then looks like that shown in Figure 1.



When sitting on alert for long periods of time the main accessory gearbox would represent an unnecessarily large load for the relatively little amount of hydraulic and electrical power needed. The proposed solution is to have a small hydraulic pump and AC generator driven directly by the APU. The hydraulic pump sized at about 0.4 litres per second would provide sufficient power to check out the majority of the hydraulic system. It would also drive the fuel flow proportioner to recirculate fuel for use as a heat sink. The generator sized at about 8 KVA would provide sufficient electrical power for equipment needed on the ground.

Several means of providing emergency power in flight have been considered. Ram air turbines are relatively simple but are of doubtful performance at extreme attitudes or very low airspeeds which are possible with modern manoeuvrable aircraft. Emergency power units operating on hydrazine or isopropyl nitrate have also been considered. However, these systems do present a logistics problem since additional consumable supplies have to be provided at each operating base. In addition, the fuels used are relatively hazardous.

The Pegasus engine has quite a large high pressure core and windmills at a reasonable speed under most flight conditions. In the event of a flameout it will supply adequate hydraulic power to keep the aircraft in a stable flight path. A dedicated NiCad battery is proposed for the FCS and engine control systems and so the immediate consequences of an engine flameout can be catered for. In the longer term the APU can be started and its dedicated hydraulic pump and generator can be used to recover the aircraft to base.

In the event of a catastrophic engine failure the aircraft, being single engined, will probably be lost. The hydraulic system accumulators will provide a few seconds of power to enable the aircraft to be manoeuvred into a safe attitude for ejection.

BLEED AIR SUPPLIES

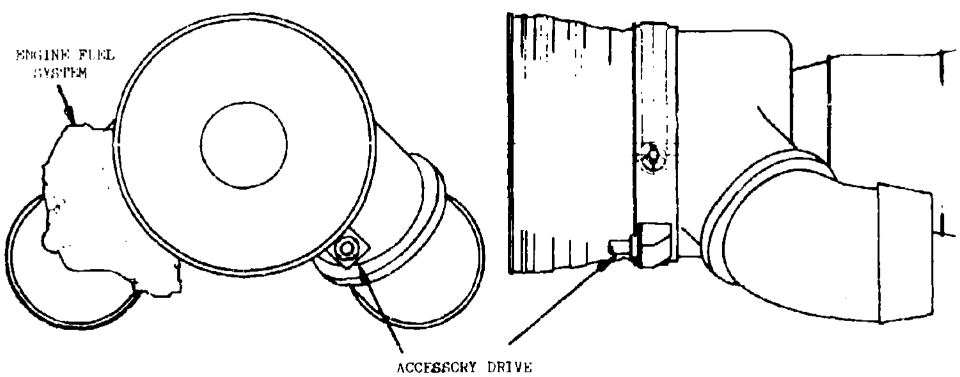
In all the STOVL aircraft studied, engine bleed air has been assumed to be used for reaction controls in hovering flight and during the transition to wingborne flight. As a rough guide the maximum amount of bleed air required is about 1 kg/sec for every 1000 kg of aircraft weight. The only practical source of such quantities of air is the main engine since APU air is generally at too low a pressure and only available in small quantities. However, since the APU must be used to provide bleed air to operate equipment cooling and aircrew environmental control systems on the ground there is a case for it continuing to be used during take off and landing to minimise main engine bleed with its consequent thrust loss. Typically, each kg/sec bleed from the main engine costs about 175 kg of engine thrust for a given TGT so any saving is well worthwhile.

ENGINE INSTALLATION ASPECTS

All the STOVL aircraft studied at BAe Brough have used engines which were derived from the Rolls Royce Pegasus vectored thrust turbofan. The Pegasus engine is a major constraint on the installation and layout of the secondary power system. It large frontal area imposes a severe aerodynamic penalty, particularly on a supersonic aircraft. The Harrier has all its engine driven accessories mounted on top of the engine but this leads to an unacceptable increase in frontal area on a supersonic design.

In efforts to get around these installation problems several means of driving a remotely mounted set of accessories have been studied. These have included using a hydraulic drive, pneumatic drive, mechanical drive shafts and dedicated auxiliary power units. However, all these methods have been found to be significantly less efficient than a conventional close coupled gearbox with a direct shaft drive. In addition, they impose reliability and vulnerability problems which lead to weight increases when attempts are made to overcome them.

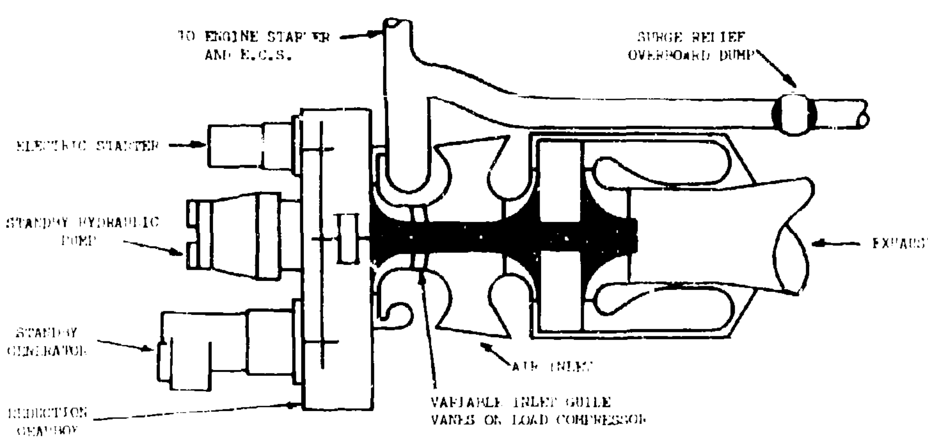
Although modern electronic controls have reduced the need for large numbers of engine mounted hydromechanical components there is still a substantial volume occupied by this equipment. On most of the proposed advanced versions of the Pegasus engine the major parts of the fuel system are collected together in the area forward of the starboard front nozzle to allow them to be faired within its profile. The area forward of the port front nozzle is therefore a convenient location for the engine driven accessories as shown in Figure 2.



This location for the accessories is conveniently situated to use a radial drive from the front of the engine HP compressor. However, there is not sufficient space to fit the APU as well as the main gearbox. In any event since the APU is to drive the emergency hydraulic pump and generator there is a good case for mounting it away from the main power supply.

AUXILIARY POWER UNIT INSTALLATION

The most flexible means of linking the APU and the main accessory gearbox is by using a pneumatic drive from an APU driven load compressor. This makes the system mechanically simple since it obviates the need for clutches or torque converters in the main accessory gearbox. An air turbine on the gearbox will give a smooth increase in load on the APU as it starts the main gearbox and engine. The APU load compressor can be isolated by means of variable inlet guide vanes and the standby hydraulic pump and generator can also be offloaded to minimise APU starting torque. A schematic view of the APU is shown in Figure 5.

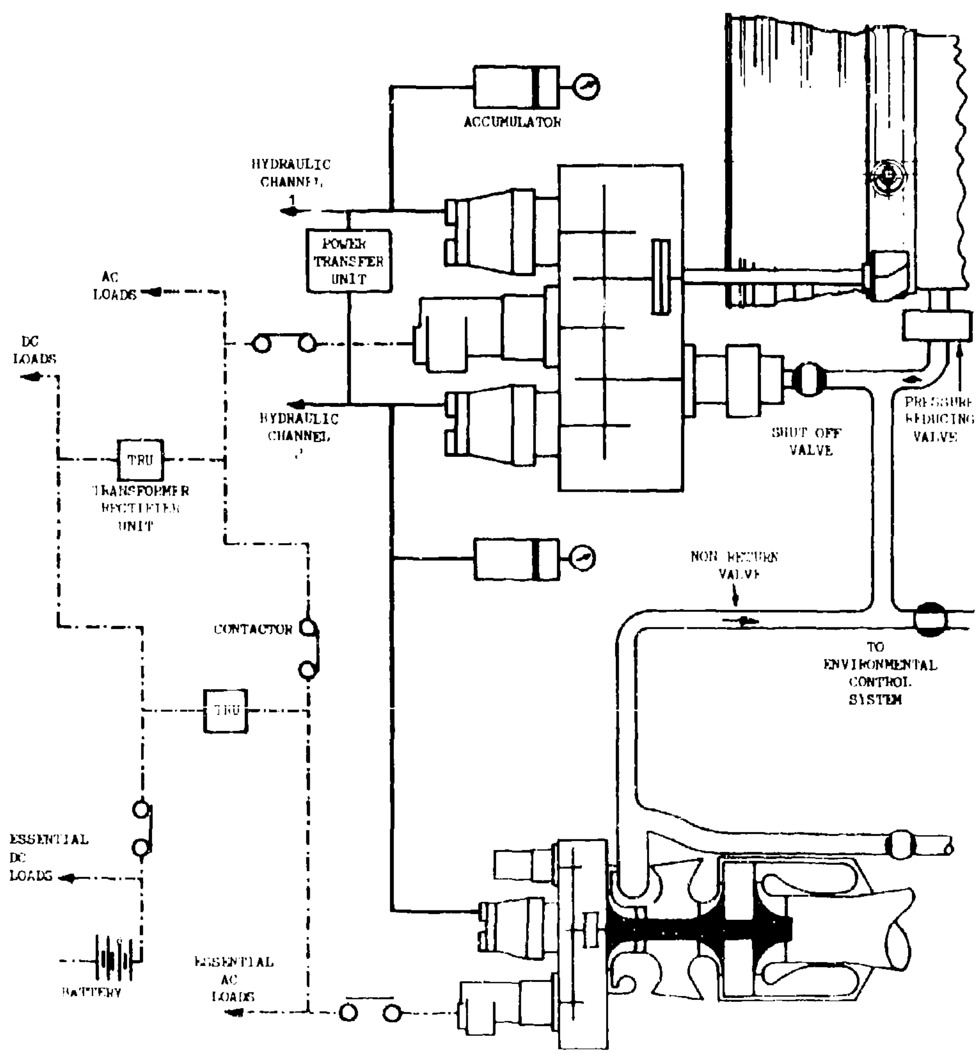


The APU needs to be capable of rapid starting in an emergency and also be capable of operating at high altitude. The preferred means of APU starting is by using an electric motor supplied from the general services battery. Alternative means of rapid engine starting have been considered and the most promising solution seems to be to use a cartridge starter. Some preliminary assessments of this system have been carried out by APU manufacturers. No insuperable problems have been revealed but it does have the disadvantage that cartridges would have to be provisioned at all operating bases.

There has been much debate about the need to operate the APU at high altitude in the event of an engine failure. In BAe Brough's experience the majority of high altitude engine incidents arise either from surges during reheat lighting or excessive intake distortion while manoeuvreing. It can be argued that modern flight control systems and engine control systems will prevent such incidents occurring on the next generation of aircraft. The Pegasus engine will windmill fast enough to provide hydraulic power after a flameout and should permit a rapid descent to a low enough altitude to relight. However, it should be possible to start the APU as a precautionary measure at times of high risk such as areas of the flight envelope where the engine has a low surge margin. Thus, although it would be an advantage if the APU could operate at high altitude it need not necessarily be able to start at such heights.

OVERALL SECONDARY POWER SYSTEM

From the discussions in the previous sections of this note the system which has evolved is as shown in Figure 4.

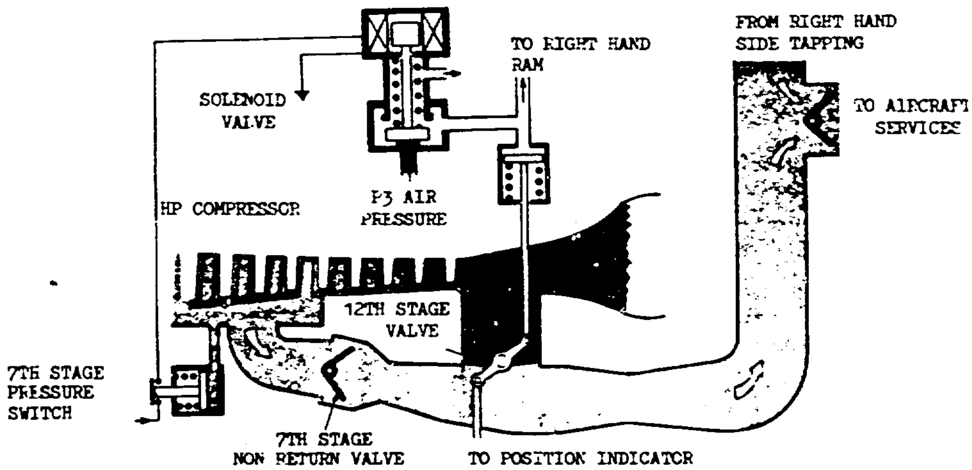


A common ducting system is used between the engine, APU and environmental control system. This allows the engine to drive the main accessory gearbox by means of compressor bleed air as a back up to the shaft power take off. The Pegasus engine does not have any particularly severe limits on shaft power off-take. However, on RB199 engines a compressor bleed air back up system can provide a useful supplement to the shaft power available while at the same time improving the surge margin of the engine.

AREAS FOR FUTURE RESEARCH AND DEVELOPMENT

As secondary power systems have grown in size they have begun to have a significant effect on engine and airframe performance. Bleed air offtakes and shaft power demands have become particularly important with the advent of high bypass ratio turbo fan engines. The supply of even relatively small amounts of engine bleed air brings a number of problems for the airframe systems designer. Typical environmental control systems require an air inlet pressure between 400 and 600 KPa. However, in order to achieve these pressures at low engine speeds the air must be taken from a high pressure compressor stage. At the extremes of the aircraft's flight envelope this may deliver air at over 2200 KPa and at temperatures approaching 800°K. As a result, precoolers and pressure reducing valves have to be used to dump much of the energy which has been very expensively put into the air by the engine.

One approach to this problem has been adopted on the Spey engines used in the RAF Phantom aircraft. Air can be bled from either the 7th or 12th stage of the HP compressor, depending on the delivery pressure of the engine. The principle of the system is illustrated in Figure 5.



After some teething troubles this system has been found to work well in practice even though the changeover between stages operates as a 'bang-bang' system. The amount of bleed air demanded from engines can also be minimised by rejecting as much heat as possible to the engine fuel supply. This will often entail complications such as recirculation back to the aircraft tanks to control temperature levels with varying heat loads and fuel flow rates. The ability to transfer fuel around can in any event be a useful aid in improving survivability after combat damage or system failures.

Such systems are only acceptable provided that they do not degrade the aircraft reliability or increase pilot workload. The advent of modern microprocessors is making it possible to provide the intelligence to cope with the control of these advanced systems. In addition the use of a data bus such as MIL STD 1553B is making it possible to transmit all the necessary sensor and command information without the weight penalty of vast amounts of wiring. With modern engine electronic control systems a much closer integration with the airframe is possible. Data on airframe power and bleed offtakes can be supplied to the controller and compared with what offtakes are available from the engine. The most efficient combination of power, bleed and engine handling bleeds can then be scheduled throughout the flight envelope, hopefully removing many of the restrictions which exist on some of todays engines.

Modern auxiliary power units are becoming smaller and more efficient. However, there are still very few available which will operate at high altitude. There seems to be a growing need for such units as aircraft power demands are increasing and their integrity is becoming more critical. One possible means of improving high altitude performance is to 'supercharge' the APU by utilising a small amount of engine bleed air. This presupposes that the APU will be started either as a precautionary measure under conditions of high risk or can be started rapidly as soon as an engine failure is detected.

Hydraulic pumps are likely to become more sophisticated even if pressure levels do not increase very much beyond todays typical 27600 KPa. Dual pressure level pumps are now quite common with a lower datum pressure for offloading the accessory gearbox during engine starting. The next stage would seem to be viable delivery pressures as well as the almost universal variable flowrate pumps. By running at a lower system pressure during cruise conditions, engine shaft power demands can be minimised at low engine speeds. System internal leak rates will be lower so less heat will be rejected within the hydraulic system. Again reliable microprocessor control will be needed to make such systems feasible.

In the field of electrical power generation the development of variable speed, constant frequency generators promises improvements in reliability over the mechanical complexities of constant speed drives However, there seems to be some way to go to get the electronics sufficiently reliable and able to withstand the temperatures associated with operation in close proximity to the engine. There is still a requirement for a reliable battery, even after all the years experience with the available lead-acid and nickel-cadmium types. A reliable charging system and state of charge indicator would go a long way towards giving more confidence in the use of NiCad batteries for emergency power supplies.

CONCLUSIONS

There is still scope for considerable improvement in the overall efficiency of aircraft auxiliary power supplies. Perhaps the greatest improvements would be achieved by a closer co-operation between the various airframe systems designers and the engine designers at an early stage in the development of the project. It is apparent that the present situation where the project aerodynamicists seem to have first say in what engine development is necessary is not always the best approach. It is essential that the secondary power system, engine bleed air offtakes and engine design must be considered as a whole rather than developed in isolation.

GENERATION HYDRAULIQUE DE SECOURS DU MIRAGE 2000

par
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INTRODUCTION

La présente communication a pour objet la description et la justification du système de puissance hydraulique secours du MIRAGE 2000. Ce système de conception originale répond en fait à un problème tout à fait particulier et spécifique de l'avion :

En effet, il s'agissait de trouver une source d'énergie d'une part et un moyen de la mettre en oeuvre d'autre part, susceptibles d'assurer le fonctionnement des servo-commandes hydrauliques de commande de vol de l'avion dans les conditions particulières suivantes :

(a) Moteur éteint

(b) Dans une phase de vol très particulière que nous appelons la "cloche" (TAIL SLIDE MANEUVER).

1. PHASE DE VOL CONSIDEREE

Cette "cloche" est une figure acrobatique ou de combat composée des phases suivantes :

(a) Une phase de prise d'assiette au cours de laquelle la vitesse aérodynamique diminue en même temps que l'altitude croît. Pendant cette phase les gouvernes aérodynamiques sont efficaces et assurent le contrôle de l'avion.

(b) Une phase au cours de laquelle l'avion suit sa trajectoire "par inertie", pendant laquelle la vitesse aérodynamique diminue et, si l'assiette initiale prise pendant la phase (a) est suffisamment grande, passe même par zéro. Pendant cette phase, et bien que les gouvernes aérodynamiques soient inefficaces pendant un court laps de temps, les commandes de vol électriques restent toujours actives et positionnent les gouvernes de façon à réagir à la situation pour que l'avion soit contrôlable dès que la vitesse reprend quelque valeur.

Au cours de cette phase à vitesse quasiment nulle, l'avion "bascule" vers l'avant du fait de son hyperstabilité longitudinale à très forte incidence.

(c) Une phase de reprise de vitesse aérodynamique au cours de laquelle les gouvernes redeviennent efficaces et de ce fait recentrent l'avion sous l'effet des commandes électriques, ce qui évite tout engagement en vrille incontrôlé et permet à l'avion de suivre la ligne de vol choisie par le pilote, même si les ordres du pilote ont été donnés pendant la phase (b) avant la reprise de vitesse.

Il faut préciser que cette stabilisation de l'avion a lieu immédiatement après le basculement (nose down) et à une vitesse encore très faible.

Ce dernier point est très intéressant : même si le MIRAGE 2000 n'est pas le premier avion à entreprendre des manoeuvres balistiques impliquant un passage à vitesse nulle, il est néanmoins capable de retrouver un haut niveau de manoeuvrabilité immédiatement après, sans que le pilote ait besoin de "tout mettre au milieu et d'attendre, en s'arrêtant de respirer, une vitesse aérodynamique suffisante". On peut dire que les commandes de vol restent efficaces à des vitesses de l'ordre de 40 Kts. Au-dessous, le travail est principalement dû aux saines qualités aérodynamiques de l'avion, en particulier son hyperstabilité en tangage à une incidence supérieure à 30°

2. LE MIRAGE 2000 ET SES COMMANDES DE VOL ELECTRIQUES

La figure qui vient d'être décrite est, certes, réalisable sur avion à commandes de vol classique, mais elle conduirait le plus souvent à des situations difficiles à contrôler sans une habileté exceptionnelle du pilote :

En effet, si la première phase de prise d'assiette est toujours possible, il arrive un moment où l'avion échappe au contrôle du pilote ; on ne sait alors prédire ce qui va se passer : A partir de ce moment ou bien la vitesse s'accroît à nouveau et l'avion redevient progressivement contrôlable, mais alors la figure est ratée, l'avion ne s'est pas arrêté en l'air, ou bien l'avion entre en vrille, configuration de vol plus ou moins stable d'où il ne sortira qu'en prenant à nouveau de la vitesse après des manoeuvres bien déterminées.

C'est grace aux commandes de vol électriques que dans le MIRAGE 2000 la cloche est réalisable en toute sécurité et sans habileté exceptionnelle du pilote :

Le but des commandes de vol électriques est de rendre l'avion contrôlable et pilotable en toute sécurité quels que soient les incidences, les charges ou les cas d'instabilité rencontrés. Elles permettent en particulier le pilotage avec une stabilité longitudinale statique négative ou voisine de zéro. Dans le cas de la "cloche", cette caractéristique facilite d'ailleurs le lancement de l'avion vers le haut dans la première phase (a) de la figure.

De plus, les limitations d'incidence et de facteurs de charge automatiquement introduites dans le calculateur de commande de vol permettent au pilote toutes les manoeuvres sans risque de dépasser ces limites. En particulier c'est ce calculateur qui pendant la cloche braque à plein piqué les élevons sans que le pilote intervienne. Par contre celui-ci pourra, pendant ce temps là, pré-positionner son manche de façon qu'à la sortie l'avion suive la trajectoire désirée et sans qu'il y ait à aucun moment perte de contrôle.

Bien sûr ces mouvements de gouvernes demandent une certaine puissance et c'est ce qui nous amène au sujet essentiel.

3. SOURCES D'ENERGIE POUR LES COMMANDES DE VOL

L'énergie nécessaire au fonctionnement des commandes de vol est normalement fournie par le moteur sous forme électrique et hydraulique. Dans le cas du MIRAGE 2000, le moteur entraîne deux alternateurs à vitesse constante du type AUXIVAR de 20 KVA et deux pompes hydrauliques autorégulatrices débitant 112 l/mn sous 4000 PSi à 6100 t/mn, ce qui correspond à un régime moteur de 10 600 t/mn. L'énergie hydraulique fournie par les pompes entraînées par le moteur est très largement suffisante non seulement pour alimenter les servo-commandes et les servitudes (train, hypersustentateurs, etc...) mais aussi pour entraîner des alternateurs destinés à fournir de l'électricité aux commandes de vol, indépendamment de celle fournie par les alternateurs principaux entraînés par moteur. C'est pourquoi, même avec le moteur éteint, à condition que la vitesse de l'avion soit suffisante pour entraîner les pompes hydrauliques par l'intermédiaire du moteur en moulinet, le fonctionnement des commandes de vol reste assuré aussi bien hydrauliquement qu'électriquement. A titre indicatif le régime du moteur en moulinet est encore de 2 380 t/mn pour une vitesse indiquée de 200 Kts environ, ce qui donne un débit hydraulique disponible total de 50 l/mn pour l'avion.

Par contre, que va-t-il se passer si le pilote de l'avion exécute une figure de vol du type de celle que nous avons décrite précédemment, au cours de laquelle la vitesse avion passe par des valeurs très basses, voire nulles ? Si le moteur reste allumé, il n'y a pas de problème puisque le régime de ralenti vol reste encore supérieur à 50 % du régime maximal, ce qui donne un débit hydraulique de 60 l/mn disponible sur chaque pompe.

Il n'en est pas de même si une extinction du moteur survient avant ou pendant cette figure. En effet cette extinction pourra conduire à une vitesse de rotation moteur en moulinet très basse voire nulle, en tout cas très insuffisante pour assurer un débit convenable des pompes hydrauliques entraînées par le moteur. Il faudra donc dans ce cas trouver une autre source d'énergie pour assurer le fonctionnement des commandes de vol qui comme on l'a vu précédemment sont indispensables au bon positionnement des gouvernes pendant cette phase, en attendant que la vitesse retrouvée puisse permettre un moulinet suffisant du moteur et éventuellement son rallumage.

4. CHOIX DE LA SOURCE D'ENERGIE DE SECOURS

Plusieurs formes d'énergie de secours peuvent être envisagées : le moulinet moteur est [illegible]é par hypothèse, de même que les procédés utilisant la vitesse aérodynamique comme [illegible] aéropompes, systèmes "ram-air" et dérivés. Il ne nous reste que l'énergie que l'on peut stocker à bord sous forme chimique : carburants ou batteries.

Un APU utilisant le pétrole de l'avion aurait l'avantage de ne pas nécessiter l'emport d'un carburant spécial. Il a par contre l'inconvénient de nécessiter une installation d'entrée d'air pour le comburant et il a surtout l'inconvénient majeur de son temps de démarrage et de mise en oeuvre. Avant d'atteindre son régime nominal, il lui faudra plusieurs dizaines de secondes ce qui est incompatible avec le problème. Nous verrons plus loin que la mise en oeuvre de ce système doit intervenir en un temps ne dépassant pas 2 secondes. De plus il y a toujours une limite supérieure à l'altitude de rallumage de ce genre de machine qui interdirait la manoeuvre au-dessus de cette altitude. On ne voit pas moyen de surmonter ces difficultés, à moins de laisser cet APU en fonctionnement permanent depuis le sol. Mais alors ceci nécessiterait des précautions d'installation et une endurance de la machine très injustifiées vis-à-vis de la rareté de l'évènement que l'on cherche à couvrir.

Un E.P.U. à hydrazine aurait pu être envisagé : la faible puissance mise en jeu et la faible durée de fonctionnement justifieraient l'usage d'une faible quantité de carburant et de comburant, mais là encore on se heurte, quoique dans une moindre mesure que dans le cas de l'APU, au problème de la rapidité de mise en oeuvre du système.

L'idée la plus simple qui viendrait à l'esprit serait d'utiliser une source d'énergie qui existe de toute façon à l'intérieur de l'avion à savoir la batterie électrique. En effet, dans le MIRAGE 2000, cette batterie alimente une pompe hydraulique de secours de 8 l/mn à fonctionnement discontinu. Cette pompe qui existe sur tous les avions du type MIRAGE a pour but de permettre en cas de blocage réacteur et donc de suppression de toute possibilité de prélèvement d'énergie sur le moteur, d'alimenter les servo-commandes de vol de façon à permettre au pilote de se mettre en position favorable à l'éjection, qui est la consigne normale dans ce cas extrême de panne.

Si on utilisait cette pompe de secours pour alimenter les servo-commandes dans le cas qui nous occupe, il n'est d'abord pas certain que son débit serait capable d'assurer les mouvements de gouverne nécessaires. En effet, mettre un avion en position favorable à l'éjection n'est pas la même chose que de le manoeuvrer de façon à exécuter une sortie de cloche. De plus l'utilisation intensive de cette pompe pendant le temps nécessaire risquerait de vider substantiellement la batterie de l'avion, ce qui la rendrait inapte par la suite à remplir son rôle de source électrique de secours, rôle qui consiste à assurer un temps d'autonomie électrique à partir du moment où les sources d'énergie électrique normales se sont trouvées indisponibles. En particulier cette batterie risquerait d'être indisponible pour le rallumage du moteur par la suite.

En résumé, cette source d'énergie doit donc être conservée précieusement et ne peut donc être utilisée pour l'usage qui nous occupe.

5. SOLUTION CHOISIE

Il a fallu donc créer un nouveau système de secours indépendant et disponible immédiatement.

Ce système consiste en une pompe hydraulique branchée en parallèle sur un des circuits d'alimentation hydraulique des servo-commandes (circuit N° 2). Cette pompe est entraînée par un moteur électrique alimenté lui-même par une pile amorçable dont l'énergie devient immédiatement disponible pour un temps limité dès sa mise en route. Cette pompe se substitue au système normal de mise en pression du circuit hydraulique N° 2 dont elle assure tous les services tant qu'une tension électrique suffisante est maintenue aux bornes du moteur.

La pompe proprement dite est une pompe délivrant 12 l/mn sous 230 bars (rappelons que la pression normale des circuits hydrauliques est 280 bars). Elle est autorégulatrice c'est-à-dire qu'elle comporte un plateau oscillant à 7 pistons dont l'inclinaison est commandée par un régulateur maintenant la pression de sortie constante et égale à 230 bars tant que le débit demandé n'est pas supérieur à la valeur nominale. De la sorte, le déplacement par tour est variable et atteint 1 cm3 par tour environ pour l'inclinaison maximale du plateau. La vitesse de rotation varie entre 10 000 et 14 000 t/mn suivant la tension électrique d'alimentation.

Le moteur électrique est du type compound ce qui lui permet d'allier un couple au démarrage élevé (indispensable pour vaincre le couple résistant de la pompe) avec des caractéristiques couple/vitesse acceptables lorsque la tension baisse.

La pile thermique est constituée par un ensemble de plusieurs paquets de 13 éléments série branchés en parallèle. Ces éléments sont constitués d'une électrode négative en métal alcalin pur et d'une électrode positive en oxyde fort (Ca Cr O4) noyées dans un électrolyte solide constitué par un mélange eutectique de chlorure de lithium et de potassium. Ces éléments ne sont donc pas actifs tant que l'électrolyte reste à l'état solide. L'initiation de la pile consiste en la mise à feu d'une composition chauffante qui amène l'électrolyte au-delà de sa température de fusion de 360°C, le rendant ainsi conducteur ionique actif. La puissance maximale est disponible au bout d'une seconde seulement, temps à partir duquel la tension aux bornes atteint 30 Volts. A partir de ce moment la tension baisse progressivement, cette baisse dépendant de la consommation. Pour la consommation initiale importante de l'électropompe (de l'ordre de 250 A), la chute de tension est rapide. Au bout de 20 secondes, la tension aux bornes n'est plus que 25 V. Ensuite la consommation diminuant, la chute est moins rapide : tension environ 20 V au bout de 50 secondes.

Ce type de pile est utilisé sur engin balistique mais en général ces piles sont à injection d'électrolyte liquide et ne fonctionnent qu'entre 10 et 30°C ce qui n'a en général pas d'importance, vu la température des silos dans lesquels ces engins sont stockés. Au contraire sur le MIRAGE 2000, il a été nécessaire de choisir une pile capable de fonctionner à basse température initiale (- 30°C au plus) d'où le choix de l'électrolyte solide.

Cette pile se présente sous l'aspect d'un boitier métallique cylindrique hermétique. Le mélange chauffant de la pile est mis à feu par deux étoupilles pour assurer la redondance. Ces étoupilles sont équipées d'un shunt de protection pour le transport.

Le système de commande d'amorçage est automatique : une tension continue de 27 V est envoyée aux bornes des étoupilles lorsque toutes les conditions suivantes sont réalisées :

(1) Vitesse de rotation du moteur inférieure à 15 % du régime maximal.

(2) Vitesse air de l'avion inférieure à 100 Kts (50 m/s).

(3) Altitude avion supérieure à 10 000 ft et inférieure à 70 000 ft.

(4) Train non sorti détecté par un contacteur triple.

Pour éviter des risques de fonctionnement intempestif au sol en maintenance l'avion étant à l'arrêt, une condition N > No (environ 5%) a été ajoutée en plus des quatre conditions ci-dessus.

La logique de commande est entièrement doublée, chaque logique attaquant chacune des deux étoupilles d'amorçage.

Enfin, comme il aurait été regrettable de ne pas pouvoir utiliser un système aussi simple pour ramener l'avion avec réacteur éteint qui se trouverait par hasard assez près d'un terrain pour se contenter d'un fonctionnement aussi court, on s'est donné la possibilité d'une commande manuelle par poussoir sous opercule qui par son action simule toutes les conditions du fonctionnement automatique.

Un témoin d'amorçage indique que le système est en fonctionnement.

6. JUSTIFICATION

Le choix d'un système aussi rustique ne se justifie qu'à cause de deux éléments importants, tous deux dûment vérifiés par l'expérience :

(1) La faible durée pendant laquelle on risque d'avoir besoin d'une telle source d'énergie.

(2) La faible probabilité de l'occurrence d'une situation nécessitant son emploi.

En effet, le système décrit ci-dessus a deux caractéristiques : sa faible durée de fonctionnement et le fait qu'il ne peut servir qu'une fois, la batterie thermique devant être renouvelée après chaque percussion.

(A) En ce qui concerne la faible durée de fonctionnement, le chiffre de 1 minute environ donné plus haut résulte du calcul le plus défavorable de la durée nécessaire pour retrouver un régime suffisant en moulinet moteur. Pour ce calcul, on suppose que l'extinction moteur est survenue au moment le plus défavorable de la cloche, c'est-à-dire au temps t1 tel que la durée pendant laquelle le régime moteur va rester inférieur à 2 500 t/mn environ est maximale. Ce temps correspond à une vitesse avion de l'ordre de 180 Kts. Et c'est bien l'hypothèse la plus défavorable :

Avant, le pilote, prévenu de l'extinction moteur, a encore une vitesse suffisante pour interrompre la figure et amorcer la manoeuvre de rallumage. Après, le temps qui reste à courir à vitesse faible devient de plus en plus court et finit par être inférieur au temps que met le moteur à "débobiner" jusqu'au régime de moulinet correspondant à Vi = 100 Kts (25 % du régime maxi.), auquel cas il n'est plus besoin de source d'énergie autre que le moteur.

Des essais au sol ont été conduits en simulant la décroissance du débit fourni par la pompe principale à partir de l'extinction moteur grace à un groupe hydraulique programmé, tout en demandant le débit instantané maximal correspondant aux valeurs mesurées en vol dans la manoeuvre la plus défavorable ("contre" des élevons arrivant au moment où le moteur a entièrement débobiné). Ces essais ont montré que le pilotage en électrique restait possible jusqu'à la fin de la manoeuvre et que la capacité de la pile, de l'ordre de 1 minute à pleine puissance, plus un excédent à puissance réduite pouvant aller jusqu'à 2 minutes et plus, couvrait largement cette exigence.

(B) En ce qui concerne la probabilité d'occurrence de la situation nécessitant la mise en route d'un tel système, notre expérience a montré qu'elle était très faible.

De nombreux essais en vol ont montré que l'extinction du moteur pendant les cloches était très improbable, et de toute façon pas plus probable que dans n'importe quelle autre phase de vol.

Durant les essais en vol, plus de 500 cloches ont été exécutées dont 100 dans toutes les configurations moteur possibles (tous régimes possibles, PC maxi, réduite ou éteinte, mouvements brusques de manette de gaz et même simulation de passage en secours du régulateur (FCU) moteur etc...) sans qu'il y ait eu ni extinction ni même autre anomalie moteur.

Cette bonne fiabilité du moteur SNECMA M 53 est en fin de compte la meilleure justification du choix du système de génération de secours du MIRAGE 2000 puisqu'on est assuré ainsi de n'avoir à s'en servir que tout à fait exceptionnellement.

On peut même dire, et ce sera la conclusion, que si les essais moteurs, qui ne sont pas terminés, continuent à être aussi bons, il est possible que tout le système décrit ci-dessus s'avère superflu, la probabilité d'extinction moteur en cloche étant si faible qu'elle ne justifierait alors que le siège éjectable du pilote, à l'exclusion de tout autre système capable de ramener l'avion.

DISCUSSION

E.Eckert, Ge

I refer to your slides "Emergency Power Sources" listing possible power sources and "Energy Storage System Range". I could not find the silver-zinc (one-shot) battery system there. As this has very high energy content and is well known for a long time and even in use with military aircraft is there a substantial reason why this has not been regarded?

Author's Reply

It is a very good question: The silver/zinc "one-shot" batteries are used on ballistic missiles and have been actually considered for the application described. The reason why we discarded them is because these batteries are "liquid-injected-electrolyte" type and cannot operate properly at a temperature lower than 0°, while the thermal battery used on the MIRAGE 2000 can start at −30°C.

M.Metohianakis, Gr

Can we use the thermal battery which usually delivers 30 volts to activate some other systems with different voltages simultaneously?

Author's Reply

I cannot see any reason why we could not use a thermal battery to activate systems at other voltages (like 100 volts for instance) provided an adequate number of voltages is added.

Supplying different available voltages simultaneously would also be theoretically possible but I think not practically feasible with the pesent technology of the battery which has been used in the case of the MIRAGE 2000.

OPTIMISATION OF ENGINE POWER OFFTAKE BY SECONDARY POWER SYSTEM DESIGN

by
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SUMMARY

This paper examines the configuration constraints presented to a systems engineer when designing a Secondary Power System, with particular reference to the minimising of engine power offtake. It briefly discusses the effects of power offtake on the engine and looks at current solutions available to the systems designer. The paper defines a Secondary Power System which is optimised for a twin engined, agile, combat aircraft.

DEFINITION

For the purpose of this paper the term Secondary Power System may be taken as the system which transmits power from the engines to and including the electrical and hydraulic generation systems, together with the engine starting and ground power systems. (fig 1)

SYMBOLS

APU	Auxiliary Power Unit
ATS	Air Turbine Starter
ECS	Environmental Control System
F	Thrust - Newtons
H.P.	Hydraulic Pump
Hp	Engine Shaft - High Pressure
Ip	Engine Shaft - Intermediate pressure
IDG	Integrated Drive Generator
M	Mass
N	Rotational speed - rpm
P	Pressure
POT	Power Offtake
PTO	Power Take Off
PRV	Pressure Reducing Valve
Q	Flow - Kg/sec.
SFC	Specific Fuel Consumption
SPS	Secondary Power System
T	Temperature

INTRODUCTION

Designing a Secondary Power System is an exercise in compromise, balancing the fulfilment of "desirable requirements" against such factors as Mass, Volume, Environmental conditions and engine power offtake, and producing the best system within the many constraints. Because the Secondary Power System is driven by the engine, the system should be designed such that its impact upon the engine's performance is minimised.

Three traditional goals for a fighter engine have been higher power-to-weight ratio, better SFC and improved reliability. These aims have resulted in the development of the relatively high by-pass ratio multispool turbo fan engine, with its smaller $\frac{N_{max}}{N_{min}}$ ratio and higher reheat-to-dry thrust ratio. The reheat boost on a 3-spool turbo fan engine can be as high as 70% and this results in a much smaller engine than its single spool equivalent, and gives a much better cruise SFC. However, the smaller core engine has a smaller core mass flow, Hp spool size, inertia and power, and because engine starting is achieved by driving the Hp spool, it follows that power offtake must be taken from the Hp shaft if a second tower shaft is to be avoided. This offtake now becomes a greater percentage of the total energy generated by that spool. Power off-take has also risen dramatically during the past 30 years (fig 2) due to the increasing sophistication and manoeuvrability of the weapons system requiring greater electrical and hydraulic generating capacity. Both trends make SPS design more difficult.

In order to appreciate the change in engine technology and power offtake, let us consider the two typical engine-secondary power system combinations shown on Fig 3. Engine A is a single shaft turbo-jet in the 15-17,000 lb reheat thrust class weighing in the order of 1,700 Kg. Engine B is a 3-shaft turbo-fan of a similar reheat thrust capability but weighing only 1,000Kg. The relative SFC's at max dry are 30 units, for engine A and 20 units for engine B.

For the next generation of fighter aircraft the "survivability" of the expensive pilot and aircraft will become a prime criterion of design. "Survivability" in this context involves the ability to outmanoeuvre and outfight the enemy weapons system. To facilitate this, the advanced fighter will need to be agile throughout the entire sub and supersonic regime. This aspect will have an immediate effect on the way in which the power offtake demands are made on the engine, as well as on the basic engine concept itself.

The aim of this paper is to look at the configuration constraints presented to the systems engineer and discuss the possible ways in which the system may be optimised using the technology available today. The paper uses this rationale in order to define a full Secondary Power System for a twin engined, unstable, agile, supersonic aircraft.

We will consider the following aspects:

System Requirements

System Constraints

System Optimisation

Definition

1. SYSTEM REQUIREMENTS

Primarily we are concerned with two of the basic areas which make up the functional requirements of any aircraft system.

Customer requirements

Technological improvements

Customer requirements will specify the basic parameters that the customer needs in order to meet the role he sees for his aircraft. For example these could be:-

- Aircraft state of readiness
- Engine start times
- Time from alert to aircraft getting airborne
- Type of standby required
- Need to be able to start engines without systems interrupt following standby
- Failure modes
- Maintenance requirements

These, in turn, will establish:-

- Need for a dedicated secondary power system
- Need for an auxiliary power unit
- Need for system redundancy
- Ejector bleed requirements
- Type of cooling system

Technological Improvements are the result of advances in the state-of-the-art and bring significant savings in mass, volume, or life cycle cost or improve reliability and maintainability.

Examples are:-

- Evolution of the Electrical Generating System
- Use of new lighter materials, titanium magnesium etc.
- High speed accessories
- Microprocessor control

It should be noted that the requirements for an 'agile' 'unstable' aircraft by necessity force the systems designer into seeking improvements in technology over and above the "evolution" of a product.

All these requirements must, of course, be accomplished within accepted 'Design Standards' (national or international) which will also establish test requirements.

2. SYSTEM CONSTRAINTS

The engineering constraints involve the consideration and minimising of:-

Engine Power Offtake

System Heat Rejection

Mass and Volume

Cost of ownership.

Engine Power Offtake

Fig 4 shows diagrammatically how the engine offtake is divided between mechanical and bleed for a typical twin engine fighter. Fig 5 gives us an idea of the waste power which must be rejected as heat to fuel or cooling air when all the processes are complete and it is this factor which has encouraged BAe Warton to do extensive Energy Management studies in order to optimise the complete energy statement of the aircraft.

The engine is predominantly designed to thrust, specific fuel consumption, and handling considerations. Power offtake will normally affect one or more of these parameters and to avoid unnecessary, undesirable effects on them, the systems engineer must design systems to minimise the offtake and must not be tempted to extract power irresponsibly.

Power offtake is not constant, but varies with the hydraulic and electrical demands made by the weapons system and control surfaces. Fig 6 shows a typical generalised compressor characteristic with lines of constant $N/\sqrt{T}$ plotted against compressor pressure ratio and non-dimensional mass flow $\frac{M\sqrt{T}}{P}$. Efficiency islands are shown in order to give an indication of the balance that the working line will need to take between the most efficient line for the engine and the one which gives adequate surge margin under all conditions. Fig 6 also shows the effects of power offtake and compressor bleed both generally and, for a fixed rpm, condition. Mechanical power offtake tends to push the engine towards surge, while air bleed moves the engine away from surge. The wider aspects of engine acceleration deceleration, and airflow distortion have been added to Fig 6 in order to give a better understanding of the total problem. It is normally the adverse combination of all these effects which results in engine surge. It is for such conditions as this that an accumulator is fitted to the hydraulic system. However all of these adverse effects are likely when a pilot needs to take sudden evasive action. i.e.:- demands power for the aircraft controls probably puts an acceleration demand on the engine and inflicts airflow distortion on the compressor inlet by increasing incidence and yaw.

Fig 7 indicates the specific quantitative effects of mechanical power offtake and compressor bleed on nett thrust, SFC and % surge margin for a particular flight case, for one spool of a current military engine, in a form which the SPS designer can use directly.

From the above it can be seen that the extraction of power offtake reduces thrust and compromises engine matching and thereby component efficiencies. Good, cost-effective systems design will result from a consideration of engine power offtake.

Secondary Power System Heat Rejection

Even for a fairly efficient Secondary Power System the heat rejection is considerable. Reference to Fig 5 shows that the predicted heat rejection from the total Secondary Power System (including hydraulics and electrical generation) in the early 1980's would be of the order of 100KW at normal operating conditions for a twin-engine aircraft. However, optimisation of current systems has resulted in considerable improvement against this prediction. The engine and secondary power heat rejection is normally rejected to fuel in the engine feed line and this is an excellent arrangement if the flow to the engine is sufficient to absorb all the heat without exceeding the engine fuel inlet temperature limit. This condition is normally met when the engine is in reheat or at max dry conditions. However, the efficiency of the small multispooled turbo fan is such that at cruise engine settings when the fuel flow is low, fuel must either be recirculated to tank, which results in a rise in bulk fuel temperature, or, the SPS oil must be cooled by ram air, again with the attendant drag penalty. Since fuel may be used to cool other systems it is more practical to use the fuel as a heat sink and protect the aircraft from excessive fuel tank temperature by means of an air cooled fuel cooler. This system involves the aircraft in a drag penalty, therefore, reduction in heat rejection will be beneficial both directly, by reducing cooler weight and volume, and indirectly, by reducing drag.

Mass and Volume

These parameters are critical for any component which is part of an aircraft system. Since larger and heavier systems must lead to a larger and heavier aircraft it follows that more fuel must then be carried in order to accomplish the same mission. However, by engaging in a complete system evaluation, the systems engineer can establish the exact design requirements for each element in the system. This will now ensure that each component will weigh only that which is necessary to obtain system performance.

Studies show, that for each kilogramme saved at this stage up to four kg. can be saved on take-off gross weight. This in turn results in:-

- A cheaper aircraft
- Reduced fuel for the same mission
- Increased engine thrust margin.

A good example of how technological advances influence the component weight can be seen by considering the development of the A.C. generator. Fig 8 shows how the specific power has increased fivefold over the last 30 years. The weight reduction has been accomplished mainly by increasing the speed of rotation of the generator from 6,000 to 24,000 rpm.

This feature has been used in the development of other pieces of equipment the hydraulic pump being one, where a proportional increase in flow results from an increase in rpm.

Some components have changed in form or principal of operation. New materials have been introduced, giving mass and other advantages. New processes make it possible to use existing materials in a new way which may also produce a mass or volume advantage.

Reduction in accessory size means that the secondary power gearbox can be lighter especially if the use of high speed accessories results in faster running, lighter shafts within the gearbox.

However, not all the components on a Secondary Power System rotate and these are unaffected by the introduction of high speed parts. Hence, there is a need to miniaturise valves, electrical controls and connections.

Cost of Ownership

Cost of ownership is the factor which should determine the validity of technological improvements and advances in the state-of-the-art. If "improvements" to any part of a complex system like the Secondary Power System are introduced at the expense of reliability, maintainability or overhaul life, then time and cost of repair and overhaul do not justify the "improvements". This is particularly true when a completely new system concept is suggested, then great care must be taken to fully analyse all the cost and spares requirements, and aircraft down time aspects before offering a solution.

It is worth noting, in passing, that cost savings resulting from weight reduction amount to over £20 per aircraft for each £1 invested in technology aimed at weight reduction, when considering a large aircraft fleet (500 - 1000).

3. SYSTEM OPTIMISATION

Having looked briefly at some of the constraints placed upon the systems designer, we may now consider some of the options which are open to him as a result of current state-of-the-art technology.

3.1 The Hydraulic System (reference 1)

The aircraft hydraulic system has traditionally been a constant pressure system with maximum pump power proportional to flow (stroke x rpm), the pump characteristic being shown on fig 9. The agile aircraft with its manoeuvrability and greater incidence considerations puts increased demands upon the hydraulic system, and the problem is not helped by the advent of control technology which permits maximum demands to be made anywhere in the flight envelope.

A number of solutions have been investigated including "soft-cut-off" pumps and "constant power" pumps whose characteristics are also shown on fig. 9.

Design studies at BAe Warton have shown that a considerable saving in power offtake can be made by varying the system pressure. Such a system was developed for the P110 aircraft and will be briefly discussed here.

Consider a system which is required to deliver 3.7 l/sec at 27 MN/m^2 per pump (one pump per gearbox). This is 100KW of fluid power, or 125KW with losses, per engine. When added to the other loads on the engine the total load approaches 200KW at max hydraulic demand. Hence the shaded area on fig 10 will effectively be out of bounds if engine constraints are to be respected.

It has also been shown that variation in maximum hinge moment for a typical primary control surface follows the curves shown on fig. 11. This means that at lower Mach numbers and higher altitudes the control forces are reduced. If the load curves are plotted then these can be converted into lines of constant pressure (fig 12) which in turn can be smoothed out into curves which can be expressed in two variables Mach No. and altitude.

On future aircraft the parameters necessary for pump "scheduling" will be obtained from the air data system and by means of modern microprocessor-based computers, the pump output pressure will be controlled as required.

Assuming that a minimum pressure of 15 MN/m^2 is decided upon, the hydraulic power demand in the case considered reduces to 55KW which results in an increase of the 'unlimited' flight envelope of about 5,000ft. Besides this gain, a reduction in total hydraulic energy usage in each flight will give a saving in fuel and a reduction in heat rejection.

3.2 The Electrical Generating System

This is currently a fairly efficient and demand conscious system of power distribution and the electrical loads on the aircraft tend to be fairly constant or switched on and off as required. Demand is not affected by Mach No or altitude. Some components, e.g. fuel boost pumps are a function of engine condition and could be matched to engine rpm rather than continue as constant speed A.C. pumps.

The advent of microprocessor based control would facilitate the matching of fuel boost pumps to engine conditions,and the careful application of maximum heating loads to the various missiles and stores. It would also allow load shedding following the failure of one generator. However, current studies at Warton indicate that the saving is small and suggests that this complication is not cost-effective on the current aircraft.

3.3 The use of the APU in flight (reference 2)

This seems at first glance to be an ideal mechanism for improving the engine power offtake situation, the APU taking part of the offtake otherwise supplied by the engine in critical engine conditions (low speed, high altitude). However, a brief examination of this proposal highlights the following problems:-

3.3.1 Installation

The intake/exhaust system must be so designed as to allow the APU to remain operative even in conditions in which the main engine may be subject to appreciable flow distortion. An installation using a flush intake on the lower surface, feeding an inlet plenum chamber, should be relatively unaffected by incidence and sideslip effects. An outlet on the upper surface would enable the benefit of any positive pressure difference, especially at high incidence, to be used. This arrangement reduces the effect of ram pressure on power output with increasing speed, but could eliminate the necessity for a separate source of emergency power. However, this installation requires considerably more volume than the simple ram intake/exhaust system and this is considered an unacceptable penalty.

3.3.2 Light Up and Running the APU at altitude

Studies at Warton, which are borne out by existing in-flight APU applications (Civil aeroplanes, etc), indicate that even a large APU will not readily light-up above 25,000ft or run reliably above 35,000 ft. This is mainly because of the effects of Reynolds number, blade thickness and blade tip clearance.

3.3.3 APU bay becomes a fire zone

The use of a dedicated in-flight APU would require the APU to become a fire zone with dedicated fire protection. This would add to the mass/volume penalty, as well as increasing the hazard situation.

3.3.4 APU Augmentation

It is almost certain that for the APU to carry part of the total power offtake over the full flight regime, it would require some form of augmentation. Two systems have been the subject of studies at Warton. The first requires the use of oxygen enrichment and is not generally considered to be mass and cost effective. The second requires main engine bleed air to be fed directly to the APU combustion chamber progressively closing down the passage between the APU compressor and combustion chamber with altitude. Although this second alternative appears to offer a viable solution it is again expensive and heavy.

From the studies into in-flight APU's carried out at Warton it is found that the efficiency of the main propulsion engine in producing power offtake even when the penalties of so doing are taken into consideration, is still sufficient to render the mass penalty of any approach utilising an in flight APU unacceptable for a twin-engined aircraft.

3.4 The use of Engine bleeds to improve Power Offtake capability

Two methods of optimising stage matching may be used:

(i) use of variable geometry guide vanes

(ii) use of 'blow-off' or 'bleed' valves

In the second case where the engine uses Ip and/or Hp blow-off valves, set to operate at a certain engine inlet condition, e.g. Mach No. altitude, total pressure etc., these bleeds could be used to augment power offtake.

Air bled from the engine compressor exit 'blow-off' valves at present,passes into the engine by-pass duct, with some contribution to engine thrust. This will be small, and the air may be better used to provide power additional to the shaft power offtake, by expansion through a turbine. This would have the effect of reducing the shaft power required to meet a particular systems power demand, which, in turn, would restore compressor surge margins, or enable a reduction in the 'bleed' valve air flows to be made.

Examples of the level of power which could theoretically be extracted from the 'blow-off' valve flow at say, 1.0M/50,000ft are:-

(a) Ip bleed, assume 0.5Kg/sec at 175 kPa and 500°K can be extracted. If expanded through a turbine to ambient at 80% efficiency this would give approximately 100Kw (neglecting duct losses).

(b) Hp bleed, assume 0.25Kg/sec at 450 kPa and 700°K can be extracted. If expanded - with some reduction in tapping pressure to give a feasible pressure ratio, would give approximately 70Kw. Whilst these levels of power are not always practically achievable, the potential power available in compressor matching bleed flows is considerable.

Additional power could be generated by the use of a bleed-and-burn system, with the penalty of the fuel flow required.

The penalty of extracting and utilising such 'bleed' flow is:-

- the loss of thrust recovered when the flow is directed into the fan duct - this is small, however, when compared with the loss of thrust due to extracting that bleed from the core engine.

- the mass of the bleed ducting and turbine together with the necessary mechanical drive from the turbine.

It is also worth mentioning that bleed extraction is not normally required within the normal (1g) envelope so that we are talking of optimising the systems in order to achieve the maximum potential of the aircraft.

3.5 Use of Engine Bleeds to augment the mechanical offtake (reference 2)

One way in which this concept could be utilised is by making use of a pneumatic link between the aircraft APU and gearbox-mounted air turbine starter motors, which could also be used in-flight for a pneumatically linked crossdrive. The necessary ducting would then be available to be utilised in the Air Turbine Starter on the gearbox driven by that engine to boost the power input available as shaft power, when the engine compressor bleed valves are open. A schematic of a 'Pneumatic Link' system of this type is shown in fig 13. In-flight when the system is required to operate, the air turbine would extract a nominal bleed flow from either the Ip or Hp compressor, as required to optimise the engine spool matching. The torque/speed characteristic of the air turbine would be arranged so as to provide a nominal power to the gearboxes. The control system would need to be so designed so as to prevent the air turbine motor providing the full requirements, since this would allow the air turbine to drive the secondary power system completely, allowing the PTO shaft to become off-loaded and thus producing a hunting situation between the two drives. It would also ensure that core engine thrust is not compromised by the extraction of too much bleed air. Since we are outside the 1g flight envelope, the engine is operating at, or near maximum conditions, (including rpm). Therefore by making the air turbine characteristics follow some unique function of engine intake total pressure (say), we should be able to obtain compatibility between the two inputs and ensure that the air turbine power is always usefully employed.

Other control systems might be envisaged or the gearbox could be designed so that part of the gearbox (that driving the hydraulic pump for example), could be driven at a higher speed by the air turbine, thus off-loading the P.T.O. shaft of that power requirement.

4. DEFINITION OF AN "OPTIMISED" SECONDARY POWER SYSTEM

In this section we shall look at the design for a Secondary Power System for which the requirements were such as to need some of the features described in Section 3. BAe Warton designed a complete Secondary Power System for an "agile" twin engined fighter, the P110. This aircraft has now become the ACA being built in collaboration with our partners M.B.B. and A.I.T. The system requirements are similar, requiring the optimisation of the Secondary Power System in order to maximise the full potential of the weapons system.

The P110 system was required to be flexible, simple, cost-effective and extremely reliable, and because of the 'unstable' nature of this aircraft there was a basic requirement to maintain, fully functional, at all times:-

1 engine

1 Hydraulic Pump

1 IDG

The following additional features were also necessary:-

- Ground standby capability
- Systems check-out capability
- Engine starting (without electrical systems interrupt)
- Independence of ground support equipment

The need for flexibility coupled with cost effectiveness suggested a pneumatic link system. This system shown in fig. 13 comprised the following:-

- Two airframe mounted gearboxes which could be driven either by PTO shafts or the gearbox mounted Air Turbines. Each gearbox drives:-
  - 2.9 l/sec 27 MN/m^2 hydraulic pump
  - 40/60 KVA I.D.G.
  - 10 Kg/sec First Stage Fuel Pump
- APU remotely mounted delivering:-

  Pneumatic power for gearbox checkout engine starting and ECS. It also provides mechanical power to drive a 20/30 KVA Alternator in order to provide standby electrical power.

Operating Modes

(i) Mechanical Each engine drives its own gearbox via a PTO shaft.

(ii) Pneumatic

(a) Engine driven Either gearbox could be driven via the air turbine by compressor bleed air from either engine. Each engine was protected by a check valve and the system was fed through a pressure regulating shut off valve. Air could then be fed to the Secondary Power System via the SPS shut-off valve and Air Turbine Control Valve and/or to the Environmental Control System via an ECS shut off valve.

(b) APU driven The APU drove either gearbox via the Air Turbine Control Valve and its own check valve. ECS air was supplied via the ECS shut-off valve.

The system offered other advantages which we will look at briefly:-

(i) The APU was remotely located giving the desired installational flexibility. An electrical generator was fitted to the APU, large enough to allow electrical systems check-out and standby, but small enough so as not to compromise the APU pneumatic output. This feature meant that the ground running requirements could be achieved without running the gearboxes, other than for a short duration hydraulic check-out (e.g. leakage check). It was envisaged that the complete hydraulic system would nominally be fully checked following engine start. The APU was self cooling.

(ii) The gearboxes were basic spur gearboxes with no complex control functions, they were simple, light and inexpensive. By designing the APU to be used in the ground running mode the overhaul life of the gearboxes was expected to be high, and linked directly to engine running hours.

(iii) Secondary Power System Cooling was simplified by not having to cope with heat rejection in the standby mode. The fuel was not needed as a heat sink, which meant an increase in mission effectiveness, following standby, particularly for the 'hot-day' considerations.

(iv) ECS air was readily available during APU running as was air for ejector cooling for any systems requiring this form of cooling.

(v) Engine starting would be achieved by motoring up the gearboxes and engines by means of the air turbine starters, electrical supply being maintained by the APU generator until the IDG came on-line. A mechanical disconnect between engine and gearbox allowed for the gearbox to be driven in isolation but meant that the gearbox needed to be stopped in order to engage the engine.

(vi) Air turbine starters and control valves were existing, cost-effective equipment, which gave a 'soft' connection between the APU and gearbox/engine.

In the engine running mode the gearboxes were driven mechanically via the P.T.O. shaft. Following engine failure or shut-down in-flight the gearbox of the failed engine would be driven pneumatically from the good engine. This would maintain surge margin on the engine at the expense of a small amount of thrust. The use of a pneumatic link cross-drive system was calcualted to give full system capability at altitudes above the '1g' flight envelope. The enhancement was certainly desirable if not necessary, considering the role of the weapons system.

Although not specifically included in the basic design the use of bleed air augmentation would have been a very useful development exercise since this would have given full systems capability (full SPS capability with aircraft manoeuvring and engine handling) well above the '1g' envelope.

(vii) Hydraulic System

The variable pressure hydraulic system (Sect. 3) was also included in the basic P110 design, giving a further enhancement of engine power offtake.

CONCLUDING REMARKS

(i) Engines of the same thrust class have become smaller and more efficient. The part of the engine from which power offtake is extracted has also decreased in size. Fig 3.

(ii) Power offtake required from the engine by the airframe systems has increased over the same timescale, making the Secondary Power System problem more difficult.

(iii) Good component and system design can influence the amount of power offtake taken from the engine, and tailor it to respect engine offtake capability.

(iv) A system designed to optimise engine power offtake can give full systems capability well beyond the normal flight envelope without compromising either engine or aircraft performance.

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ACKNOWLEDGEMENTS

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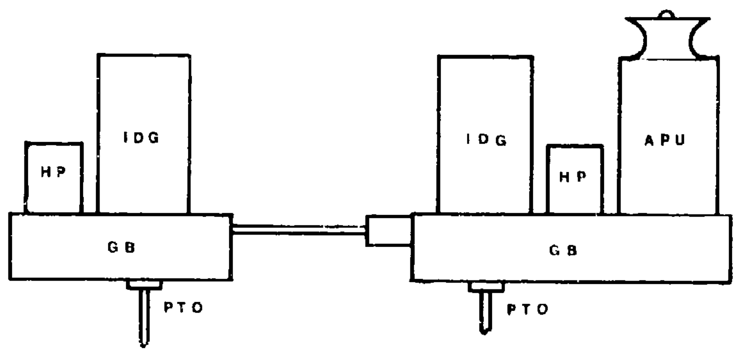

Fig 1 Secondary Power System

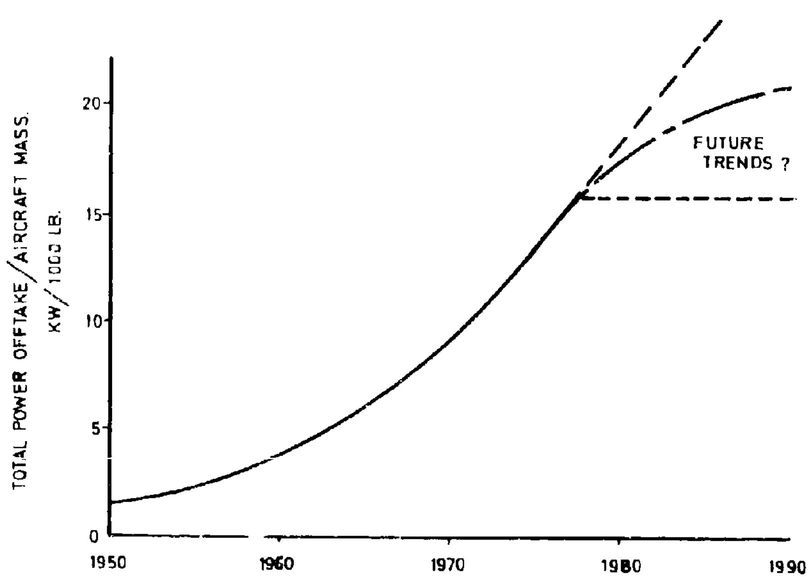

Fig 2 Power Offtake Trends

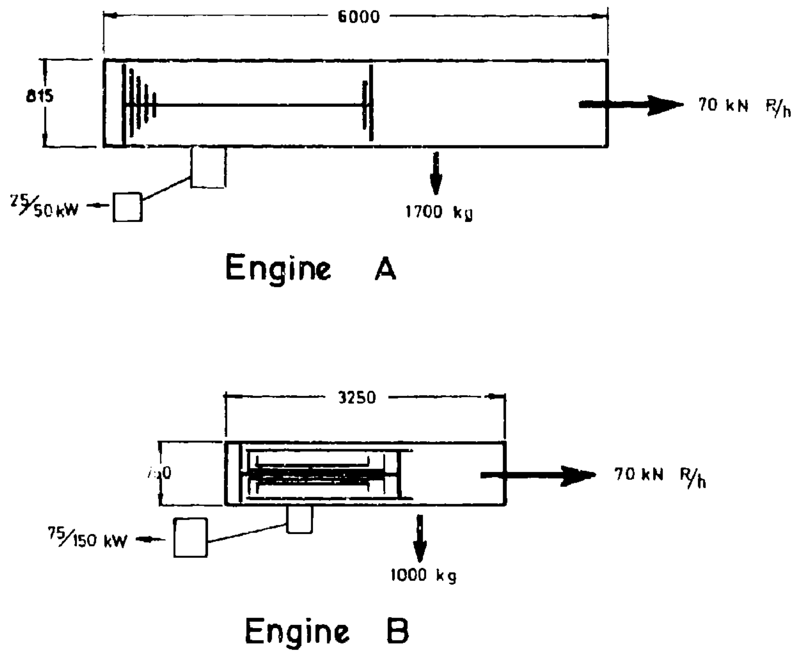

Fig 3 Engine - SPS Comparison

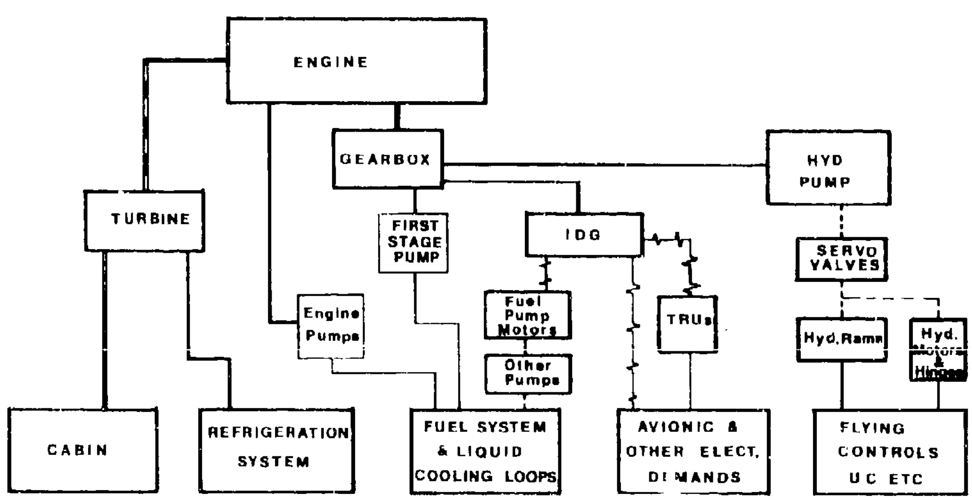

Fig 4 Secondary Power Distribution

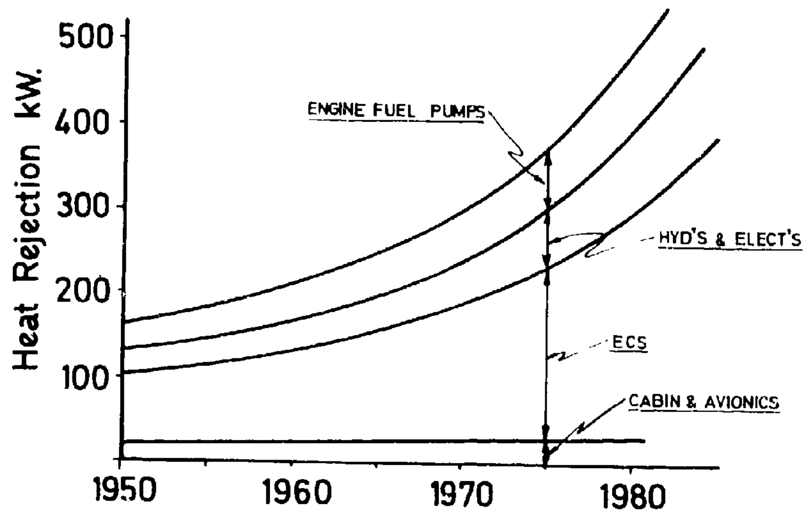

Fig 5. Growth of Aircraft Heat Rejection with Time

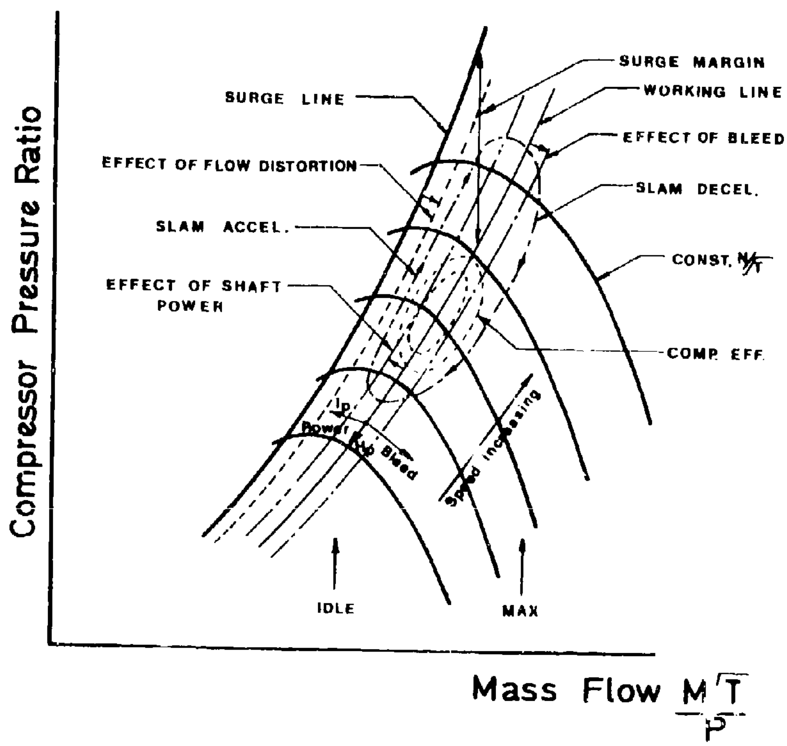

Fig 6. Generalised Compressor Characteristic

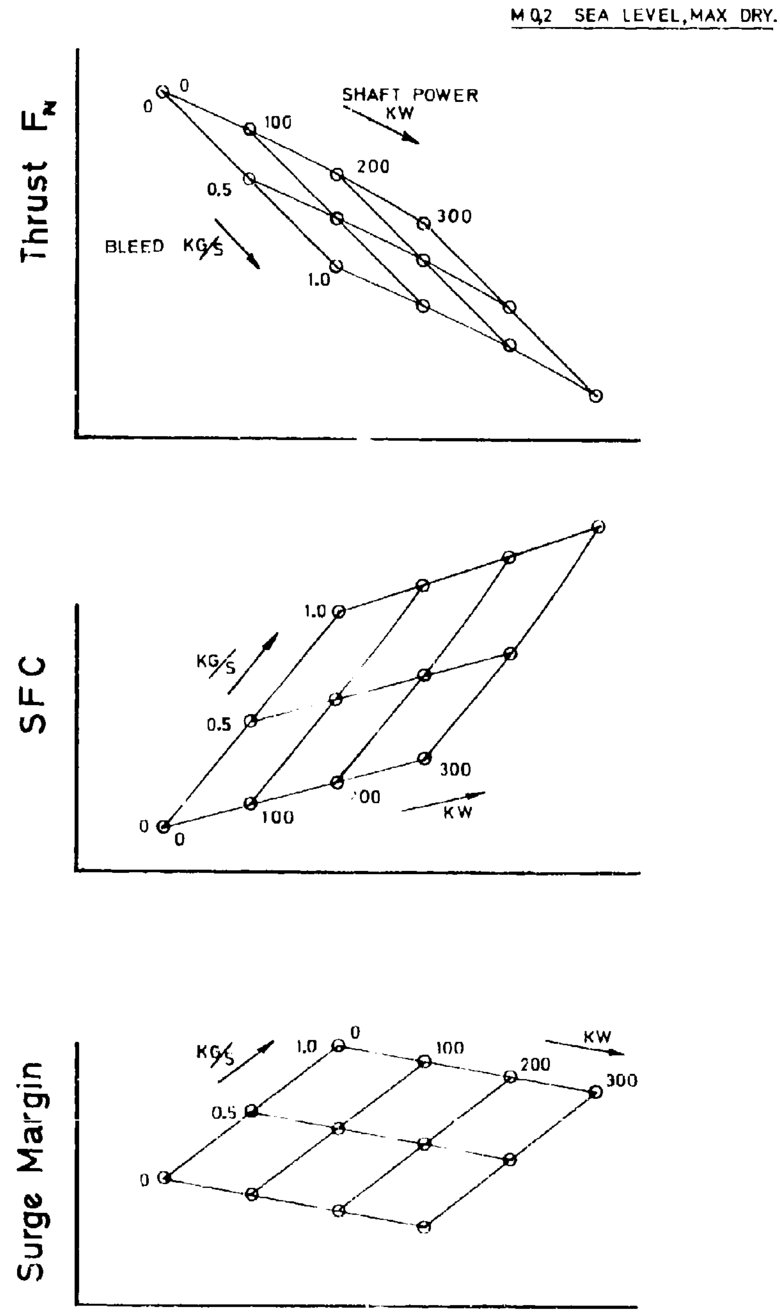

Fig 7
Variation of Thrust, SFC & Surge Margin with P.O.T.

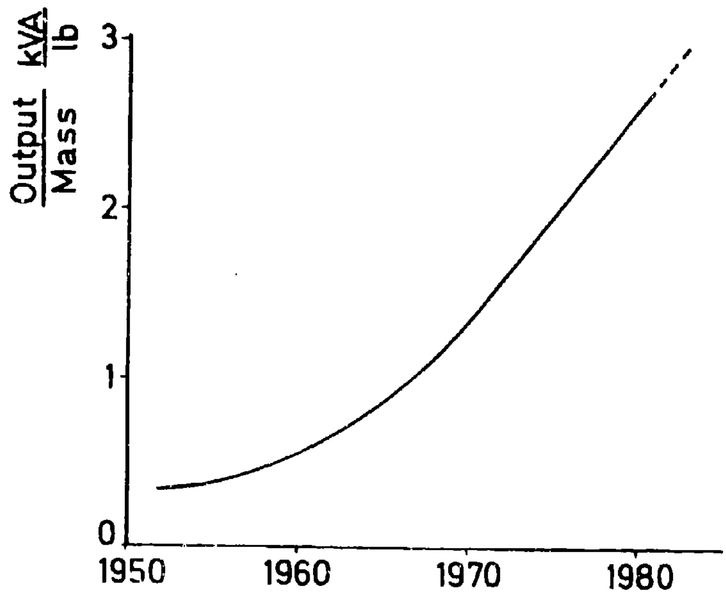

Fig 8

Aircraft A.C. Generator Power to Weight Ratio

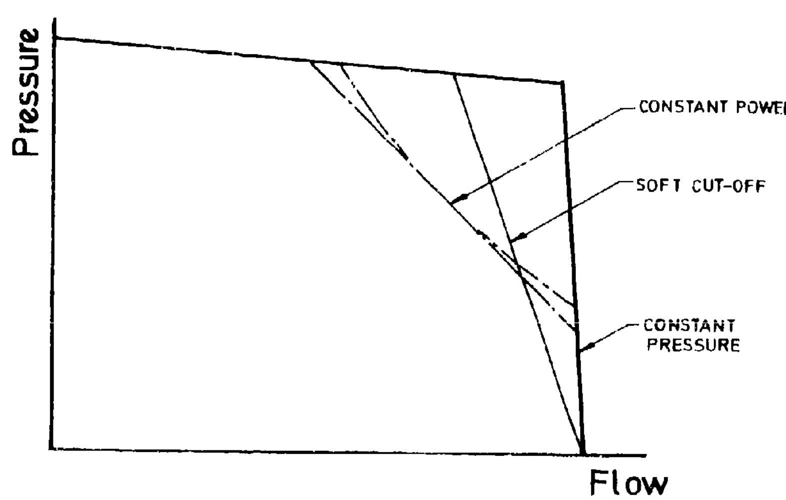

Fig 9 Hydraulic Pump Characteristics

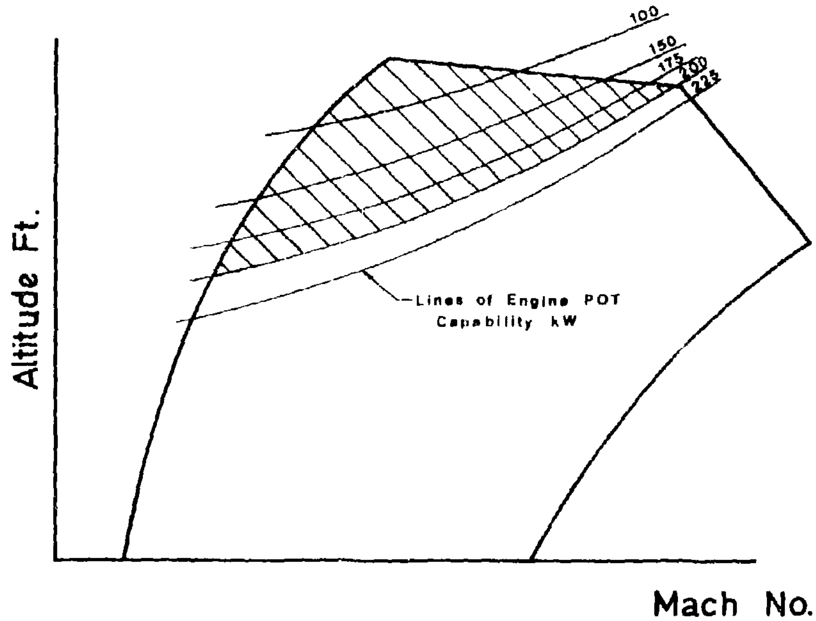

Fig 10. Engine Power Offtake Capability

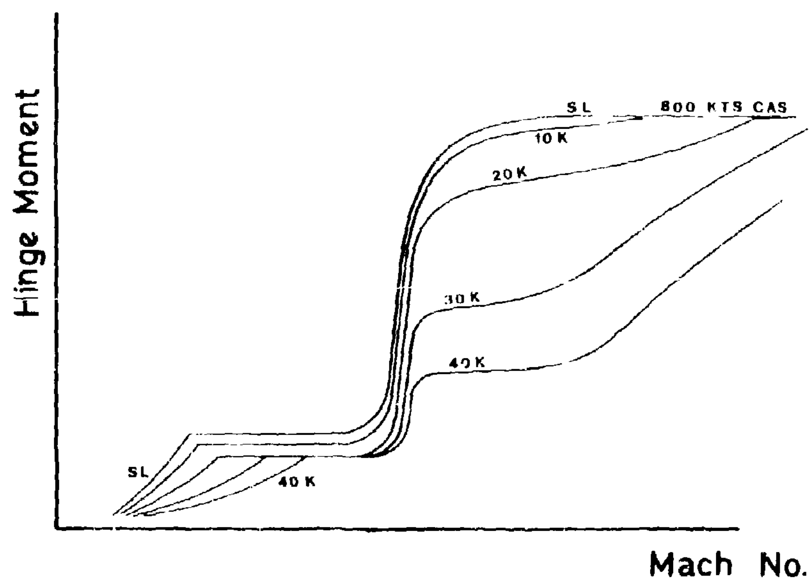

Fig 11. Typical Control Surface Hinge Moment

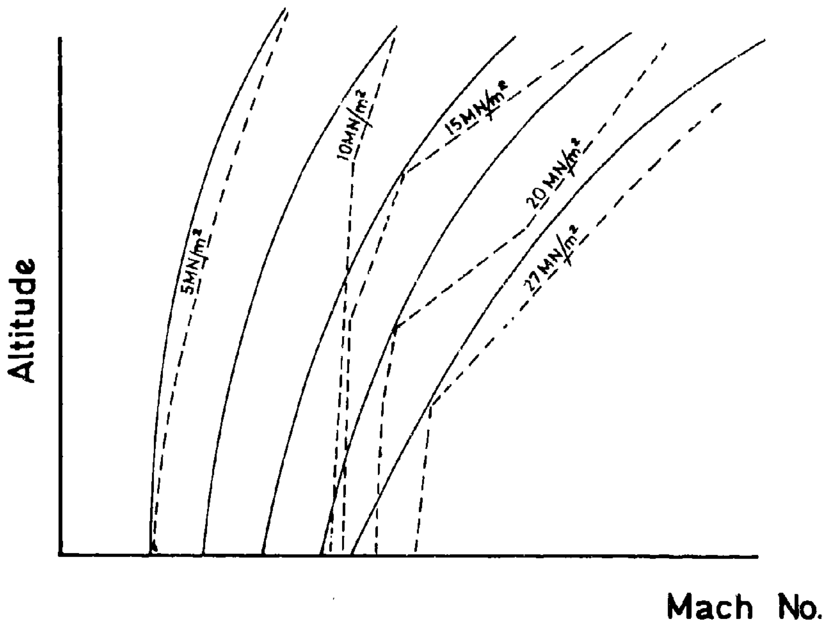

Fig 12. Hydraulic Pump Constant Press. Curves

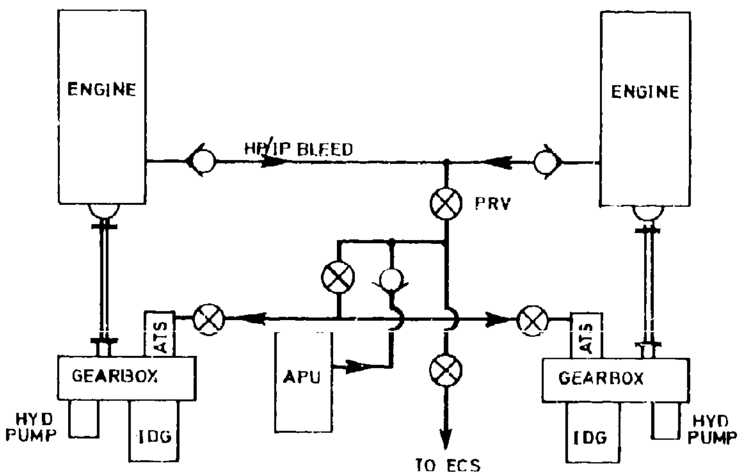

Fig 13. Pneumatic Link Secondary Power System

HOT GAS APU STARTER FOR ADVANCED AIRCRAFT APPLICATIONS

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ABSTRACT

Design, analysis, and testing has been conducted on a hot gas rotary vane motor for aircraft APU starting over the environmental temperature range of -65°F to +130°F, (-54°C to +54°C).

Experimental testing of the motor was conducted with gaseous nitrogen and with hydrazine based monopropellant hot gas decomposition products. Initial testing indicated problems with excessive gas consumption and binding of parts due to differential expansion. Subsequent revision to the motor configuration reduced gas consumption by 52 percent compared to the original baseline and eliminated end clearance sensitivity. Analytical studies, verified by test results, indicated the effect of friction coefficient, vane weight, venting, and blade linking on overall internal friction. Design approaches were evolved to minimize overall friction and loads on the vanes.

With design revisions implemented to solve initial problems, a motor successfully demonstrated operation at environmental temperatures down to -65°F as well as repeated restart capability. Design criteria have been evolved to allow application of the rotary vane motor to specific aircraft starting requirements.

NOMENCLATURE

A	=	base area of vane
A_N	=	radial component of acceleration
A_T	=	tangential component of acceleration
a	=	empirical constant, 0.25
b	=	constant, 0.7
F_3	=	tip force between vane and stator
F_{END}	=	contact force at end of vane
F_{RB}	=	reaction force at base of vane
H_R	=	height of vane exposed beyond rotor
J	=	proportionality constant, 788 ft-lbf/Btu
LV	=	length of vane within rotor slot
M_V	=	vane mass
P	=	pressure
PF	=	force produced by pressure differential acting over exposed vane area
PF_{base}	=	vane base pressure
PF_{tip}	=	vane tip pressure
Q_1	=	ideal available energy, Btu
Q_A	=	actual available energy, Btu
Q_F	=	energy loss due to friction, Btu
Q_L	=	energy loss due to leakage, Btu
Q_Q	=	energy loss due to heat flow, Btu
Q_R	=	energy required to produce desired performance, Btu
TV	=	vane thickness
V	=	volume
W_N	=	radial force due to acceleration acting on blade mass
W_T	=	tangential force due to accleration acting on blade mass
XV	=	length of vane within rotor slot
A	=	actual efficiency
γ	=	ratio of specific heats
μ	=	coefficient of friction

SUBSCRIPTS

1	=	inlet control volume between blades
2	=	isentropic expansion from condition 1 to control maximum volume
3	=	ambient atmospheric conditions

CONCEPTUAL DESIGN

Current military aircraft auxiliary power units (APU's) are generally started with either a battery powered electric motor or with a pneumatic accumulator driven hydraulic motor. At low temperatures (-40°F to -65°F) these systems often contain insufficient energy for a single start attempt and have no capability for restart.

A review of the problems and shortcomings of the current pneumatic-hydraulic APU starter concept has led to the definition of an advanced APU starter system. This system employs a high pressure hot gas source in a rotary vane expander as the prime mover to start the APU. In the baseline design, the hot gas is provided by the decomposition of a hydrazine-based fuel blend although other hot gas sources could be employed.

The major benefits to be derived from this type of advanced system include:

1. Reduced weight and volume for a single start.
2. Multiple start capability with minimal time between restarts.
3. True -65°F system start capability.

A conceptual sketch of the system is shown in Figure 1 which also includes a schematic of the system approach. High-pressure nitrogen is contained in a removable pressure vessel (2) and provides the gas source for pressurizing the fuel. The gas control valve/regulator (4) may be permanently installed in the aircraft and regulates nitrogen pressure to the fuel tank. The hydrazine-based fuel supply is contained in a removable positive expulsion pressure vessel (5). The fuel control valve (6) sequences startup flow to assure that the maximum torque limits are not exceeded as the gas generator (7) decomposes fuel into hot gas and delivers it to the hot gas motor (8). Fuel and gas capacity may be sized to provide the desired number of starts prior to refilling the tankage.

Technology exists for all components of the system except for the hot gas rotary vane expander. This paper discusses the design, analysis and test efforts conducted to develop the device. This work was conducted by Rocket Research Company under contract F33615-76-C-2148 with the Air Force Aeropropulsion Laboratory, Air Force Systems Command, Wright-Patterson AFB, Ohio.

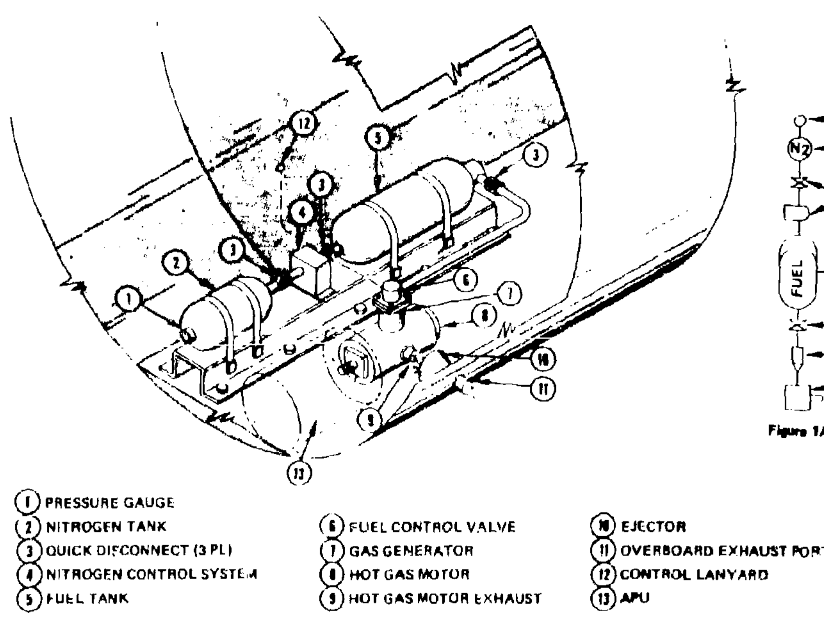

Figure 1. Hydrazine APU Starter Concept

STARTER OPERATING REQUIREMENTS

The design requirements for the hot gas rotary vane expander were based upon the torque speed characteristics of an advanced APU, which is felt to represent the most stringent starting requirements for future aircraft. These requirements are shown in Figure 2 for three ambient temperatures. The critical requirement occurs at -65°F (-54°C) where a high breakaway torque is required at zero speed, with a second torque peak slightly above 20,000 rpm (2,090 rad/s). A 10:1 gearbox reduces the required motor speed to a reasonable operating range and increases the required motor torque by the same factor. Since the torque speed curve for the rotary vane expander is relatively flat, the midspeed range was selected as the design point for the motor, including allowances for gear box efficiency (85 percent) and a 32-percent torque margin.

BASELINE MOTOR DESIGN DESCRIPTION

The dimensions of the motor and porting arrangements are shown in Figure 3. For high expansion ratio and adiabatic efficiency, the arc of admission must be kept small, generally less than 17 degrees (0.3 rad). In this range Reference (1) shows efficiency improvement up to as many as 12 vanes. Tradeoffs of rotor and vane strength led to a baseline motor design with eight blades.

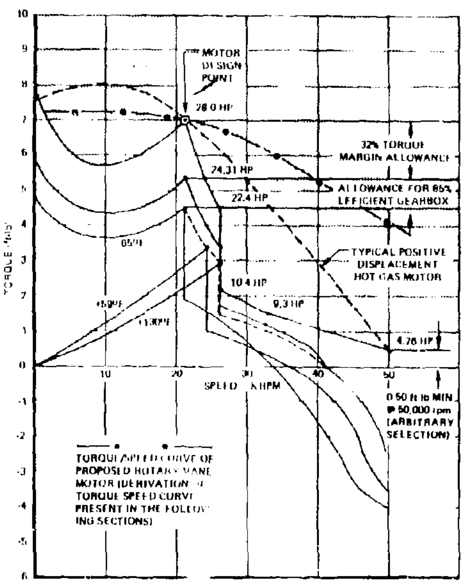

Figure 2. Starter System Envelope

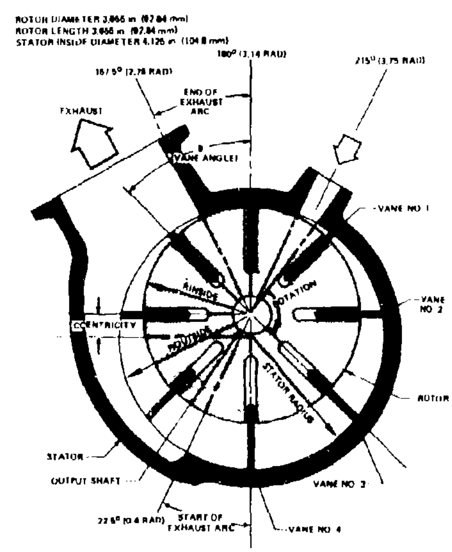

Figure 3. Rotary Vane Motor

The major parts of this motor, shown in cross section in Figure 4, include the ball-bearing mounted rotor, eight rotor vanes, the stator shell, end plates which support the rotor bearings and position the rotor eccentrically in the stator bore, and various seals and springs. Each end plate contains an eccentrically mounted boss. These retain flexible split cam rings which bear on the base of each blade and provide a force to preload the blades against the stator. The mating contact points on the blades are provided with metal slippers, pinned to the vane. Segmented end seals located circumferentially between the vanes at both ends of the rotor are spring loaded against the end plates to minimize leakage to the bearings. The bearings are further protected from hot gas by deflectors arranged in labyrinth form together with large overboard vents to preferentially duct gas away from the bearings. The disassemblied motor is shown in Figure 5.

Motor clearances were established based on a combined structural and thermal analysis assuming nominal gas inlet conditions of 1,000 psig (6,895 kPa) and 1,600°F (871°C), and deflection occuring due to pressure loading as well as thermally induced stresses. Major materials of construction include:

Rotor	Waspaloy
Stator	Hastelloy C
End Plates	Hastelloy C
Deflectors & Bearing Seats	Titanium 6Al-4V
Split Cam Rings	Inconel X-750
Vanes	P-658 RCH Carbon

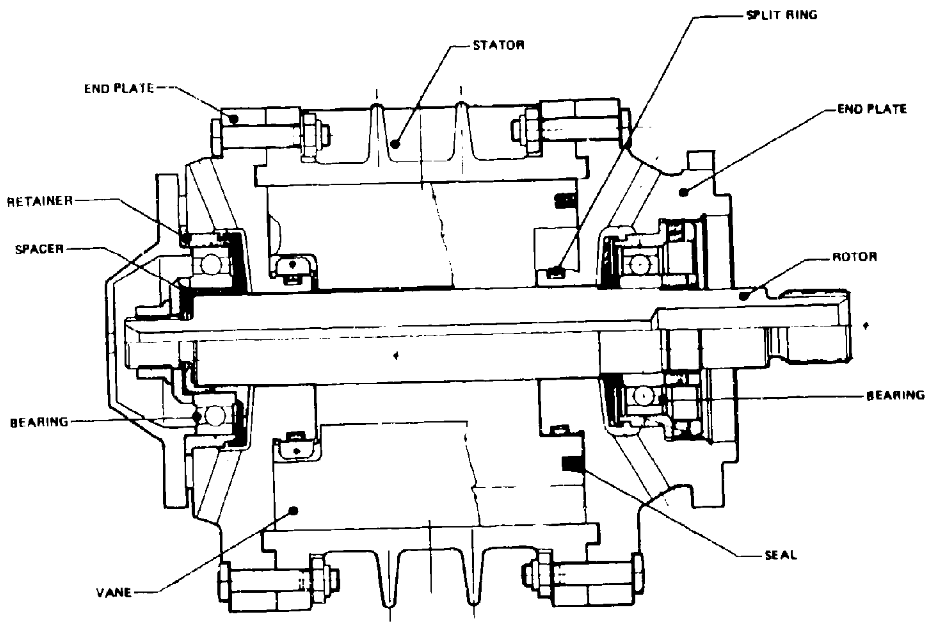

Figure 4. Cross Section of Rotary Vane Motor

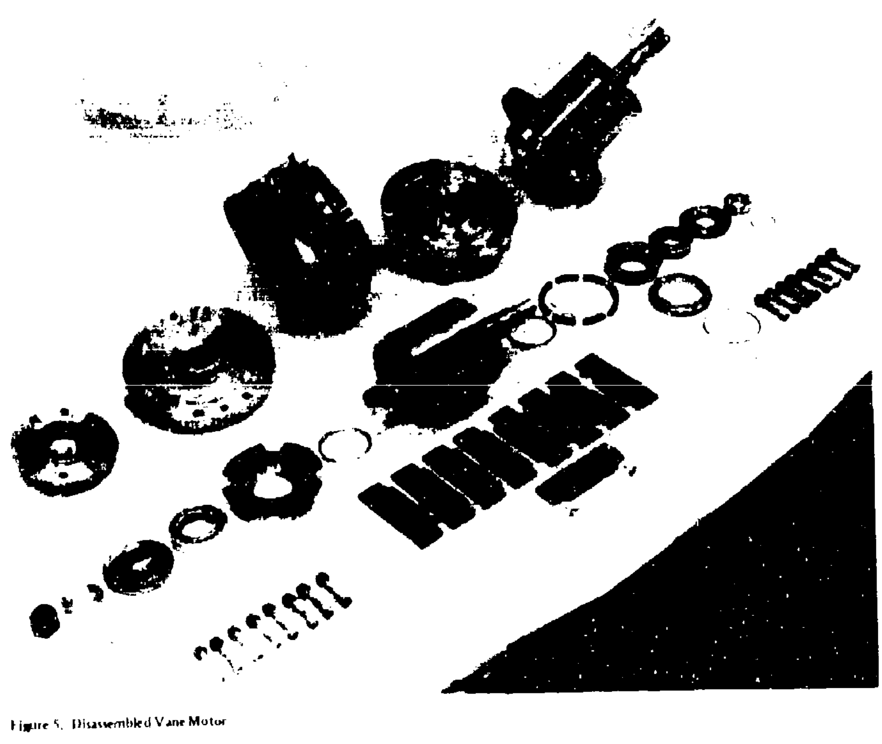

Figure 5. Disassembled Vane Motor

THERMAL-STRUCTURAL ANALYSIS

A computer program was prepared to assist the structural design by predicting the temperature of major piece parts in the motor. Thermal networks were established representing conduction, convection and radiation modes of heat transfer. The conduction network is shown in Figure 6.

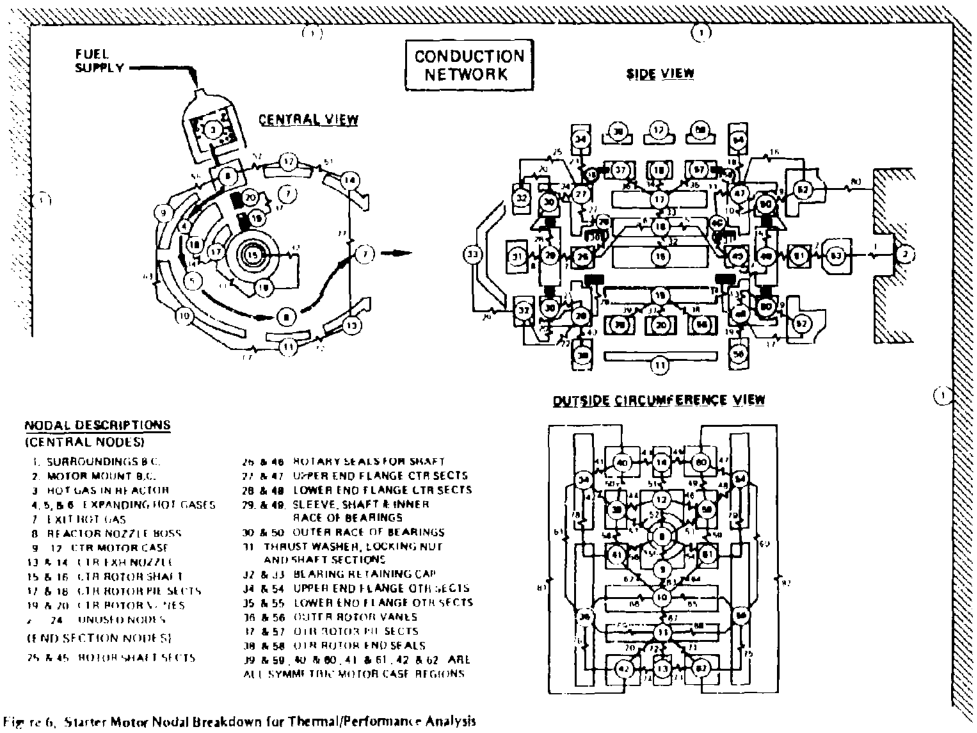

Figure 6. Starter Motor Nodal Breakdown for Thermal/Performance Analysis

A detailed thermal network was employed to accurately predict local thermal gradients and hot spots. Such predictions are important because they are used for thermal expansion and rotor/vane/stator tolerance determination. Such tolerancing is critical due to the opposing requirements to minimize leakage losses while preventing metal-to-metal contact and motor seizure.

The thermal-structural model consists of 62 nodes, 82 conduction resistance elements, 86 convection resistance elements, and 60 radiation elements. Nodal finite difference heat balance equations are solved on a CDC 6600 computer with a 500-node capacity thermal analysis program.

Externally the motor was assumed to be exposed to ambient air ranging from -65° to $+130^{\circ}$F (-54° to $+54^{\circ}$C). Free convection was assumed along with radiation to the ambient environment. Conduction coupling of the motor by the relatively massive gearbox and APU was assumed to be minimal due to the rapid transient involved.

Based on performance calculations and anticipated operational procedures, cold and hot environment thermal analyses were made assuming 12– and 7-second firings for the low and high temperatures respectively, followed by a 1-minute soakback, subsequently followed by another firing and soakback period.

Major local piece part temperatures resulting from the analysis are plotted in Figures 7, 8, and 9. Figure 7 shows significant thermal gradients exist in the rotor and vanes from root (nodes 17 and 19) to exposed ends (nodes 18 and 20). Node 17 is 0.7 inch (18 mm) radially in the rotor from the surface node 18. Node 19 is 1.26 inches (32 mm) inside the vane from surface node 20.

Figure 8 shows temperature gradients for structures situated near the bearings, plus gradients within the bearings themselves. Figure 9 shows typical stator case temperature and gradients.

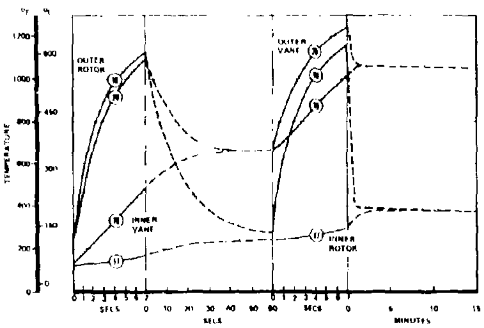

Figure 7. Rotor and Vane Transient Temperatures (Hot Environment)

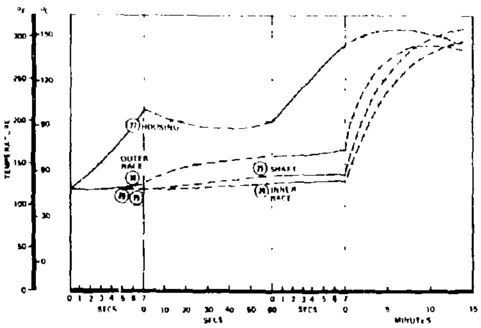

Figure 8. Bearing, Shaft, and Housing Transient Temperatures (Hot Environment)

PERFORMANCE ANALYSIS

Performance analyses were conducted to predict motor speed and fuel consumption as a function of hot gas inlet pressure. The analyses also calculated shaft power available out of the gearbox, torque (available and required), motor efficiency and heat, leakage, and friction energy losses. For the analysis, the motor was divided into a nodal network (similar to the thermal-structural analysis but with less nodes) as shown in Figure 10, and finite difference transient solution techniques were used for performance analysis.

Leakage and friction loss calculations were based upon work performed under contract by Professor C. H. Wolgemuth of Pennsylvania State University. The performance model is developed around the ideal energy available from the general expander open cycle depicted in Figure 11. Summary work around the cycle gives the following expression (Eq. 1) for ideal energy available, Q_1.

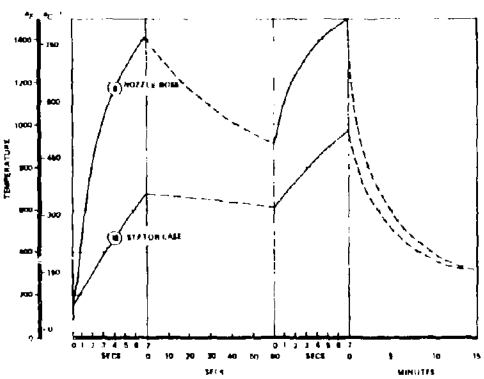

Figure 9. Stator Local Transient Temperatures (Hot Environment)

$$Q_1 = \frac{(P_1V_1 - P_2V_2)}{(\quad - 1)\, J} + (P_2 - P_3)\,\frac{V_2}{J} \tag{1}$$

The heat, friction, and leakage losses must be subtracted from Q_1 to calculate the actual output. This results in the following expression (Eq. 2) for the efficiency of the motor:

$$A = \frac{Q_1 - (Q_Q + Q_F + Q_L)}{Q_1} \tag{2}$$

The available energy to produce useful shaft work is then calculated from (Eq. 3):

$$Q_A = \eta_A Q_1 \tag{3}$$

If Q_A is greater than the energy required by a sufficient margin, the motor design is satisfactory.

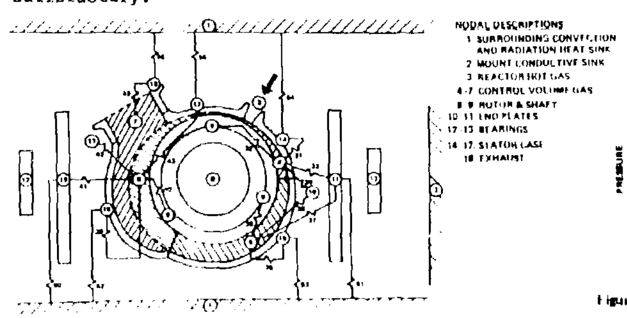

Figure 10. Thermal/Performance Model Convection Network

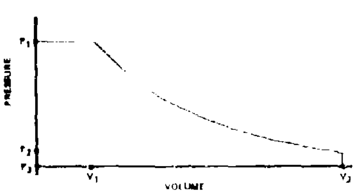

Figure 11. Thermodynamic P-V Diagram for Expander Motor

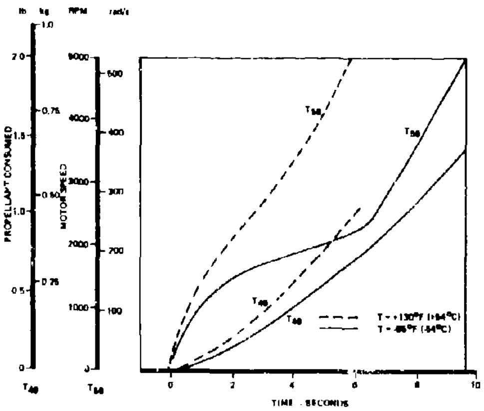

Figure 12. Predicted Starter Performance

The thermal-performance computer program was utilized to analyze the motor performance over the temperature range of -65°F (-54°C) to +130°F (+54°C). This analysis indicated that an inlet pressure to the motor of 1,500 psia (10,340 kPa) was required to produce the desired acceleration rates.

Figure 12 presents typical performance predictions for the motor over the environmental temperature range showing speed and propellant consumption versus time. As noted, the analysis predicts the motor will crank the APU to 5,000 rpm (524 rad/s) in 6 to 10 seconds and that the fuel consumed per start ranges from 0.97 to 1.34 lbm (0.44 to 0.61 kg). More fuel is expended at the lower temperature due to the higher APU torque requirements. The analysis indicates that leakage losses are predominant at low speeds, and frictional losses control at high speeds.

COLD GAS TESTING

Initial test efforts were conducted with high-pressure gaseous nitrogen to characterize the motor operation. The test schematic for this phase of the testing is shown in Figure 13. As noted, the starter motor is coupled to a flywheel via an overrunning clutch, and motor startup is controlled by a pressure ramp generator in the test system.

The flywheel, clutch, shaft and other rotating parts have a combined inertia of 27.7 lbf-ft^2 (1.17 kg-m^2) and were sized to produce the same time to reach 5,000 rpm (524 rad/s) as would occur with the actual APU torque curve of Figure 2 although the slopes will differ throughout the run. The nominal flywheel is a compromise between ambient and high temperature requirements. A second flywheel with an assembly inertia of 41.2 lbf-ft^2 (1.74 kg-m^2) is employed to simulate the -65°F (-54°C) APU torque requirement.

Initial cold gas testing consisted of stall torque tests to assure maximum torque li   were not exceeded. With 1,000 psig (6,890 kPa) inlet pressure and the vane directly un   the inlet, measured stall torque was 80 ft-lbf (108 N-m). The maximum permissible torque is 120 ft-lbf (163 N-m). With the inlet pressure held constant at 300 psig (2,070 kPa) and the vane position varied in 5° (0.09 rad) steps, the stall torque varied from 30 to 60 ft-lbf (41 to 81 N-m).

Following the stall torque tests, initial spinup tests were attempted. These initial tests revealed clearance problems, vane breakage at the cam ring interface, and excessive end seal drag. Subsequent tests with phenolic vanes and design changes to the cam ring eliminated these problems, and vent slots were added to the vanes to provide improved force balance and increased performance.

Final cold gas testing was conducted with carbon/carbon composite vanes incorporating a vented slot. A total of eight tests were conducted with the motor as shown in Table 1. The time to speed achieved in each of these tests was consistent with the desired motor performance; however, gas consumption at low speed was excessive. The gas consumption was relatively independent of speed and slightly above analytical predictions at maximum speed. The results were sufficiently successful that a decision was made to proceed with initial hot gas testing.

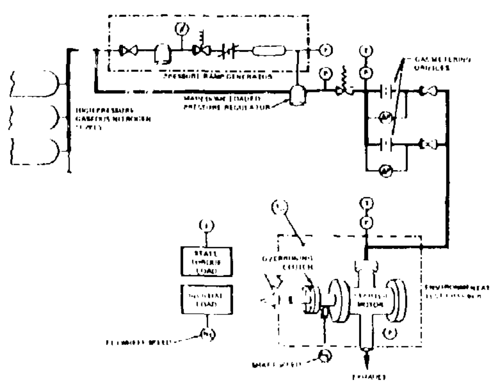
Figure 13. Test Setup Cold Gas Testing

Table 1. Carbon/Carbon Composite Blade Performance

Test	Conditioning Temperature °F	(°C)	Peak Ramp Pressure lbf/in.^{2} (kPa)	Time to Maximum Speed, sec.
1	50	(10)	500 (3,450)	14.0
2	50	(10)	1,000 (6,900) 1	7.6
3	50	(10)	1,250 (8,620) 1	6.6
4	50	(10)	1,250 (8,620) 1	5.9
5	50	(10)	1,200 (8,270)	5.8
6	50	(10)	1,200 (8,270)	6.3
7	65	(54)	1,200 (8,270)	6.6
8	+130	(+54)	1,200 (8,270)	6.4

1 Pressure not maintained at end of run.

INITIAL HOT GAS TESTING

The test system employed for hot gas testing is shown in Figure 14. The pressure ramp generator is retained from the cold gas installation and is used to pressurize a tank containing a monopropellant hydrazine fuel blend. Flowmeters, fuel control valves, and a gas generator are added as shown for the hot gas test system.

The propellant selected for the tests is a ternary blend consisting of 58 percent hydrazine (N_2H_4), 25 percent hydrazinium nitrate ($N_2H_5NO_3$), and 17 percent water (H_2O) by weight (termed TSF-2 fuel). This mixture has a freezing point close to -90°F (-67°C) and has thermochemical performance as indicated in Tables 2 and 3. It had been previously used in Reference 2 testing of a hydrazine-fueled aircraft starter cartridge. The ammonia dissociation fraction shown in Tables 2 and 3 varies from 0.4 to 0.6 depending upon the environmental temperature.

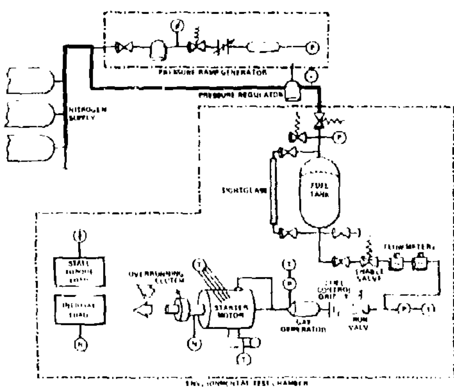

Figure 14. Test S[illegible] Gas Testing

Table 2. Thermochemical Performance Characteristics of TSF-2 Fuel

NH_3 Dissociation Fraction X	Fuel Supply Temp. °F (°C)	Gas Temp. °F (°C)	Molecular Weight (M_c)	Ratio of Specific Heat Capacities λ_c
0.4	65 (54) +77 (+25) +160 (+71)	1,760 (960) 1,929 (1,054) 2,033 (1,112)	16.500 16.500 16.500	1.2424 1.2361 1.2327
0.6	65 (54) +77 (+25) +160 (+71)	1,574 (837) 1,747 (969) 1,853 (1,012)	15.310 15.310 15.310	1.2726 1.2653 1.2612

Table 3. Exhaust Gas Composition of TSF-2 Fuel

Ammonia Dissociation Fraction X	Exhaust Gas Composition, % by Volume			
	Nitrogen	Ammonia	Hydrogen	Water
0.4	24.4	23.4	23.3	28.9
0.6	20.2	14.4	32.4	26.9

The ex[illegible] low speed gas leakage noted in cold gas testing caused thermal expansion problems [illegible] hot gas testing. The motor consistently jammed due to differential heating of [illegible] parts which eliminated end clearances. Increasing the end clearances resulted in [illegible] and aggravated the problem rather than providing a solution. In addition [illegible] excessive leakage, some warpage of the cast stator was noted.

[illegible] test problems associated with stator warpage, a new heavy-wall stator was mach[illegible] stock. The exhaust outlet in the stator was modified to a series of holes from an [illegible] slot to equalize vane wear resulting from crossing the slot.

To reduce the differential thermal expansion problems, a series of tests were conducted to evaluate means of reducing leakage in the motor and thus total heat input. Results of this testing indicated that:

1. The machined bar stator produced 34 percent less leakage than the cast stator, apparently due to lower shell deflection and reduced leakage areas.

2. The end seals produced significant drag and were not effective in reducing overall motor leakage.

3. The major leakage path was occurring through the bearing vents. A 52 percent drop in flow occurred when the vent ports were plugged.

Results of the above tests, combined with binding and vane breakage noted in previous hot gas tests, indicated four areas needing redesign. These areas were:

1. An improved method of maintaining vane-to-stator contact

2. Improved end seals

3. A means of increasing end clearance without increasing overall leakage

4. Reduction in the forces on the vanes or an increase in vane load bearing capability.

DESIGN MODIFICATIONS

To obtain vane-to-stator contact, the cam ring actuation system was replaced with four pairs of pushrods installed in the rotor between opposed pairs of vanes. The pushrods were spring loaded to accommodate an approximately 0.030-inch (0.76-mm) geometrical length change which occurs between the stator diameter (0-degree position) and the chord across the stator through the rotor center (90-degree position). Several cold gas nitrogen tests with unvented phenolic vanes resulted in improved time of 7.8 seconds to 5,000rpm (524 rad/s) with 1,125-psia (7,760-kPa) inlet pressure. The gas flow rate was 85 percent of the baseline cast stator run.

To improve end sealing and increase the end clearance, a design modification was made to the motor as shown in Figure 15. The rotor and stator were each shortened by 0.5 inch (12.7 mm). Closure plates, each 0.25-inch (6.4-mm) thick, were bolted to each end of the rotor; and spacer plates, machined eccentrically with respect to the stator (concentric with rotor), were also installed. Three-piece, spring-loaded carbon seals, installed in the rotor end plates, contact the I. D. of the eccentric spacer to form the end seal. This modification reduces leakage into the end cavity with a reduction in thermal distortion effects; makes end-cavity leakage independent of end clearance, allowing increased end clearance as necessary to accommodate axial growth; and leakage around the ends of the vanes is prevented from being vented overboard, allowing useful work to be done.

Cold gas nitrogen tests to charactrize the above design changes, along with spring-loaded pushrods and phenolic vanes vented on the trailing edge, demonstrated significant improvement in motor performance. The tests indicated that addition of the end seals produced an order-of-magnitude reduction in end cavity pressure as shown in Figure 16. Time to achieve 5,000 rpm (524 rad/s) was reduced to 5.2 seconds, and gas consumption was reduced to 68 percent of the cast stator run (Run 6 of Table 1). Other tests to evaluate leading edge versus trailing edge vane slots indicated a 14 percent improvement in time to speed with leading edge slots.

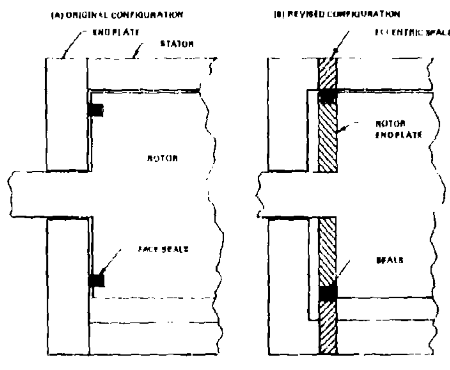

Figure 15. Comparison of Vane Motor Configurations

Figure 16. Effect of Seals on Leakage-Produced End Cavity Pressure

Several alternate blade materials were examined to improve the blade strength. Included in the evaluation were various high-temperature lubricant coatings on metal blades to reduce the friction coefficient. Alternate blade materials examined were:

1. Inconel 600 blades coated with SURF-KOTE C-800, the trade name for a metal-glass-fluoride, high-temperature lubricant applied per NASA Specification PS-101 (References 3 and 4). The coating is applied as a plasma spray and consists of 30 percent nichrome, 30 percent silver, 25 percent calcium fluoride, and 15 percent glass by weight.

2. A sintered nickel-chromium alloy impregnated with barium fluoride-calcium fluoide eutectic (Reference 5).

3. A special high-temperature grade of cast iron containing high aluminum content. The composition consists of 1.5 percent carbon, 3.65 percent silicon, 0.7 percent manganese, 21.5 percent aluminum, and 72.6 percent iron by weight.

4. Carbon/carbon composite infiltrated with silicon carbide.

5. 7075 aluminum with a proprietary Banadize coating (Lovalt Technology Corporation, Santa Fe Springs, California).

Initial testing with the metal vanes resulted in slow times to achieve motor speed and/or the inability to achieve the required terminal speed indicating high internal friction. The high internal friction was due in part to the greater weight of the metal vanes (up to six times that of the composite vanes) and the higher friction coefficient of the coated metal vanes. This problem was compounded by excessive wearing of the high-temperature lubricant coating. Testing with the carbon/carbon composite vanes infiltrated with silicon carbide resulted in vane breakage during all tests.

VANE FRICTION ANALYSIS

To better understand the effects of vane weight and friction coefficient on motor performance, a series of analytical studies was performed by Professor C. H. Wolgemuth under subcontract to RRC. The analytical model previously developed was modified to include pushrods between the vanes. A somewhat simplified approach was taken which ignored the pushrod spring force, but allows transfer of force between pairs of vanes. The vane free body diagram is shown in Figure 17. The model includes normal forces due to ω^2 and d^2r/dt^2 components, tangential forces resulting from $d\omega/dt$ and Coriolis components, pressure forces in the normal and tangential directions, and gravitational forces. In the axial direction, friction force (identified as F_{END} in Figure 17) is developed only on that portion of the vane exposed above the rotor as it contacts the eccentric spacer of the revised design.

In order to minimize or eliminate vane weight effects, a concept was analyzed in which the opposing vanes are rigidly lined together to counterbalance the weight. A model of this concept is shown in Figure 18.

Pressure in the control volume behind the vane (lagging control volume) was compiled from the leakage model using an inlet pressure of 1,115 psia (7,690 kPa). Results are shown in Figure 19, where the centerline of the inlet port occurs at θ = 215 degrees (3.75 rad). Forces, available gas power, and friction power loss versus angle were computed for various combinations of friction coefficient, inlet pressure, rotational speed, and vane porting for both the rigid-linked vane and pushrod vane models. Results of these calculations are summarized in Table 4.

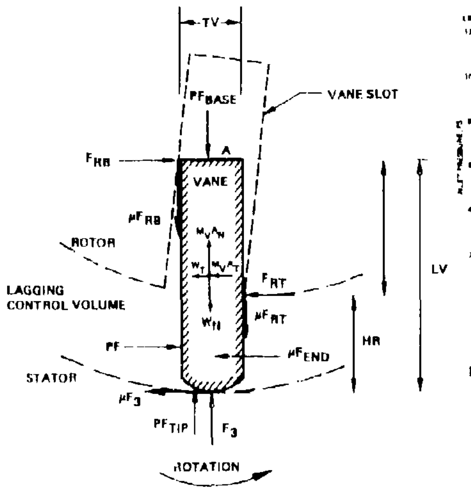

Figure 17. Free Body Diagram of Single Vane

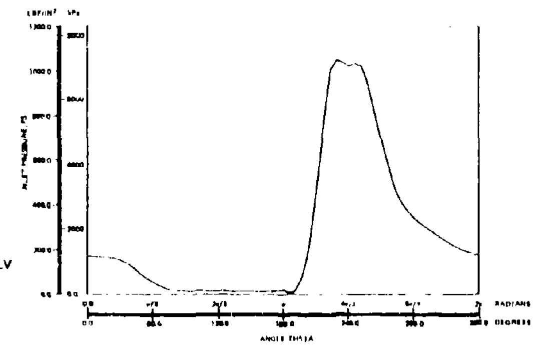

Figure 19. Pressure in Lagging Control Volume as a Function of Vane Angle

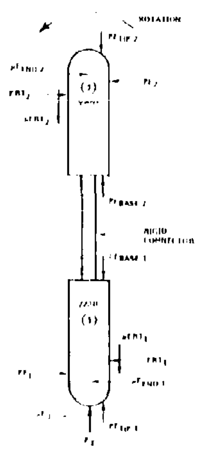

Figure 18. Model of Linked-Rigid Vanes

Table 4. Friction Power and Fluid Power per Vane for Various Conditions

Linked-Rigid Vane Model

Coeff. of Friction	Vane Weight (lbf)	Speed RPM	Pressure psia	Ported to Control Volume	Fluid Power/Vane (hp)	Friction Power/Vane (hp)	F_3 (θ)* lbf (°)	F_3 (θ)* N (rad)
0.2	0.18	4,000	700	lagging	6.49	2.82	308 (11)	1,370 (0.19)
0.4	0.18	4,000	700	lagging	6.49	5.51	341 (6)	1,520 (0.10)
0.2	0.18	4,000	1,115	lagging	10.51	4.17	451 (17)	2,010 (0.30)
0.4	0.18	4,000	1,115	lagging	10.51	8.07	492 (6)	2,190 (0.10)
0.2	0.18	4,000	1,500	lagging	14.22	5.42	584 (17)	2,600 (0.30)
0.4	0.18	4,000	1,500	lagging	14.22	10.43	637 (11)	2,830 (0.20)
0.2	0.18	2,500	1,115	lagging	6.57	2.41	404 (17)	1,800 (0.30)
0.2	0.18	6,000	1,115	lagging	15.76	7.29	548 (17)	2,440 (0.30)
0.2	0.06	4,000	1,115	lagging	10.51	1.83	399 (17)	1,770 (0.30)
0.4	0.06	4,000	1,115	lagging	10.51	3.31	433 (11)	1,925 (0.20)
0.2	0.06	2,500	1,115	lagging	6.57	2.32	384 (17)	1,710 (0.30)
0.2	0.06	6,000	1,115	lagging	15.76	6.06	432 (17)	1,920 (0.30)
0.2	0.06	4,000	1,115	leading	10.51	[illegible]	29 (17)	130 (0.30)
0.4	0.06	4,000	1,115	leading	10.51	[illegible]	69 (17)	310 (0.30)
0.2	0.18	4,000	1,115	leading	10.51	2.38	80 (17)	360 (0.30)
0.4	0.18	4,000	1,115	leading	10.51	5.88	0	0 (0)
0.2	0.18	2,500	1,115	leading	6.57	1.40	33 (17)	150 (0.30)
0.2	0.18	6,000	1,115	leading	15.76	4.42	177 (17)	790 (0.30)
				Pushrod Model				
0.2	0.18	4,000	1,115	lagging	10.51	6.45	533 (6)	2,370 (0.10)
0.4	0.18	4,000	1,115	lagging	10.51	11.36	740 (6)	3,290 (0.10)
0.2	0.06	4,000	1,115	lagging	10.51	4.74	477 (6)	[illegible] (0.10)
0.4	0.06	4,000	1,115	lagging	10.51	9.33	658 (6)	[illegible] (0.10)
0.2	0.18	4,000	1,115	leading	10.51	5.65	216 (22)	[illegible]
0.4	0.18	4,000	1,115	leading	10.51	11.53	178 (6)	[illegible]

*Peak value of F_3 and the angle ( ) at which it occurs in the vicinity of a fully extended vane (θ = 0)

Figures 20 and 21 summarize vane tip and end forces (F_3 and F_{END} in Figure 17) versus rotation angle for pushrod and linked vane configurations. As noted, the effect of linking is to reduce the peak normal force by 33.5 percent. Also, as noted in Table 4, linking of opposing blades results in a 40-percent reduction in friction horsepower. The effect of leading edge venting is shown in Figure 22 for the linked vane configuration. Of major importance is that the normal force on the vane is zero at maximum vane extension and achieves a peak of only 250 lbf (1,110 N) nearly 75 degrees (1.3 rad) later when the vane extension from the slot is substantially reduced. Also, friction power is reduced by 27 percent over that for trailing edge venting.

Figure 23 presents friction power per vane versus friction coefficient for cases run at 4,000 rpm (419 rad/s). The figure indicates the effect of vane mass, venting location (leading or trailing edge), and linking of the vanes.

Major results of the vane friction analysis study are:

1. Venting of the vanes to the leading edge and low mass vanes result in the lowest friction and vane forces.

2. Linking of the vanes further reduces frictional loads over those obtained with spring-loaded pushrods.

One word of caution in interpreting the results of the friction analysis is that the analysis does not simultaneously evaluate changes in leakage which may occur. Thus, a change that reduced friction may increase leakage and degrade overall motor performance.

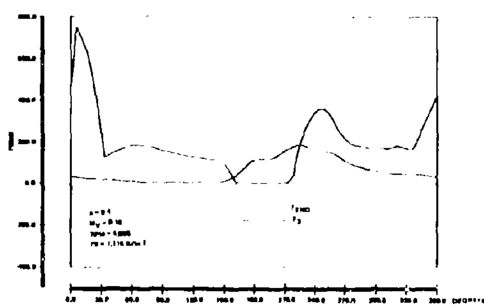

Figure 20. Tip and End Forces Pushrod Model Vented to Lagging Control Volume

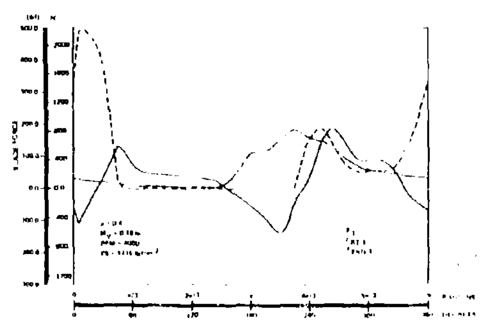

Figure 21. Vane Contact Forces Linked Rigid Model Vented to Lagging Control Volume

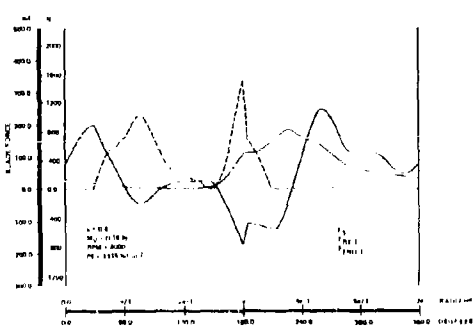

Figure 22. Vane Contact Forces for Linked Rigid Model Vented to Leading Control Volume

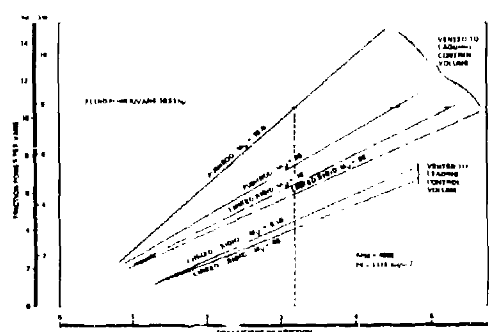

Figure 23. Friction Power per Vane Versus Coefficient of Friction

MODEL CONFIRMATION TESTS

Additional cold gas tests were conducted on the motor to experimentally verify the results of the friction analysis. A set of flame spray coated Inconel 600 vanes were subjected to machining to remove hollow cores from the base end of the vanes. The vane weight was reduced 39 percent from 0.18 lbm (82 g) to 0.11 lbm (50 g). The vanes were installed in the motor vented to the leading edge and with spring loaded pushrods. This configuration achieved time to 4,870 rpm (510 rad/s) of 6.3 seconds and was still accelerating. Previous testing with this vane with trailing edge venting and at 0.18 lbm (82 g) weight per vane indicated inability to obtain speeds above 4,000 rpm (419 rad/s). Referring to Figure 24, this implies a friction coefficient of 0.314.

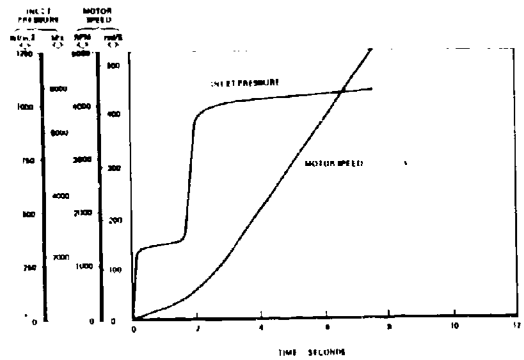

Figure 24. Ambient Temperatures Hot Gas Motor Start With Small Flywheel

Initial testing of a set of flame spray-coated Inconel 600 vanes in a linked vane configuration was conducted with venting to the trailing edge. Due to eccentricity, clearances at the vane tip varied from 0.003 to 0.030 inches (0.08 to 0.76 mm) over half a revolution. With this configuration a maximum speed of 3,000 rpm (314 rd/s) was achieved in 15.5 seconds. To reduce vane tip clearances, a special stator was fabricated. A second bore (offset to the inlet side) was cut into the stator with a radius and location of the center optimized for minimum clearances which were reduced from a maximum of 0.030 in. (0.76 mm) to 0.004 in. (0.10 mm). The linked-vane assembly could not be reversed in the stator to produce the more optimum venting to the leading edge and was thus run as previously. With this configuration, time to reach a speed of 5,000 rpm was 9.7 seconds.

FINAL MOTOR CONFIGURATION

Results of the friction analysis and model confirmation tests were analyzed to select a motor configuration for final demonstration tests. Results of lightweight vanes and venting to the leading edge demonstrated significant improvement in motor performance. Further, the analysis indicated that the forces are reduced significantly on the vane tip area at maximum vane extension when venting to the leading edge is employed. The lower forces led to reconsideration of using the carbon/carbon composite vanes since they are light in weight (0.06 lbm) and have a low coefficient of friction (less than one-half of any of the metal vanes tested). The final configuration selected for confirmation testing consisted of:

1. Carbon/carbon composite vanes with venting to the leading edge

2. Use of the oval stator, which allowed use of solid pushrods

3. Use of carbon seals at each end of the rotor as shown in Figure 15.

FINAL HOT GAS CONFIRMATION TESTS

With the above design modifications incorporated into the motor, final hot gas testing was conducted to demonstrate ambient temperature starting, $-65^{\circ}F$ ($-54^{\circ}C$) starting, and hot restarting following an initial start sequence. Excellent operation of the motor was verified at each of the above conditions.

Figure 24 depicts motor operation starting from ambient temperature using the small flywheel with an inertia of 27.7 lbf-ft^2 (1.17 kg-m^2). Figure 25 presents a test starting from $-65^{\circ}F$ ($-54^{\circ}C$) ambient temperature conditions using the large flywheel with an inertia of 41.2 lbf-ft^2 (1.74 kg-m^2). A hot restart using the large flywheel is shown in Figure 26. This test was made immediately following an ambient temperature start to full speed with the large flywheel.

As noted in Figures 24, 25, and 26, a pressure ramp start is used in which the inlet pressure is approximately 350 psia for 1.5 seconds and then ramped up to 1,000 psia for the remainder of the test. This procedure is employed to minimize engagement forces during startup and may or may not be required depending upon individual applications. Gas temperature inlet to the motor was approximately $1,600^{\circ}F$ ($871^{\circ}C$) for all three tests shown.

As noted in the figures, the time to accelerate to speed (including ramped start effects) is approximately 7 seconds with the small flywheel and 12 seconds with the large flywheel. These times are consistent with performance predictions for the motor which were previously discussed.

Based on the flywheel inertia and the speed-versus-time curve, the calculated torque output of the motor is approximately 70 ft-lbf (96.8 kg-m) up to the cutoff speed of 5,000 rpm (524 rad/s). This output is consistent with the desired output of Figure 2.

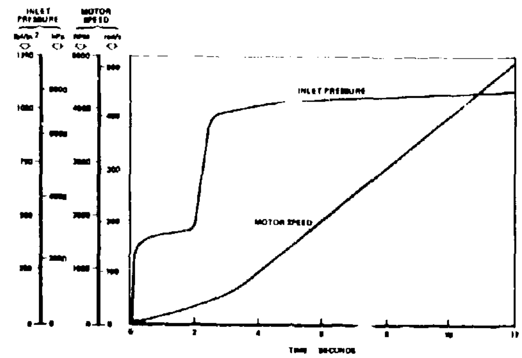

Figure 25. -65°F (-54°C) Hot Gas Motor Start With Large Flywheel

SUMMARY AND CONCLUSIONS

Development testing has been successfully conducted to develop design criteria and successfully demonstrate the feasibility of a hot gas rotary vane motor for providing aircraft APU starting over a -65° to +130°F (-54° to +54°C) temperature range. The hot gas rotary vane motor overcomes the temperature limitations of current systems and provides multiple start capability.

Initial testing of the original motor design with cold gas nitrogen and hot gas hydrazine-based decomposition products uncovered problems with the cam actuation system, excessive leakage, sensitivity to clearances, and vane breakage. Analytical studies verified by engineering tests demonstrated design solutions to the problem areas. Revisions to the motor configuration to incorporate pushrod actuation, revised end seals to reduce leakage, and venting of the vanes to the leading edge all demonstrated significant improvement in motor operation and resulted in successful operation over the required range of environmental conditions. Design criteria successfully developed under this program may be effectively used in the design of hot gas rotary vane motors for specific applications.

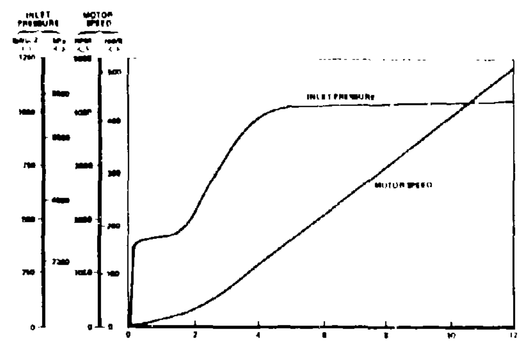

Figure 26. Hot Restart on Hot Gas Motor With Large Flywheel

ACKNOWLEDGEMENT

This work is sponsored by the Air Force Aeropropulsion Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, under Contract F33615-76-C-2148. The authors wish to acknowledge the guidance and contributions of Dr. B. L. McFadden (AFAPL/POP) who has served as the WPAFB project manager directing the contract work.

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DISCUSSION

H.Wittenberg, Ne

You have described a feasibility study – including experiments – about the hot gas APU starter system. Can you give some indication about the time required to develop the system ready for operational use in the field?

Author's Reply

The program described demonstrated the capability of a hot gas rotary vane motor to provide the starting capability of advanced APUs for military aircraft. Rotary vane motor design criteria was developed which will allow application of the motor to specific aircraft application. Based on the current development status it is estimated that a time period of approximately 18 months would be required to develop a motor for specific application and to qualify it for operational status.

W.Hoose, Ge

Could you give an estimated figure of the total weight for such a system, dependent on torque of the applied APU?

Author's Reply

A rough weight of a system to provide 3 starts is 16 lbs. This weight is for a system sized to produce the torque output shown in the paper. The weight includes all tankage, valving, fuel and the rotary vane motor.

Ph. Ramette, Fr

Parmi les différents matériaux que vous avez examinés pour les aubes, vous avez indiqué le composite carbone-carbone infiltré de carbure de silicium introduit?

Author's Reply

The exact amount of silicon carbide material deposited on the carbon/carbon composite is not known. I would estimate the amount at approximately 30% by weight.

A JET FUEL STARTER FOR LOWEST SYSTEM LIFE CYCLE COST

by
Ray Smith
Director Advanced Development
& Technical Planning
TELEDYNE CAE
1330 Laskey Rd
P.O. Box 6971
Toledo, Ohio 43612

Abstract

A modular Jet Fuel Starter design ~~employing~~ using an Expendable Gasifier (EG) ~~concept~~ indicates a substantial life cycle cost savings over a conventional overhauled unit. To achieve these savings, a low cost gasifier is a mandatory requirement. The EG does this by maximizing cast-to-near-net-shape components, using aluminum cold end castings and by minimizing the use of expensive high temperature alloys. To allow these design approaches, gasifier performance parameters, specific power, compressor pressure ratio, cycle temperature and fuel consumption, ~~have been~~ were selected to minimize component stress and sensitivity to refined dimensions.

To further enhance future downstream cost savings, the gasifier interfaces have been established to fit optional JFS and auxilarily power unit installations, and to include turbojets and turbofans suitable for unmanned vehicles.

Performance levels and initial structural integrity have been validated by both component and engine testing of a turbojet demonstrator, using hardware fabricated by production manufacturing technology methods.

Introduction

The on-board Jet Fuel Starter (JFS) is an aircraft secondary power system which generally has a low priority and receives near-last development consideration. However, its life cycle cost has significant fleet system impact. The United States Air Force Aero Propulsion Laboratory, Aerospace Power and Propulsion Division, has ongoing programs to evaluate future requirements, with a strong emphasis on reducing life cycle cost. The Expendable Gasifier concept is an outgrowth of these programs; the gasifier of the JFS is discarded at the end of its useful (2000-start) life.

The Teledyne CAE Model 206 Jet Fuel Starter (JFS), Figure 1, is designed to start large engines in the 11,000 daN thrust class. This small, light-weight, 170 kw starter can be mounted either directly on an engine gearbox or to a remote airframe-mounted gearbox connected to the main engine through a power take-off shaft. The design driver has been low system life cycle cost, achieved by reducing development, acquisition, logistics and maintenance costs. (JFS fuel consumption does not have a measurable impact on life cycle cost, since the nominal operating time to start a large engine is only 30 seconds).

Installation and Performance

The unit is 35.6 cm wide, 48.0 cm high, 76.2 cm long and weighs 68.1 kg; it is rated at 170 kw and an SFC of 878 gm/hr/kw. The subsystems are functionally integrated to automatically start the main engine, and include an electrical interface for cockpit instrumentation, command and control.

Once the start sequence has been initiated via hydraulic motor power, the JFS requires only a fuel supply: its alternator provides all the electrical power needed for operation including termination of the start sequence. In the event electrical power is lost, the starter will automatically terminate the start sequence and close the fuel supply. The pilot (operator) may also command a starter abort or a motoring function. All shutdown commands are reset automatically so that no overt actions are required prior to initiating another start.

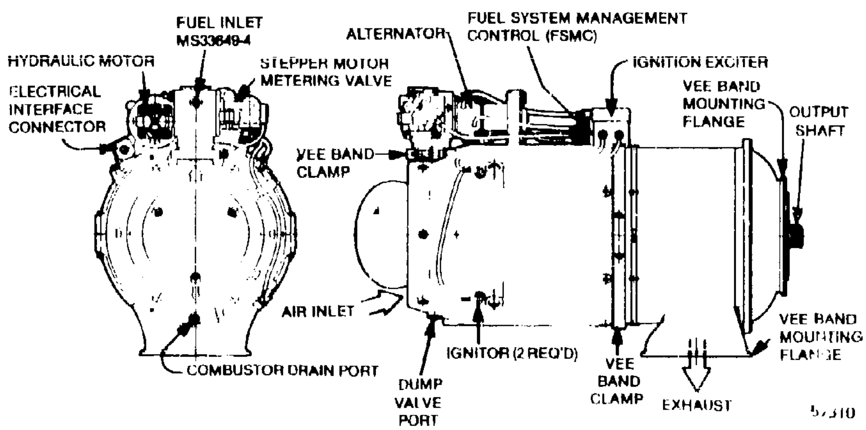

FIGURE 1 MODEL JFS 206 JET FUEL STARTER INSTALLATION.

Output power and torque characteristics are shown in Figure 2 as a function of JFS output speed. The shaded band indicates the starter design cutout speed range. Typical main engine starting requirements are compared to the JFS torque output characteristics for sea level static (SLS) condition at 59°F and 125°F in Figures 3 and 4, referenced to the higher main engine power take-off (PTO) shaft speed. JFS and typical turbofan main engine acceleration transients are shown in Figure 5 as percentage of shaft speed vs. time for a SLS 52°C day. The gasifier accelerates to full speed in less than 10 seconds after start command and holds at full speed until the cut-off of the JFS operation at approximately 29 seconds, whereupon the main engine continues to accelerate under its own power to idle at 40 seconds.

Optional application options include:

- o Direct drive: the reduction gear and overrunning clutch are eliminated, without change to the free power turbine or its suspension. The unit is integrated with either an airframe- or an engine-mounted gearbox.
- o Direct drive, front PTO: the power turbine output shaft drives through the hollow EG shaft.
- o Aircraft systems ground check-out: combines main engine starting with motoring capability to drive the aircraft generator and hydraulic pumps. Addition of a small auxiliary compressor provides air system check-out.
- o Primary propulsion turbojet and turbofan derivatives for unmanned vehicles (described in Reference 1).

Multi-application use of the EG would reduce acquisition costs by increasing production rates, in accordance with the initial program objectives.

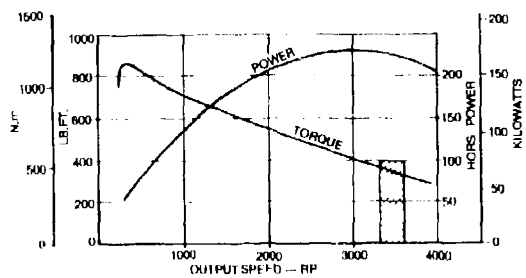

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FIGURE 2 JFS — TORQUE AND HORSEPOWER CHARACTERISTICS.

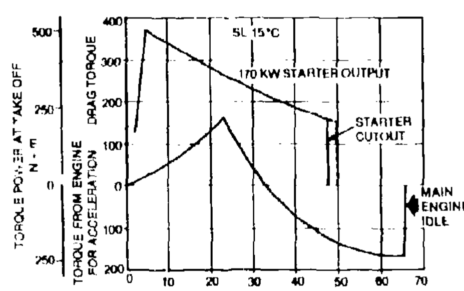

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FIGURE 3 TYPICAL ENGINE/STARTER CHARACTERISTICS — SL:59°F.

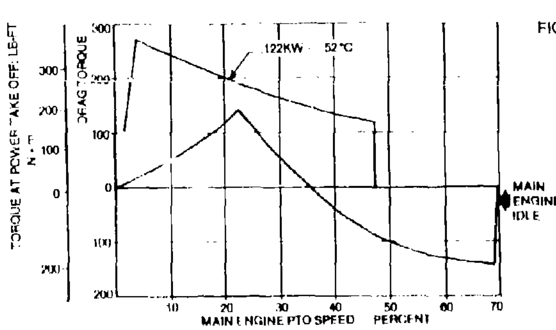

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FIGURE 4 TYPICAL MAIN ENGINE/JFS STARTER CHARACTERISTICS — SL:52°C.

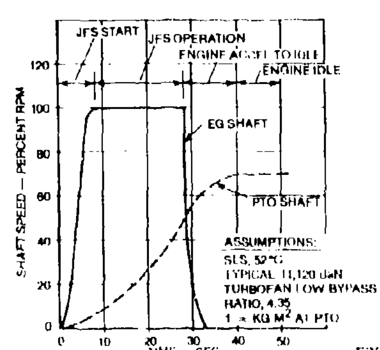

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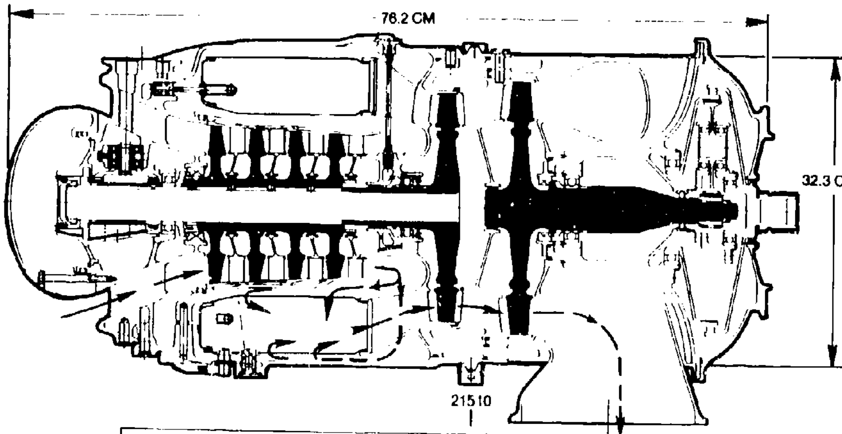
FIGURE 5 ESTIMATE OF JFS TRANSIENT DURING STARTING OF TYPICAL TURBOFAN ENGINE.

Description

The EG concept is an outgrowth of ongoing Air Force Aero Propulsion Laboratory development programs to reduce the cost of aircraft power systems (Reference 2). The design is driven by a requirement for low acquisition cost, wherein it is cost-effective to discard the EG at the end of its useful life, rather than overhaul the unit. The design life to satisfy this criterion has been established as 2,000 starts as a jet fuel starter, or 15 hours of maximum power operation as a propulsion engine.

The jet fuel starter (JFS) (Figure 6) consists of three modules; the expendable gasifier, the power output unit and the system management control (SMC). The EG consists of a four-stage axial compressor, a reverse flow annular combustor, and a single stage axial turbine. The power output unit includes a single stage axial flow turbine, reduction gear, overrunning clutch and the power output shaft. The System Management Control (SMC) consists of the fuel pump, the EG starter motor, an alternator and the control unit.

Air entering the engine passes through the compressor and bifurcates into two channels. Over 50% of the air passes through the hollow turbine inlet nozzle vanes and flows along the outer wall of the combustor liner; the balance flows along the inner wall of the combustor liner. The gas exiting from the combustor passes through the EG turbine, the power output turbine and then is ducted by a scroll to a single radial outlet. The folded combustor/compressor arrangement was chosen to minimize installation axial length.



PERFORMANCE (SLS, 59F)			
POWER	172 KW	CPR	2.86
SFC	878G M/HR/KW	WEIGHT	68 KG
AIRFLOW	1.75 KG/SEC	POWER/WEIGHT	2.53 KW/KG
TIT	983 °C	N_{PTO}	3600 RPM

FIGURE 6 MODEL JFS 206 JET FUEL STARTER.

The integrally bladed axial compressor rotors (Figure 7) are C355 aluminum, cast to size; only the mounting surfaces and rotor tips are machined. The four rotors are all identical castings and only the tip diameters are machined differently to provide the proper compressor flowpath convergence. A low compressor tip speed (900 ft/sec) was selected to permit the use of low cost cast aluminum rotors. These rotors are attached to the shaft by three radial pins to eliminate splines and close tolerance pilot diameters.

The compressor stators (Figure 8 illustrates the first stage) are also precision aluminum castings, with only the mounting and other surfaces required for close clearance control being machined after casting.

The combustor (Figure 9) is a multipiece fabrication consisting of sheet metal inner and outer shells, a base plate, a forward support flange and 14 cast vaporizing tubes. It is supported at the forward end by rivets attaching to the outer flange of the first stage compressor stator and to the aft end by a slip-joint connection to the turbine inlet nozzle. The combustor liner is cooled by the means of closely-spaced small holes, pre-perforated in the flat stock sheet metal, eliminating the need for expensive cooling bands. The combustor sheet metal is Haynes 556 (cobalt base) alloy and the vaporizer tubes are cast IN 155 (iron base) alloy.

The turbine inlet nozzle (Figure 10) is a monolithic investment casting of N155 (iron base) alloy, selected for its good castability, weldability, oxidation and strength characteristics up to 1800°F. The hollow vanes provide regenerative vane cooling to reduce the deleterious effects of combustor hot streaks. The turbine inlet nozzle is also attached to the main frame by means of radial pins, and provides the support frame for the rear bearing and the turbine shroud.

The gasifier turbine rotor is investment cast from IN100 (nickel base) alloy and electron beam welded to the 17-4PH (iron base) alloy shaft, as shown in Figure 11.

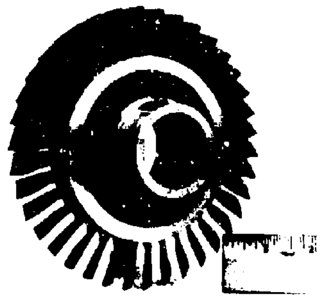


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FIGURE 7 CAST COMPRESSOR ROTOR. 57316

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FIGURE 8 CAST INTEGRAL FIRST STATOR. 57317

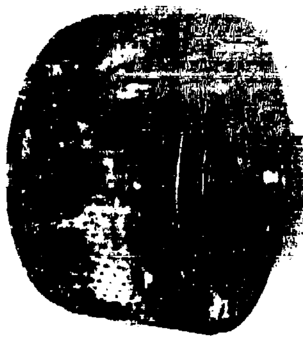

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FIGURE 9 COMBUSTOR ASSEMBLY. 57319

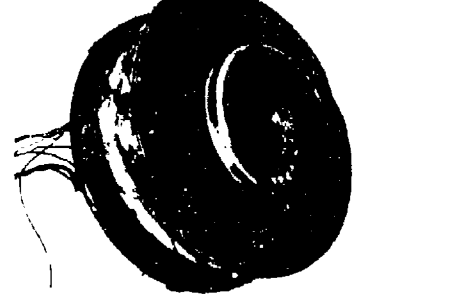

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FIGURE 10 CAST TURBINE INLET NOZZLE. 57359

The main frame (Figure 12) is an aluminum sand casting and contains fuel passages for primer and main fuel nozzles, as well as for rear bearing lubrication. The shaft assembly (Figure 6) is supported in two bearings: the front thrust bearing is located in the main frame center body, and the rear roller bearing is supported through the turbine inlet nozzle.

The front bearings and gear set are lubricated by a proven self-contained sump lube system originally developed for the Teledyne CAE J402 engine (Reference 3). The nose cone is injection molded from translucent plastic to provide a ready visual indication of the oil level in the front bearing compartment. The bearing temperature is stabilized by inlet air flowing over the center body. The rear bearing is fuel lubricated by a system also developed and proven on the J402 (Reference 4). The compartment temperature is stabilized by compressor leakage air flowing through the bearing compartment and then up the back face of the turbine rotor.

Detail design, fabrication, component testing, and demonstration gasifier testing have been accomplished to date (Reference 2). Preliminary design of the power output module and SMC has been conducted to identify weights, volumes, control operation and safety parameters of the JFS.

Life Cycle Cost (LCC)

The primary objective of the JFS design has been to reduce secondary power system life cycle cost. The major contributor to JFS and APU life cycle cost is the gasifier, which generally requires an overhaul four times as often as the power output module or the systems management control.

To address the objective, the JFS is of modular construction (Figure 13), consisting of the expendable gasifier, power output module and the system management control module (SMC). The three modules are assembled by means of quick assembly and disconnect (QAD) rings, and the assembled JFS is mounted by a QAD ring. The modular design permits quick replacement of the EG on-board the aircraft.

The life cycle cost factors and the design approach to reducing these factors are summarized in Table 1. Table 2 compares the maintenance actions required to remove and replace the EG vs. a conventional JFS. Replacement of the EG requires five functional maintenance operations, as opposed to seven for the JFS, a reduction of approximately 30% in both maintenance manhours and turnaround time.

Service data show that the major LCC element for a JFS or auxiliary power unit (APU) is overhaul cost, as shown in Figure 14; Life Cycle Costs were analyzed for a 10 year weapon system life for a fleet of 1000 aircraft. Overhaul costs were calculated by determining replacement costs of the EG module of the JFS after every 2000 starts, as produced at a rate of 500 units per year for five years; actual user overhaul rates were applied for the current operational JFS unit. The analysis showed that the EG concept incorporated in an APU will save $250 million; in a JFS, LCC savings would be $270 million over a conventional system.

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FIGURE 11 CAST TURBINE ROTOR EB WELDED TO SHAFT.

FIGURE 12 CAST MAIN FRAME. 23772

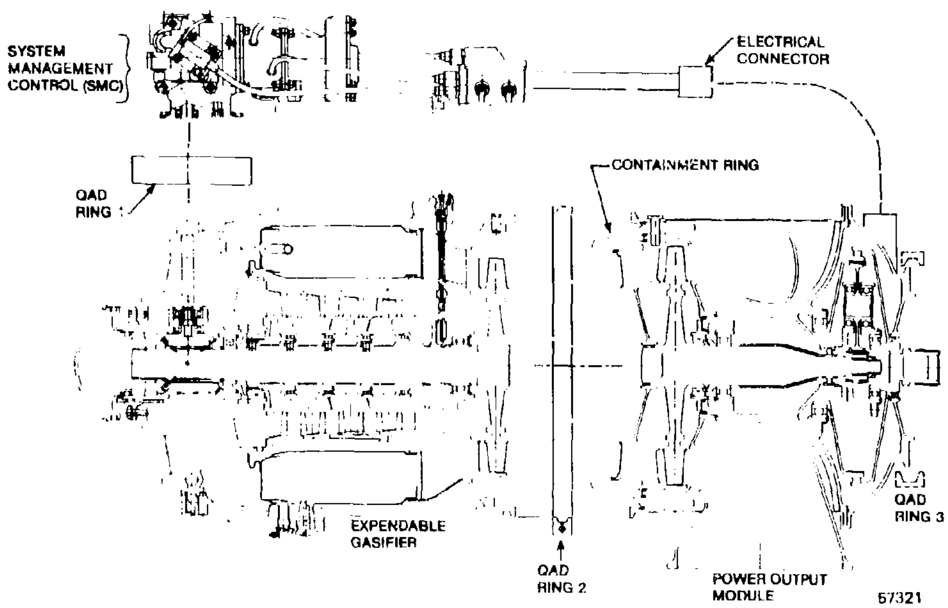


FIGURE 13 JET FUEL STARTER MODULAR CONSTRUCTION.

TABLE 1 DESIGN APPROACH FOR REDUCING JET FUEL STARTER LCC

LCC FACTOR	DESIGN APPROACH
Development Cost	Conservative performance and structural design using simple features, proven in Advanced Development
Acquisition Cost	Maximize the use of cast components and minimize subsystem complexity: — machining timing/tooling — capital investment — inspection problems/manhours — parts count — assembly labor
Logistics Cost	Reduce unit cost by increased rate production — multi-application capability. Minimize the number of parts in inventory (gasifier is expendable), thus reduce: — logistics management — tech orders — tooling
Maintenance Cost	Reduce the number of parts to be serviced. Minimize turnaround time. Maximize service on aircraft, and reduce: — depot tooling and facilities capital — overhaul engineering support — line technician skill and training requirments

TABLE 2 — COMPARISON OF MAINTENANCE ACTIONS TO REMOVE AND REPLACE THE EG VS. JFS.

MAINTENANCE FUNCTION	OPERATIONS TO REMOVE AND REPLACE	
	EG	JFS
Access Cover	Yes	Yes
Air Inlet Duct	Yes	Yes
Exhaust Duct	No	Yes
Fuel Line	No	Yes
Hydraulic Lines	No	Yes
Ignitor Leads	Yes	No
Electrical Interface	No	Yes
QAD Ring 1	Yes	No
QAD Ring 2	Yes	No
QAD Ring 3	No	Yes
TOTAL	5	7

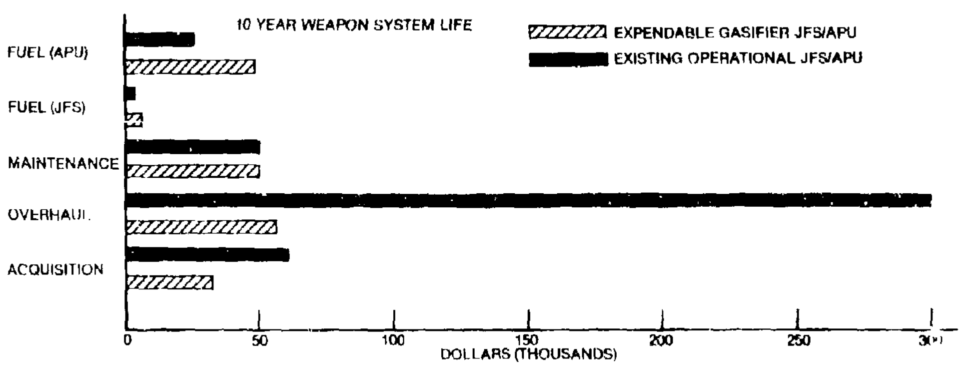
Program Test Results

The EG has completed a four-phase program as follows:

Phase 1 System Design
Phase 2 Preliminary [illegible] Design
Phase 3 Detail Design and Combustor Test
Phase 4 Fabrication and Testing of the Compressor and Gasifier (as a Turbojet)

During phases 3 and 4 the total test time accumulated was:

	BUILDS	TEST TIME (HRS)
Compressor Rig	1	23
Combustor Rig	6	33
Turbojet Engine	2	13



EG CONCEPT SAVINGS FOR A FLEET OF 1000 AIRCRAFT
$270 MILLION AS JFS
$250 MILLION AS APU

40473

FIGURE 14 EXPENDABLE GASIFIER LIFE CYCLE COST COMPARISON IN FY 1980 $.

Compressor rig testing showed the success of the common rotor blading approach: adequate performance (although slightly less than the design goal) was achieved in the first test, Figure 15. In the 23 hours of rig testing, no abnormal mechanical or aerodynamic conditions were experienced. Data analysis revealed that the performance deficiency was related to low predictions of rotor exit aerodynamic blockage for the high diffusion factor design. A direction for design modification and performance improvement was thus established; however, the achieved performance was deemed adequate for gasifier demonstration.

The combustor rig tests were also successful. In addition to validating efficiency, temperature rise and outlet pattern factor, a primary objective was to measure liner temperatures on the full-coverage, film-cooled design. Premature failure in low cycle fatigue or oxidation could occur in the intended 2000-start JFS if temperature or thermal gradient limits were exceeded. Liner hole modifications were made during the six builds to achieve the desired characteristics, with results as summarized in Table 3. All objectives were achieved or exceeded, except for pressure loss, which was 7 percent higher than the 5 percent design value. Initial test values were 16 percent, attributed to excessive losses in the 180° turn at the compressor discharge, coupled with the flow split at the hollow nozzle vanes. The vane entrance was modified by rounding, and the diffuser was cut back, reducing the system pressure drop to 12.3 percent, the limit which could be attained without redesign and new hardware. (Subsequent engine testing indicated a 10.1 percent pressure loss in the gasifier environment).

TABLE 3: COMBUSTOR RIG PERFORMANCE CHARACTERISTICS

	Design Goal	Rig Test Results
Pressure Loss — Percent	5.0	12.3
Efficiency — Percent	95.0	95.0
Exit Temperature — °K (°F)	1255 (1800)	1255 (1800)
Temperature Rise — °K (°F)	1097 (1516)	1097 (1516)
Exit Radial Temperature Profile Factor	0.07	0.05
Exit Circumferential Profile Factor	0.20	0.154
Maximum Liner Temperature — °K (°F)	1033 (1400)	922 (1200)

During the combustion tests, turbine nozzle flow capacity and losses were also calibrated and found to be very close to design.

These component performance characteristics were combined in a computer prediction of gasifier performance as a turbojet demonstrator. They showed that design power output could be achieved by rematching to a higher rotor speed (107 percent of design).

The expendable gasifier is shown configured as a turbojet demonstrator in Figure 16. The engine was tested with two different compressors, first with machined rotors and then with cast aluminum rotors over a range of altitudes and operating lines (jet nozzle areas). The engine performance with the cast

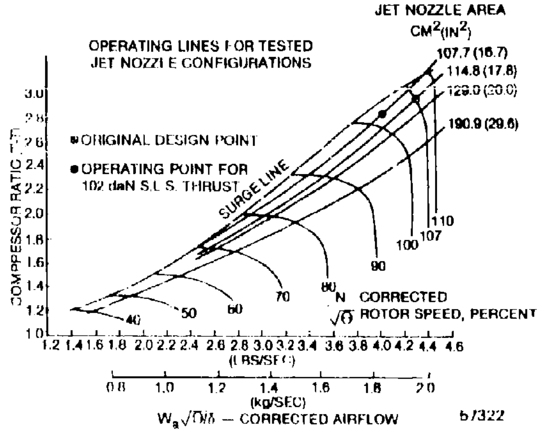


FIGURE 15 TEST COMPRESSOR MAP WITH TURBOJET OPERATING LINES SUPERIMPOSED.

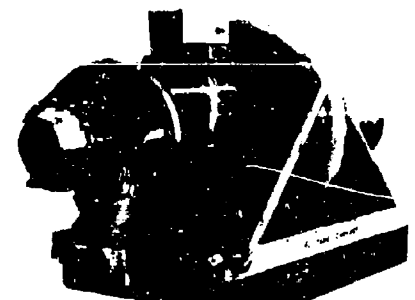


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FIGURE 16 EXPENDABLE GASIFIER CONFIGURED AS A TURBOJET DEMONSTRATOR, READY FOR TEST compressor rotors is shown in Figure 17 compared to predictions - good agreement is evident. The demonstrated sea level static thrust in this configuration was 82.3 daN. When tested with machined steel rotors, the engine demonstrated 102.3 daN thrust at 971°C turbine entry temperature, some 12°C below the design value.

Comparisons of cast and machined rotor performance, in terms of sea level static specific fuel consumption vs. thrust are shown in Figure 18 and turbine inlet temperature vs. rpm in Figure 19. The lower performance of the configuration with the cast compressor rotors results from the reduced efficiency and air flow shown in Figure 20. Inspection of the cast compressor rotor showed deviant blade angle settings and blade spacing; since the rotors were structurally sound (as determined by spin test to destruction), they were tested to demonstrate their structural integrity and to quantify the performance degradation. To attenuate the problem, the compressor rotor casting tooling was modified to provide a stabilizing ring on the outer diameter (Figure 21) in a manner similiar to the cast stators. This provides rigidity for mold removal and serves as a fixture to hold the blades during heat treatment. Since all rotors are machined at the tip diameter, machining off the cast ring has only a minimal impact on cost.

Engine data analysis showed that the compressor (Figure 15) and combustor performance closely approximated rig values; combustor pressure loss was actually lower in the engine, and turbine efficiency exceeded design objectives by over 1.5 percent.

A start envelope was established during testing, using manual fuel scheduling, a hydraulic starter and windmill; successful starts were accomplished over the range of 0 to 6561m and at ram air Mach numbers from 0 to 0.5. It was judged that improved start scheduling, as might be expected from a fuel control development, would be required to enlarge this range.

Engine steady state operation over the altitude range was successful, with the only limit being a rear bearing temperature rise which precluded sustained operation at the 107 percent overspeed condition. Since this speed was an artifact of the limited advanced development program scope, further lube system refinement and component development to reduce maximum speeds would be expected to alleviate the problem.

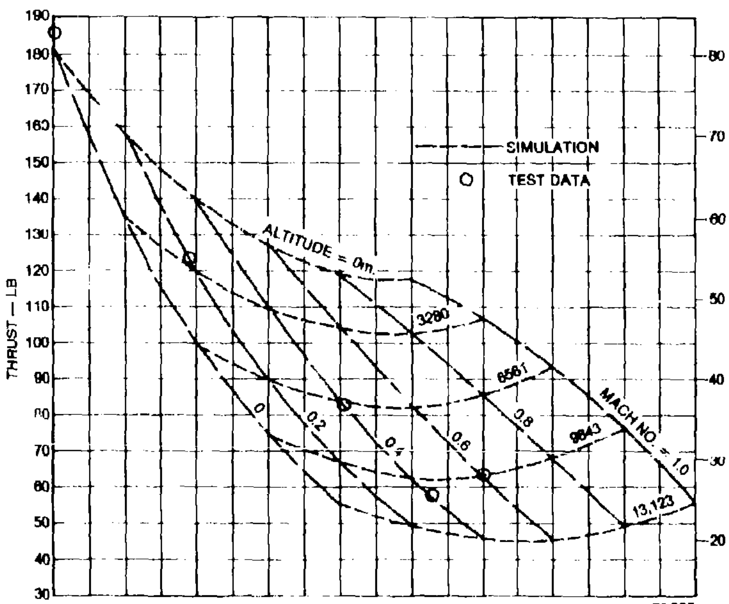

FIGURE 17 EXPENDABLE GASIFIER ALTITUDE PERFORMANCE @ 100% SPEED

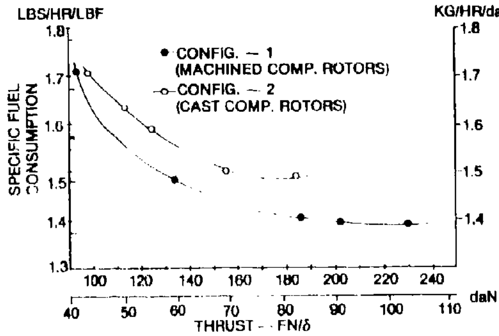

FIGURE 18 EXPENDABLE GASIFIER TURBOJET ENGINE SPECIFIC FUEL CONSUMPTION VERSUS THRUST COMPARISON FOR BOTH CONFIGURATIONS.

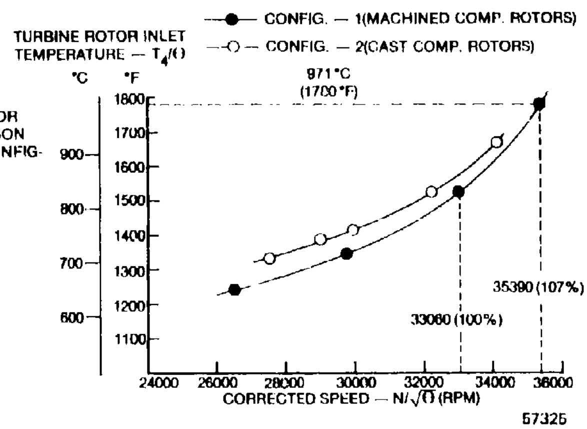

FIGURE 19 EXPENDABLE GASIFIER TURBINE ROTOR INLET TEMPERATURE (TRIT) COMPARISON FOR MACHINED AND CAST ROTOR CONFIGURATIONS.

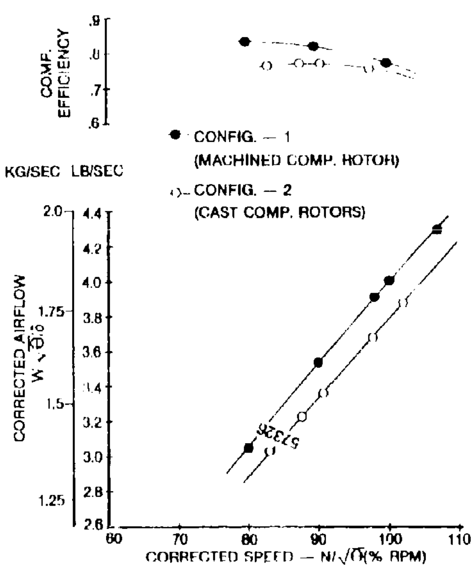

FIGURE 20 EXPENDABLE GASIFIER COMPRESSOR EFFICIENCY AND AIRFLOW VERSUS ROTOR SPEED FOR TWO CONFIGURATIONS.

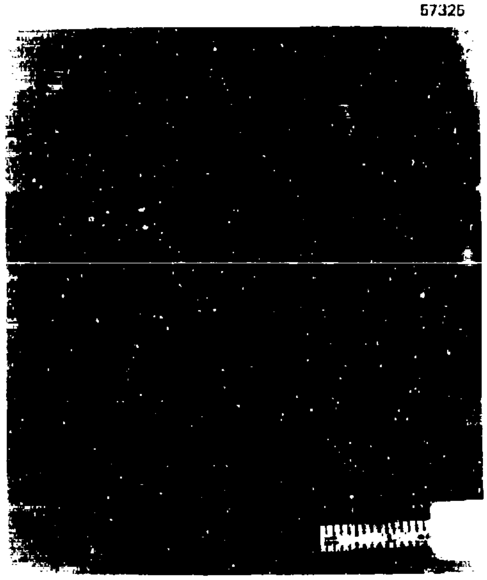

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FIGURE 21 COMPRESSOR ROTOR CASTING WITH RING CAST ON OUTSIDE DIAMETER.

CONCLUSIONS:

Life cycle cost analysis indicates that substantial savings can be realized by using the Expendable Gasifier (EG) concept to power either a jet fuel starter (JFS) or an auxiliary power unit (APU): the unit is discarded at the completion of its useful life - not overhauled.

Modest performance levels (pressure ratio, temperature, specific power, SFC volume and weight) are a necessary consequence of the EG concept: incorporation of minimum production cost features, such as cast-to-near-net-shape components, self-contained lubrication systems, and low cost raw materials requires relaxation of tip speeds, stage loadings and critical surface and clearance tolerances.

The potential EG payoffs extend beyond a single JFS/APU application, to include optional JFS configurations and propulsion units for unmanned vehicles using one or more common modules.

EG low cost production fabrication technology, performance and structural integrity have been demonstrated by component rig testing and gasifier testing. Sufficient analytical, fabrication and test data have been collected to warrant expanded development effort on a complete JFS, and to validate the system over an equivalent 2000 starts.

Secondary power systems are low priority, among the last to be considered in major weapon system programs. Advanced demonstrator programs, typified by the four phase EG effort, are necessary to quantify and demonstrate the technology needed for future weapon systems.

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4. Bird, C. R., Hagebush, C., "Flight Testing of Small Engines for Diversified Missions", AIAA Paper 75-999, Aug. 1975

DISCUSSION

H.I.H.Saravanamuttoo, Ca

The expendable gas generator is designed for a life of 2000 starts. Is the power turbine also limited to the same number of starts?

Author's Reply

No. A detailed design of the power turbine has not been completed, however it is expected that a life of 8,000 (or more) starts can be attained. The power turbine design problem statement is much easier than that of the gas generator due to its lower turbine inlet temperature.

A.L.Romanin, US

I Are the overhaul costs compared on a per start or direct comparable basis?

II Have the existing overhaul costs been itemized to the extent that you are comparing gas generator versus gas generator only?

III The work was quoted as government funded. What is the contract number?

IV Is this unit designed primarily as an expendable jet engine?

Author's Reply

I The overhaul costs are on a direct comparable basis. The comparison is on a per unit basis for a 10 year weapon system life and a fleet of 1,000 aircraft.

II The existing overhaul costs have been itemized to the extent that the gas generator cost per overhaul, is separated from the cost (per overhaul) of the power output and fuel system management control modules.

III The work was done under government contract number F33657-76-C-2055.

IV The unit was designed primarily for a jet fuel starter, therefor the intent was that the jet fuel starter requirements have precedence over those of an expendable jet engine.

COST EFFICIENT ON BOARD POWER FOR AIRLINE OPERATION

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Lufthansa German Airlines
2000 Hamburg 63, W. Germany

Summary This report provides a brief -

1. Introduction / technical review of airline experience with Auxiliary Power Units (APU) as influenced by the fuel crisis.

2. Typical APU design features and performance outputs, supplying bleed air and/or electrical power on board the airplane are described.

3. Economical impact on APU operation due to:
   - fuel cost increase and
   - high level of APU maintenance cost is discussed

   Airline's remedy action by: remedies as
   - incorporation of fuel saving modification programs and
   - reduction of APU operating time.

   Technical consequences and effects on operating procedures, alternative power sources are considered.

4. Utilization of alternative power sources on the ground, providing on board power, by means of mobile or stationary systems for electrical and pneumatic supply with respect to:
   - system compatibility with on board power requirements
   - availability and necessary cost investments.

5. Plans and practical approaches, aiming for optimized, cost efficient on board power, seen from an airline's point of view, are provided.

6. Conclusion / recommendations for future APU-operation.

1. Introduction: Technical review of airline experience with Auxiliary Power Units (APU), influenced by the fuel crisis

When in 1964 the first Boeing 727-030 aircraft were delivered with an on board Auxiliary Power Unit, known as the APU, being installed in the fuselage keelbeam and capable of supplying pneumatic and electrical AC power to the aircraft systems, this has become a real technical novelty to everyone in the airlines business. The technical advantage of the APU, mainly its practicability providing energy to the aircraft together with a very high degree of operational independency of airfields and ground facilities has been the reason, why the APU became such a useful equipment. During the subsequent years almost every new aircraft design in the commercial field has been certified and delivered with an APU, what would substantiate the usefulness of this modern aircraft technology.

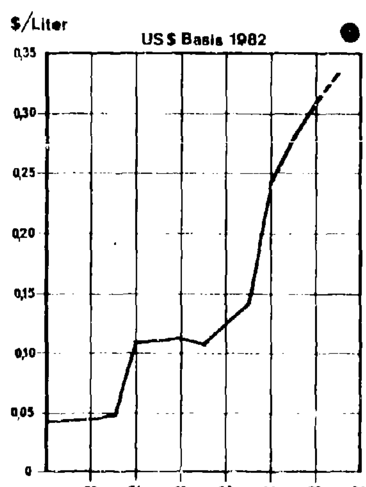


Fig. 1 FUEL COST INCREASE (per Year)

The technical APU features had encouraged the airlines philosophy to take full operational advantage of the APU as secondary power equipment on board the aircraft. Unfortunately, already some years later, this philosophy had to be changed and readjusted, due to fuel economical considerations.

When in 1973 the first fuel crisis came up, this has affected the economics of aviation industry. Latest in 1979, when the second but more severe fuel price increase had followed, the fuel cost efficiency of the equipment used in the airline business had to be evaluated. Particularily, commercial airlines, amongst them Lufthansa, suffering from this second fuel cost impact, have been forced now to review the economical aspect of operating main engines and APU's, aiming for optimized operational procedures, which would reduce the fuel consumption, but still meet the technical requirements of the system equipment and the power demands on board the aircraft.

Figure 1 is showing the kerosin fuel price increase during the past 10 years. At the moment we are facing a somewhat relaxed upward trend of fuel cost per liter, compared to 1979/80 with an annual escalation rate of approximately 11 percent (from 1979 to 1980 this figure had nearly doubled).

Lufthansa, like every other commercial airline has faced the burden and the economical challenge, which had been imposed upon the airline operation since 1979, with continuous efforts to search for a new economical operating concept of supplying cost efficient on board power to the fleets. As a first step, in cooperation with an airline fuel conservation group, an APU working group was established, consisting of system engineering, flight operations, ground support staff, and airline economy experts. This working group undertook the task of investigating following four major aspects:

- Determining the technical requirements of APU ground power supply for different fleets.
- Investigating alternative power supply systems on the ground, their availability and compatibility with aircraft electrical systems, pneumatics and air conditioning systems.
- Development of cost efficient procedures for ground power supply.
- Necessary investments for an airline to establish ground power systems on major airports.

2. Typical APU design features and performance outputs

Prior to the discussion of the study items and the results of the APU working group it appears necessary to give a brief survey of the APU's, including their design and performance, as being operated by todays commercial airlines like Lufthansa.

APU Performance Data

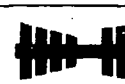
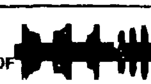
Aircraft Model Fleet:	APU Model	Equival. SHP	Bleed Air(Ws) Output lbs/min	El. load SHP/KVA	Fuel cons. Wf lbs/hr	SFC lbs.Fuel/hr SHP	Ratio Wf/Ws
Boeing 727-230 Boeing 737-230	GTCP 85-98CK GTCP 85-129B 	245 230	111 102	50/31 25/16	298	1,22	2,7
Boeing 747	GTCP 660-4 	822	507	63/40	945	1,15	1,9
Douglas DC 10 Airbus A 300	TSCP 700-4 TSCP 700-5 	705 570	385 308	142/90	505 430	0,72 0,75	1,3 1,4
Airbus A 310	GTCP 331-250F 	525	250	102/65	365	0,70	1,5

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Figure 2

Figure 2 presents a review of the various APU's used in the field. Let me commence with the APU model used in short to medium aircraft fleets. The first row of figure 2 shows a schematic of the APU, installed in the Boeing 737 and 727 aircraft. This 85-series APU is a single spool gas turbine with constant speed and integrated bleed air flow output. A two stages radial compressor is driven by s single stage radial turbine with an exducer to achieve the power requirements, providing bleed air (111 lbs/min.) and electrical power supply (400 Hz, 115V) by driving a 40 KVA generator. In its standard version this APU is very reliable.

The APU in the second row of the slide is the largest power unit on board our aircraft being installed in the Boeing 747. This is again a bleeder type APU with a single shaft design, consisting of a two stages axial turbine driving a four stages axial flow compressor. To provide sufficient surge margin during start up of the APU, an interstage bleed valve is being provided between the 2nd and 3rd compressor stage, in order to stabilize the compressor air flow in the acceleration regim. After having reached the full governing speed (20000 rpm) a modulating surge valve controls the bleed air flow required to maintain stabilized flow under all bleed load or shaft load conditions, preventing the APU compressor from a surge or a stall.

In the third row of the figure 2 a schematic of the APU 700 series is shown, being operated in the Douglas DC10 and in the Airbus A300.

This is a two spool integral bleeder type APU with variable inlet turbine nozzle guide vanes in front of the first stage low pressure turbine. The variable NGVs of the first stage low pressure turbine, controlled by a fuel actuated stator vane positioning device, determine the actual N1 APU spool speed as required by the load/power demand. The high pressure spool is kept constant regarding its speed and consists of a radial impeller being driven by a single stage axial flow turbine. The low pressure compressor being driven by a two stages turbine consists of 3 low pressure axial stages.

The APU shown in the last row of figure 2 is the most recent design concept of Garrett, being used in the Airbus A310, with a load compressor mechanically coupled to the power section but aerodynamically kept fully seperated regarding its bleed air output. This bleed air flow seperation is one of the great advantages of this new APU, apart from its simplified design concept. In the rare case of a compressor bearing seal leakage the leaking oil would not contaminate the bleed air supplied to the airplane fuselage cabins, as this could occur in conjunction with the preceeding APU models belonging to the bleeder type. The A310 APU consists of a single shaft power section with a two stages radial compressor and a three stages axial turbine.
The load compressor producing the bleed air output for the air conditioning system and main engine start is controlled by variable inlet guide vanes which, regarding their position, are automatically set to control the bleed air output.
Common to all APU's is an automatic starting system which only requires one switch position, i. e. master switch in start. This is being achieved by a speed control device which is connected to the acceleration and load schedule, based on electronic APU controlling.

As far as the design concept of the APU models is concerned, illustrated in figure 2, all APU's are lightweight design oriented, what has been one of the important design criteria, set up by the aircraft manufacturers. To meet both, the power requirements on board the aircraft and the APU lightweight design, a gas turbine compressor power design has been the only one practical approach of developing such a powerful equipment being permanently installed on board the aircraft. Of course, the rather low overall efficiency of the APU gas turbine, under full load in the order of 20 percent, is a negative factor with respect to the high fuel costs. The performance weight ratio for the APU's is in the range of 1.8 to 2.3, what is rather high, compared with the ratio of Diesel engines being used as mobile ground carts. The APU manufacturer, due to the fuel cost impact to airlines, has responded to the operators need of operating a more cost efficient APU by having designed the second APU generation of twin spool compressor power APU gas turbines, like the APU for the DC10 and the A300. Compared with the APU's being operated in the Boeing aircraft these APU models have a high cycle pressure ratio and are more cost efficient.

The proof, that the design goal of a more fuel efficient APU has been met, becomes obvious when comparing the amount of fuel flow versus air bleed flow, represented by the ratio W_F over W_B and shown in the right hand column of figure 2. Accordingly, the twin spool compressor power APU TSCP 700 provides a 32 percent lower ratio than the APU GTCP 660-4 of the Boeing 747 aircraft, i. e. less fuel required for producing the same amount of bleed air output. This also compares to the lower specific fuel consumption in pounds fuel per hour and equivalent shaft horse power of the 700 series APU's. All APU models are properly matched with respect to the bleed airflow demand of the aircraft air conditioning system, the bleed air discharge temperature and the electrical power and its frequency of 400 Hz, 115 Volts, alternate current output.

3. Economical impact on APU operation and remedy actions

What are the APU costs, an airline is encountering today ?
Figure 3 (see next page) illustrates the evaluation of the total APU costs per operating hour for the various fleets in operation. The graph clearly states the enormous cost impact to the airlines during the recent 3 years. Calculating the total variable cost increase per APU operating hour during the period of 1980 - 1982 , an average increase rate of 30 percent for this period, based on 1980 cost figures, is indicated for the APU's, installed in the Boeing 727, 737, Douglas DC10 and Airbus A300. For the APU 660-4 in the Boeing 747 the cost increase rate per APU operating hour is around 24 percent, what is an average figure of 8 percent cost increase per year.

The total variable costs per APU operating hour consist of the overall maintenance costs (i. e. direct engine costs per APU operating hour, covering the repair and overhaul shop expenses, including line maintenance costs, plus the fuel costs. As can be realized from figure 3, the fuel costs are the largest portion of the total APU costs, between 67 and 88 percent in 1982, depending on the APU model. The total maintenance cost - including line maintenance and overhaul cost - amount to a figure of 12 to 39 USD per operating hour for the APU's presented.

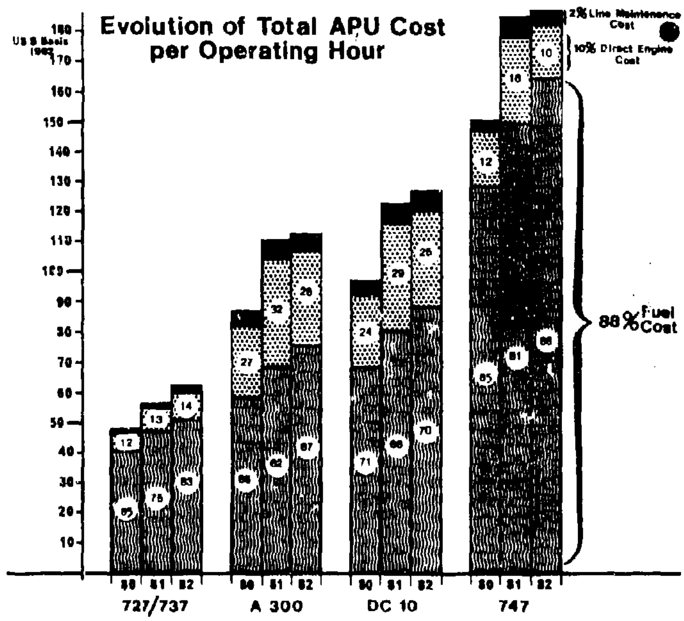

Figure 3

Considering the APU direct engine costs per operating hour for the twin spool engine type TSCP 700, installed in the DC 10 and in the A300 aircraft, a higher percentage is indicated than in case of the single spool engine, like the 85 series APU or the 660-4. This in fact would be expected due to the more complex APU design, involving more parts in the engine which might fail. It should be mentioned here, that the material cost has been increased during 1982 up to 40 percent of the cost figures during the preceding time period. If comparing the DC10 APU total cost figures with those of the A300 APU, both being of the same basic design, a slightly higher APU total cost figure in case of the A300 APU is shown. This can be explained by the additional maintenance and overhaul activities required due to the higher heat cycles/hour ratio of this APU when being operated in the short to medium airplane A300.

Figure 3 again stresses the high fuel cost impact on APU operation, and at that time, is provoking a vital interest in taking measures to reduce the APU costs or even introduce alternative ground power sources wherever practicable. Consequently, one of the first activities of the APU task force in our airline had been the reduced operation of the APU's on board the aircraft. Whenever possible, we have been limiting the APU operating time to those occurrences, where its use for producing electrical power and/or bleed output is inevitable, i. e. mainly for supplying pressurized air to air-conditioning systems for cooling and heating the airplane cabin and for main engine start up.

Further, with respect to the fuel conservation effect, all modification programs, offered by the manufacturers relative to APU fuel saving were investigated and,provided these modifications would pay off within an appreciable period of time, they had been already or are being embodied. Let me just mention here the most important fuel saving modifications embodied on the various APU systems:

Boeing 747

. Downtrim of the exhaust gas temperature schedule during load application, with the max. EGT under full combined APU load output limited to 550°C (This program under certain operating conditions could conflict with airport/high field levation due to marginal bleed air flow, limited by the max. possible EGT).

. Deletion of the ram air scoop at the APU air inlet conture of the 747 fuselage. After embodiment of this Boeing modification, which is supposed to reduce the aerodynamic drag by 0.12 percent, the APU cannot be started inflight but still can be operated inflight for producing bleed air power up to 16.000 ft and electrical power up to 23.000 ft.

. Manual load management (2 pneumatic ECS packs only).

Douglas DC10

. N1-speed reduction, limiting the low pressure spool under load conditions to a value of 94 percent N1 instead 97 percent with overriding capability to 100 percent N1 for main engine start.

Douglas DC10 continued
. Revising the APU 700-4 starting fuel schedule to avoid EGT peak conditions and reduce thermal stresses in the hot section, mainly turbine nozzle stator vane and high pressure turbine blade distress.

Airbus A300
. N1-speed reduction with variable speed limit selection/normal and override mode (91 percent for environmental control system supply and for main engine start; in override 97 percent for main engine start, retaining 97 percent for wing anti-ice under all cond.)

Boeing 727/737
. Timed acceleration schedule during APU start-up, requires modification of fuel control unit (subject still under evaluation).

Let me refer to the fact, that the majority of the fuel saving activities is directed towards down-trimming of the APU engine performance in conjunction with optimized fuel flow rating. The reduction of the material cost for engine high cost drivers, primarily hot section parts, is an important side effect of the fuel saving modification program. As a general maintenance step towards fuel saving the trailing procedure for airplanes is performed without operating APU's.

4. Utilization of alternative power sources on the ground

From the high APU operating cost figures, presented in paragraph 3, it becomes obvious, that alternative ground power equipment, being more economical than the APU's, should replace APU operation on the ground, if practical; thus reducing the cost burden to the airline.

Let us investigate now the possible ground sources for on board power supply. There are two major groups of alternative ground power support systems. Both meet the on board power requirements of commercial aircraft, and if available, can be used as substitute for APU ground power operation. The first group comprises the fixed stationary power systems and the second group covers the mobile power systems.

On figure 4 today's most commonly used alternative ground power equipment in airline business is presented in a block diagram, supplying all kind of secondary power required on board the aircraft, either electrical power or pneumatic power or a combination of both.

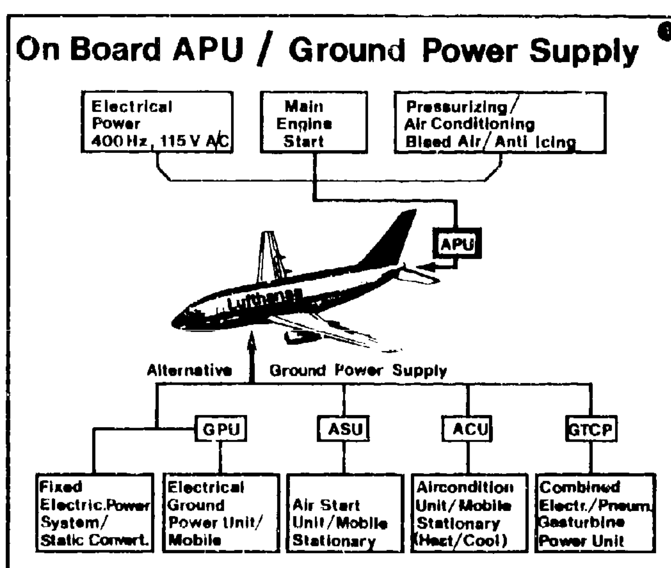

Figure 4

Let us briefly review the technical feature of the ground support equipment, shown in figure 4.

The electrical power normally is produced by a mobile electrical power generating unit, known as the GPU, which is mounted on a mobile trailer or truck and is driven by a Diesel engine. The engine is flexibly coupled to a brushless alternator. The GPU can be self-propelled by the engine power through a gearbox , which is directly connected to the rear axle of the truck through the propelling shaft. The generator power produced is a continuously rated output of around 100 KVA, 115/220 Volts, 3 phase 400 Hz. The output voltage of such a machine is maintained within ± 1 % of the nominal value of the aircraft socket under all conditions of load and power factors. This output voltage control in most cases is achieved in the generator system by means of a transistorised voltage regulator. The frequency is maintained within 1/2 percentage under steady state conditions.

Electrical systems if stationary are referred too as "Fixed Electrical Power Systems". Let us consider the most important details:
Fixed, high frequency ground power systems, capable of providing the electrical on board power for narrow and wide body aircraft in civil aviation, consist of one or two stationary electric motor/generator assemblies which convert 50 (or 60 Hz) city network power to 400 Hz output by direct coupling to a 400 Hz generator. Normally a synchronous, brushless motor/generator set is being used,with the rotating group axis mounted in a vertical or horizontal direction. Due to a minimum of component parts, which rotate at relatively low speed, requiring lubrication as the only maintenance action, a maximum of trouble free operation is being guaranteed. Thus the motor/generator assembly is regarded a very reliable piece of equipment amongst other aircraft support systems.

Based on worldwide experience three basic voltages are being used in 400 Hz centralized systems of major airports:

- Low Voltage - 115/200V, 4 wires, 3-phase system
- Medium Voltage - 575V, 3 wires, 3-phase system
- High Voltage - 4160V, 3 wires, 3-phase system

The low voltage system (115V) being used as decentralized system, requires an input power of 380V, 3-phase, 50 Hz, with 1500 rpm speed for the brushless generator, which is of the revolving field type. The generator output is 115V, 4 wires, 3-phase system, 400 Hz power. One disadvantage of this low voltage system is, that it can only serve one aircraft with limited distance (appr. 20 meter) permissible between the motor generator set and the aircraft ground connection, in spite of large multiple conductors and 3 cables per phase to compensate voltage transmission losses. The utilization of a line drop compensator would allow increased distance up to approx.170 m. Thus for example from a single motor generator set with a rated output of 200 KVA multiple gates (up to 10 ea) with each gate rated at 60 KVA can be served in conjunction with narrow body power supply. This is possible,as the average electrical demand of a B737 or B727 is not higher than 20 KVA. The advantage of the low voltage system is given by the fact, that no voltage transformers are needed,unlike medium and high voltage systems.

The medium voltage system (575V) is comparable to the low voltage system, showing the same performance characteristics and construction details, except the generator output is 575V, 3 wires, 3-phase, 400 Hz. The great advantage of the medium voltage system is its flexible capability, to serve multiple gates with normally sized cables over remarkable distances along the airport terminal.
Up to now only electrical motor/generator power systems have been referred to. Another means of generating 400 Hz electrical power is indicated by the possible use of static converters, installed in fixed power systems. According to our knowledge static converters should be more economical than motor/generator power systems.

Pneumatic ground support equipment, like the Air Starter Units (ASU) or the Air Conditioning Units (ACU), do not differ from each other, regarding their principle design. Typical for both designs is the requirement to deliver a well defined output flow and air pressure as well as a certain bleed air output temperature. The most common way of supplying pressurized air on the ground is achieved by operating a Diesel engine driven rotary screw type compressor with a step-up gearbox, which is attached to the engine via a flexible coupling. Normally the rotary screw compressor is designed for starting jet aircraft. By reducing its output pressure the machine is also used to power air conditioning on board the aircraft or to warm up engine air intakes and de-ice windscreens. The necessary change in operation mode from startair to low pressure air supply is achieved by operating a switch position which reduces the air pressure from approx. 40 psig to 25 psig for aircraft aircycle package operation. The pressurized bleed air is supplied from the truck through an air hose to a 3 inch diameter connection fitted to the aircraft.

The rotary screw compressors provide output air capacities in the range of 130 to 300 pounds per minute, which is comparable to the bleed air output of the APU's. In each case the air of the rotary screw compressor is warmed and totally oil free. Either mounted on a truck, trailer or skid or even stationary, somtimes as a parallel system to provide the bleed air output needed, the rotary screw compressor is capable of servicing the wide body aircraft like B747, DC10 and A300, as well as all narrow body aircraft. With a 280 pounds per minute bleed air output of the rotary screw compressor the speed of the turbo charged Diesel engine is around 2000 rpm. The noise of such a machine

when attenuated is around 85 dB(A).
The GTCP gas turbine type used as mobile ground power equipment and capable of supplying all kind of combined power output does not mean any economical advantage, as its fuel consumption is comparable to that of APU's.

Knowing the technical features of the various types of umbilical ground support equipment, it has been of great interest to assess the hourly operating and maintenance costs, comparing them with the APU costs.

ELECTRICAL GROUND POWER SUPPLY (VARIOUS SOURCES) HOURLY OPERATING AND MAINTENANCE COSTS (USD)				
AIRCRAFT TYPE	TYPICAL ELECTR. REQUIREMENT (KWH)	400 HZ FIXED POWER SYSTEM	GPU (DIESEL ENGINE)	APU
727 / 737 (NARROW BODY)	16	1.2	6.5	41.0
DC 10 / A300 (WIDE BODY, 3 & 2 ENGINES)	27	2.0	8.6	89.0
747 (WIDE BODY 4 ENGINES)	40	2.9	11.0	179.0
COST FOR 1 KWH (1982 $)		0.068	0.187	-

Figure 5

In figure 5 a calculation of the hourly based operating and maintenance costs (in 1982 USD) is made for the supply of typical electrical power requirements to the different fleets. To everybody's surprise the cost for the 400 Hz Fixed Power System is only a fraction of the cost for a GPU; the comparison becomes really spectacular with respect to the APU costs, spent for the same amount of electrical energy (factor 60 !). However, only the variable costs are being compared here. To make an accurate cost analysis and economical assessment of ground power support equipment, the cost investment for designing and installing such a power system have to be considered as well.

Lufthansa has made economical studies to optimize the usage and the destinct operation of ground support equipment, also relative to the layout of future airports. In conjunction with a new airport the return on investment would be reached within one or two years already, considering the installation of a Fixed Electrical System.

Let me summarize the results of our studies:

- Fixed Power Systems, made available for suppling electrical power (400 Hz/115V A.C.) are most economical on major airports, depending on the number of take-offs and landings per day and the number of gates. Apart from the great potential in cost and fuel energy savings the Central Fixed System, when used for supplying aircraft at terminal gates, has the following additional advantage over the GPU and the APU: No environmental pollution due to exhaust cases; no noise problem and no congestion around the parked airplane at the ramp.

- Central ground compressors for gate operation, supplying pneumatic power on board the aircraft, are justified for major airports only in case a high landing frequency per day and a uniform dispatch profile are maintained. More economical would be a decentralized stationary pneumatic system, achieving highest cost savings, when compared with the APU.

- Mobile ACU's and ASU's, either used as single or combined system version, are cost and fuel saving, when compared with the APU. Wherever ground equipment is operated on a rental basis (valid for most LH outside stations), the savings are reduced from approximately 70 percent down to 25 to 40 percent per operating hour, depending on the ground support mode and the type of aircraft operated.

- The APU is fuel and cost efficient, if operated during push back for main engine starting and during taxiing out.

Recent considerations during taxiing-out are to start those main engines, which are not required for rolling the aircraft along the taxiway from its parking position to the take-off point, as late as possible, i. e. by APU power. In case of a DC10, for instance, the fuel saved by the centre engine during taxiing out, due to delayed engine start-up, would be 11 kg/min. compared with 2.5 kg/min. consumed for the extended APU operating time at idle power (N1 = 63 %); i. e. 8.5 kg/min. saving. To give an example: a taxiing-out time of 5 minutes and a delayed take-off of additional 10 minutes, due to traffic congestion at rush hour, would mean a saving of 50 USD per event.

- APU electrical power supply on board the aircraft is economical, provided the APU has to be operated for air conditioning, i. e. cooling or heating at the gate position.

- The economical decision, to either operate the APU or any alternative ground support equipment, depends on a number of airline operational factors and, therefore, should be made individually on a station by station basis. The most important factors, influencing the operational requirements and power supply conditions at any airport, are: aircraft type, airport equipment availability, passenger load factor, passenger comfort, transit or final destination stop, ground time, local temperature profile, seasonal peaks, cabin temperature prior to landing, environmental control system demand, galley demand.

5. Plans and practical approaches to achieve cost efficient on board power

From the cost comparison, presented in paragraph 4, it becomes obvious that the highest potential of fuel and cost savings would be achieved, if electrical ground power from a fixed system or a GPU is being provided as the only one requirement, without the need for parallel cooling or heating of the aircraft cabin. This condition does exist in those cases only, where the ambient temperature would permit the neglection of the air conditioning supply to the cabins. Therefore, prior to establishing any revised operational procedures for pneumatic power supply, Lufthansa has reviewed the temperature of the cabins and the cockpit, with special respect to the previous and revised temperature ranges to be applied as temperature comfort zones. Several evaluation programs have been performed, measuring the temperature profile on board our aircraft mainly the cooling-down or heating-up time of the cabin on the ground with and without passengers, shifting under the influence of outside ambient conditions (i.e. sun radiation, open cabin doors). Also effects of cabin pre-conditioning by main engine bleed air supply during aircraft approach had been evaluated.

To support the experience made with revised temperature comfort zones for different fleets under the influence of environmental conditions, yearly statistics of a German weather research station have been reviewed to establish a guide line and estimate of the average air temperature (OAT) distribution during an annual period.

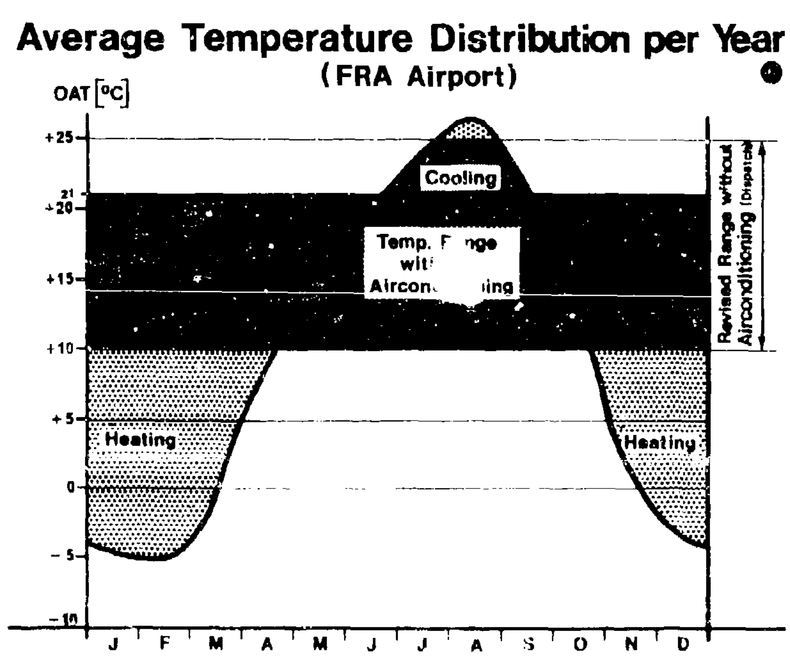

Figure 6

Figure 6 presents a plot of the average ambient temperature as distributed per year. The local area is Frankfurt, being the Lufthansa main base. As can be seen on figure 6 the lowest temperature in the airport region is in February, early spring time, and the highest temperature peak in mid August. Based on the annual average temperature distribution curve in approximately 50 percent of all events it would be necessary to heat the cabin, and only in 10 percent cooling-down would be required. Although the annually based average temperature distribution profile would differ from station to station, it supported our basic understanding of critical and non-critical temperature zones in the aircraft cabin. On figure 6 the temperature range, without air conditioning supply is presented in its previous limits (10° C to 21° C) and in its revised limits (10°C to 25° C for dispatch). Another reference line, at + 5° C is recommended for use under airplane transit conditions. Exceedence of the above mentioned non-critical temperature zones in either direction (increasing or decreasing) would require air conditioning supply by the APU or ACU equipment. The revised temperature reference limits are feasible due to the fact, that latest at time of passenger boarding the APU will be started for providing APU pneumatic and electrical power to the aircraft.

After a successful evaluation period of 1.5 years on board our A300 fleet, we have gained a confidence level, which permits the application of the revised procedures in close cooperation with flight operation and system engineering.

Airline Supplementary Procedures
for Economical Use of APU

X Preferred Choice of Power-supply, required on the Ground.

Airplane Operational Mode	OAT °C Ground Temperature Range	Ground Supply GPU Elect.	ACU Heat	ACU Cool	ASU MES	APU Elect. Load	APU Pneumat.	Recommended Procedures for Cockpit Crew and Ground-Staff/Maintenance
Approach	below 5°C							Based on forecasted cold or hot OAT's, warm up or cool down cabin to extend time during which on the ground APU is not required for Airconditioning
	above 25°C							
After Landing	below 5°C	X	X			X	X	Provided, electrical ground power is available and ground time is >1 hour, do not start APU after touch down. (Exception: 747, APU operation required for pull in airport position). Electric GPU should be connected to Aircraft immediately after on block.
	between 5°C and 25°C	X				X		
	above 25°C	X		X		X	X	
Transit <1 Hour								With passengers, remaining on board, APU should be started after landing, regardless recommended OAT temperature range (Narrow Body)
Transit >1 Hour	below 5°C	X	X			X	X	Electrical ground power GPU shall be used until passenger boarding, if possible, but at least up to 15 minutes after being on block.
	between 5°C and 25°C	X				X		
	above 25°C	X		X		X	X	
Dispatch	below 10°C	X	X			X	X	In case of prolonged ground time, start up APU approx. 45 minutes before dispatch. Principally, regardless OAT Figure, APU should be started latest at passenger boarding.
	between 10°C and 25°C	X				X		
	above 25°C	X		X		X	X	
Main Eng. Starting						X	X	MES shall be performed during Aircraft push back as late as possible. Afterwards shutdown APU.
Taxiing out							X	If taxiing-out with delayed start of the main engine, APU is kept running for starting engine prior to take off
Parking								No APU operation

Figure 7

Figure 7 presents details of the recommended procedures under various aircraft operational modes with the preferred choice of the most economical power supply, i.e. either ground equipment or on-board APU. The procedures reflect the economical use of the APU and can be described best by following general rules for ground maintenance personnel and cockpit crews:

- At airplane dispatch with OAT's between 10° C and 25° C and during transit between 5° C and 25° C normally cabin and cockpit do not require air conditioning.

- Once cockpit and ground personnel have ensured - by checking a weekly updated status list - that appropriate and reliable ground power equipment is available at a specific airport, this ground power shall be supplied to the aircraft after landing, without starting the APU.

- In general, the use of the APU for air conditioning should be initiated only, if passenger comfort and/or cockpit cabin crew working conditions are impaired.

- The cockpit crew takes full responsibility to judge the need of APU operation for air conditioning purpose.

- With APU running, both pressurized air for air conditioning and shaft load for electrical power shall be provided, since economical.

- Based on forecast OAT's on the airport the aircraft cabin should be pre-air condioned during approach (i. e. cabin should be warmed up in case ambient ground temperature is below 10° C and should be cooled down in case ground temperature is above 25° C. A temperature differential of 3 to 4° C will already provide sufficient margin to extend ground time without the need for air conditioning supply).

As a first step the supplementary procedures for the economical use of the APU's in our fleets have been set up and introduced into the flight crew operations manual and meanwhile are being applied to all fleets, operating on European and North American routes. A similar cost efficient program on other routes will follow. We are aware of the fact, that the recommended procedures are conditional procedures; this means, they depend on the individual decision of flight crews, judging on the momentary specific requirements for air conditioning and passenger comfort on board the airplane.

As a direct consequence, after having started the APU reduction program, the evolution of the APU usage factors, shows a drastic decrease, primarily on the wide body fleets B747, DC10 and A300 (by 40 %, 35 % and 48 % respectively, during period 1980 to 1982).

Figure 8 is referring to the evolution of the APU usage factor.

Evolution of APU-Usage Factor ●

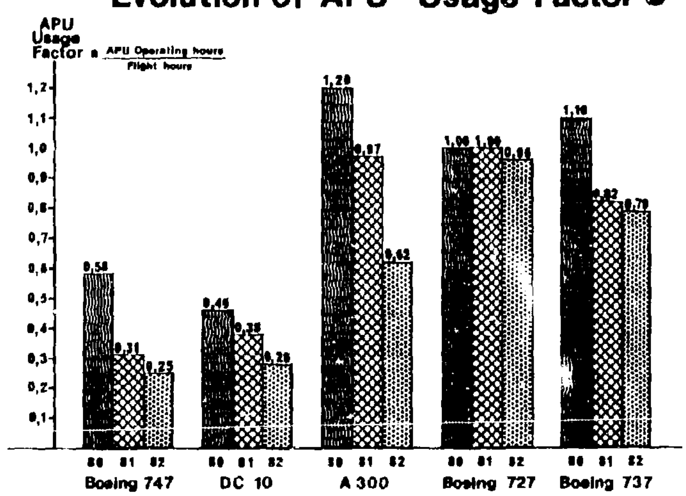

Figure 8

As can be seen from the graph, the narrow body APU's are still used more often than expected. Apparently this is due to short to medium operation of the B727 and B737, and transit times shorter than 1 hour.
The APU operating costs during the observed period have also decreased, as expected. The downward trend is indicating an APU operating cost reduction in the order of 30 percent for 1982.

6. Conclusion and recommendations for future APU operation

The promising results, which up to now have been reached by the introduction of the cost efficient airlines program, optimizing the economical use of on board power support equipment, justify the feasibility of the recommended operating procedures. The author believes that the utilization of the Auxiliary Power Units in the Lufthansa fleets can be further reduced, although, with respect to the Boeing 747 APU operation, the APU usage factor has already decreased to a level, which would represent a minimum practicable APU operating time at all.

The Auxiliary Power Unit being capable of supplying multiple power to the aircraft makes commercial airlines entirely independent of ground facilities and in addition will improve the airplane redundancy during dispatch and in flight. This might be required because of a main engine power generation problem or for wing anti-icing reasons. APU's, installed in commercial fleets, will retain the operational flexibility of an airline and, therefore, have still their technical and economical justification, in spite of fuel saving considerations.

It should be the task of the aircraft manufacturers and the APU manufacturers, in close cooperation with the airlines, to specify and design future APU auxiliary power systems for commercial aircraft, which are optimized regarding the aircraft power demands and at the same time are more cost efficient to airlines.

DISCUSSION

P.Vaquez, Fr

What has been the influence of modifications recommended by manufacturers on maintenance cost?

Author's Reply

The incorporation of fuel saving modifications (A300 and DC10 APU N1 speed reduction) had a twofold positive influence on the maintenance costs: First, the APU fuel cost per operating hour are reduced by 2, 5 percent, and second, due to the reduced N1 spool speed (i.e. 5 to 6% less than the certified value) the cyclic life of the APU hot section parts is being improved. This means reduced overhaul costs due to less parts replacement, mainly high cost drivers.

C.Rodgers, US

Have you determined additional fuel cost required to pre-cool cabin on approach as compared to using the APU?

Author's Reply

Additional fuel costs to pre-cool or warm-up the aircraft cabins during approach, in accordance with hot or cold ground temperatures, had been calculated for wide body aircraft, based on certain assumptions. Figures have still to be verified by actual tests. Accordingly the additional engine fuel costs due to engine bleed air subtraction with distinct temperature requirement will be less than APU fuel costs on the ground (i.e. approximately 25% of the actual APU fuel costs).

R.Smith, US

Fuel costs were shown in 1982 US \$. Does this reflect your cost (actual) as purchased over your route structure and then adjusted to reflect currency exchange rates?

Author's Reply

Yes! But please note: As the actual APU fuel consumption per operating hour is not measured during operation (there are no APU fuel flow-meters installed in the aircraft!), our calculation is based on average APU fuel consumption figures, taken from APU test bench results under typical load conditions, times APU operating hours, recorded over the period of time.

APU OPERATIONAL EFFICIENCY

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SUMMARY

of — which determine

This paper presents some observations based on experience with a new generation ~~of~~ Auxiliary Power Unit (APU) installed in ~~a~~ commercial transport aircraft. The subjects considered are operational requirements and efficiency of the APU in the various operating modes. Particular importance is attached to such matters including ~~the factors~~ ~~determining the~~ APU performance characteristics with respect to a cost effective APU system. These considerations are conducted with a look at ~~future~~ development trends, seen from the ~~point of~~ viewpoint view of the aircraft manufacturer.

1. INTRODUCTION

Increasing fuel prices, and the resulting higher costs for the production of energy have, over the past few years, made the airborne auxiliary power units(APU) in commercial transport aircraft a major point of interest.

In future generations of transport aircraft, the airborne APU will continue to be the basis of the on-board, independent power supply and it is, of course, obvious that, from the operator's point of view, costs should be kept to a minimum.

Initially, the Airbus A310 and its APU will be used to illustrate the present state of the art of APU development, and also to examine the possibilities of minimizing the operational costs of the APU from the aircraft manufacturer's point of view; eventually, specifications and requirements will be established as Design guidelines for the APU manufacturer.

Another topic discussed under this title is the integration of the APU into the "Inflight Auxiliary Power Generation System" of the aircraft, with due consideration of the effects on APU design.

2. THE APU IN PRESENT-DAY COMMERCIAL TRANSPORT AIRCRAFT

2.1 HISTORY OF APU DEVELOPMENT

Based on the history of APU development, this section further explains the present state of the art of APU technology: three development stages lead to the generation of APU's that are today flying in current commercial transport aircraft.

1st stage - Integral-bleed, single spool APU
2nd stage - Interstage bleed, twin spool APU
3rd stage - Direct driven load compression, single spool APU.

1ST STAGE

Figure 1 (see appendix) presents a schematic build up of an integral-bleed, single spool APU. The unit is made up of a single-shaft gas turbine, a gear box, which is direct-driven from the gas turbine, and a generator driven from the gear box. The APU is operated at constant speed throughout, since the generator speed must be kept constant; bleed air supply is bled downstream of the gas turbine compressor. Again, this type of APU is identified by an unsophisticated constructional buildup and mechanical control. The generation of the bleed air supply by the gas turbine compressor does, however, lead to a lower level of efficiency for the APU under partial load conditions.

With this conception, every effort has been directed towards improving the APU design as a measure to counteract the increasing cost of fuel.

2ND STAGE

The interstage-bleed auxiliary power unit represents an intermediate stage in APU development, a typical build-up is shown in Figure 2 (see app.) and is similar to that of a twin-spool engine, each with it's own compressor, turbine and controls. Bleed-air parts are situated between the LP compressor and the HP compressor. The high-pressure spool drives the gear box from which the generator drive is taken, and thus this high-pressure spool speed must be kept constant, - this is a prerequisite for a generator driven from the gear box. The APU power output can be regulated to meet the prevailing demand by controling the speed of the low-pressure spool accordingly. Due to the fact that drive-power and usable-power generation are thermodynamically coupled in this type of APU design, sophisticated and expensive controls are necessary; the resulting considerably increased complexity is responsible for the relatively high-cost maintenance operations which are required for this APU type.

3RD STAGE

The latest stage-so far- in APU development is represented by the type described in paragraph 2.2.1; this APU type combines system simplification with a high degree of efficiency under partial load conditions. Full electronic control is a considerable contribution to the mechanical simplicity of the APU.

Summing up the above-described developments, it can be said that in the near future few essential changes will be made to the unsophisticated basic buildup of the APU; therefore, any plans for a cost reduction in the field of APU operation will have to concentrate on, and probably be restricted to, the improvement of individual components of the APU; e.g. higher efficiency of compressor and turbine, improved reliability, etc.

2.2 THE AIRBUS A310 AND ITS APU

The airborne auxiliary power unit supplies bleed air and electrical power for self-contained operation of the aircraft, that means that aircraft operation is independent of ground power sources.

The major design criteria of the APU are dictated by the primary tasks to be performed on the ground, with main engines off. The following tasks are typical of the APU installed in the A310 -

- provide bleed-air for main-engine starting (MES) on the ground, at airfield attitudes ranging from - 1,000 ft (-305 m) to 8,000 ft (2,440 m) AMSL.

- provide bleed-air for air conditioning of passenger and flight crew compartments, at airfield heights ranging from -1,000 ft (-305 m) to 8,000 ft (2,440 m) AMSL.

- provide full rated shaft power to drive a 90 kVA AC generator, while meeting the demands of air conditioning and/or main engine starting bleed-air supply simultaneously.

On the A310, no provision is made for the direct supply of the hydraulic system, via an APU-driven pump, it is however, possible to provide hydraulic power supply during APU operation by using the airborne electrical power system and an electrically driven pump.

The APU, which is primarily designed for ground operation, is also available in flight to fulfill a number of tasks, viz:

- to provide bleed air for air conditioning only, up to an altitude of 20,000 ft (6,096 m).
- to provide bleed air for emergency anti-icing of wing leading edges, up to an altitude of 20,000 ft (6,096 m), whilst also meeting minimum environmental control system performance requirements.
- provide shaft power to drive a 90 kVA generator, which can be loaded with 90 kVA up to 35,000 ft (10,668 m) and with 75 kVA up to 41,000 ft (12,497 m).

With a view to obtaining and maintaining the above mentioned in-flight performance values, the design of the APU installation has been adapted to flight conditions, while, at the same time, the retaining of low installation pressure losses for ground operation was given due consideration as an overall condition.

In-flight performance data for the APU are of particular importance for twin-engined aircraft. With one main-engine generator - or one main-engine bleed-air system - failed, the aircraft remains fully operational, as the APU, together with the serviceable system of the other engine, will provide redundancy of systems. Thus the APU contributes considerably to the high dispatch reliability of the Airbus, which stands at 98 percent or higher for the A300 in airline service. The APU operating envelope for the A310 is presented in Figure 3 (see app.): it shows that the APU can be operated throughout the entire flight envelope of the aircraft, this is a typical feature of the new generation of 200-seater short/medium range transport aircraft..

2.2.1 DESIGN AND CONSTRUCTION OF A MODERN APU

Design and construction of the APU should be aimed firstly at economical operation, whilst, as far as possible, meeting the following particular requirements:

- . low weight combined with high performance
- . low fuel consumption (even under partial load condition)
- . economical maintenance
- . long service life
- . simple handling.

Here, again, the APU of the Airbus A310 can be used to demonstrate the present state of the art, particularly with regard to the above detailed engineering specifications.

2.2.2 THE ENGINEERING FEATURES OF A MODERN APU

The constructional and control features of the APU are prerequisite to the economical operation of the unit. The present generation of auxiliary power units is exemplified by

- . modular design
- . direct driven load compressor
- . load compressor output regulated by inlet guide vanes (IGV)
- . full digital electronic control
- . single spool / constant speed

Fig. 4 (see app.) shows the schematic build-up of an APU incorporating these features; it is, basically, a single-shaft design, but is divided into three modules - power section, lead compressor, gear box. The single-shaft gas turbine provides direct drive for the load compressaor and the gear box.

To comply with the constant-speed drive requirement of the gear box-driven generator, the entire APU thus operates at constant speed.Nevertheless, to ensure an individual supply to the aircraft penumatic system, variable inlet guide vanes are located upstream of the load compressor. The airflow is controled by modulating these inlet guide vanes based on aircraft demands, as defined by an electrical demand signal from the APU electronic control box. Air developed by the load compressor which is not utilized by the aircraft pneumatic system, must be vented overboard through the surge control valve.

The control features of the APU comprise essentially the fuel supply control, IGV adjustment and load compressor surge protection by means of a surge control valve.

This type of APU has a minimum number only of airframe interfaces; thus, the load compressor, the power section and the cooling fan are supplied from one common air intake. Exhaust gas, gear box vent and surge air are ducted together within the APU exhaust duct.

The build-up philosophy of this APU conforms to the general requirements specified in 2.2 with the following features:

By separating the power section and bleed air generation, it is possible to choose a simple thermodynamic design for the power section, and, hence, to combine a high degree of efficiency with a low power-to-weight ratio and a relatively low specific fuel consumption.

The load compressor output is regulated by IGV's, where output is a function of aircraft system demand, and power absorption of the engine compressor is reduced to a minimum, if shaft power only is demanded (approx. 20 percent of max. power output). The advantages of this arrangement are good adaptability, and a resultant low fuel consumption for the APU under partial load conditions.

The APU also offers a good in-flight performance when power is demanded for generator operation only, since load compressor power can be considerably reduced.

The simple build-up and the modular construction provide for easy installation and good accessibility. As a result, the unsophisticated APU system offers many advantages in maintenance and handling characteristics.

The full-authority microprocessor electronic controler is a major element in the APU technology advancement achieved. The controler provides safe, precise, and fuel efficient operation.

Equally important, the use of the microprocessor, in conjunction with other Built-In-Test Equipment (BITE), makes it possible to isolate quickly component malfunctions, and display this information on an LCD back-lit panel, see Figure 5 (see app.).

The system checks for faults prior to each APU start and then continues to monitor system operations while the APU is running. A Self-Test mode allows a quick check of maintenance actions, and mini-flag interrogation permits detailed system trouble-shooting. Non-critical faults are recognized, and alternate values are substituted into control logic to allow continued operation.

An adjustment feature incorporated in the front panel of the ECB allows reduction in APU output power, if a particular airline route structure allows APU de-rating in order to minimize APU fuel consumption, and maximize APU life.

2.2.3 THE APU POWER RATING

Power rating of the APU installed in the Airbus A310 is governed by the output demands of the various aircraft systems, as represented by the power demands of the airconditioning system during operation on the ground, the airborne electrical power loads, and the pneumatic power required for main engine starting; these principally define the dimensions of the APU power output.

The power requirements of the air conditioning system, when supplied by the APU, are identified by short-term cabin cool down/pull up periods, e.g. at an ambient temperature of 38°C on the ground, it is possible to cool an aircraft interior which has heated up to 38°C, down to 27°C, within 30 minutes.

The power consumption of the engine starting system is governed by the planned start-up time; thus the A310 is able to reach a start-up time of t = 30 sec., or less (ISA standard conditions, sea level).

The APU installed in the A310 is able to satisfy full rated generator load demand together with the required max. bleed air output; electrical power loads are independent from the air conditioning or the engine starting system (pneumatic) loads.

In practice, extreme loading of the APU occurs only occasionally, and Figure 6 (see app.) shows a typical APU duty cycle. This illustrates that the full thermodynamic performance of the APU is used during only 5 percent of its operational time; during 95 percent of the operational time, the demanded power output is considerably lower than the rated power of the APU. Since, due to the unsophisticated build-up and the high degree of reliability to be achieved, the thermodynamic design is rather conservative, the comparatively low average loading of the APU will have a positive effect on the reliability and on the service life.

3. APU OPERATIONAL EFFICIENCY

The APU operating costs are influenced by the following factors:

. operational time
. level of demanded power output
. degree of efficiency of APU
. maintenance and overhaul costs (reliability)
. indirect costs shared by APU as part of aircraft weight.

The APU manufacturer can contribute considerably to the efficiency of the APU and to the level of maintenance/overhaul expenses, as he is responsible for the build-up and the thermodynamic design characteristics of the APU; he will also select suitable materials to limit the APU weight. The design of an APU, therefore reflects the continuous effort to achieve the synthesis between fuel consumption (efficiency), weight, reliability and material costs, as related to the required performances. Based upon the principle of the direct driven load compressor, the APU build-up has now reached an optimal stage of development, so that any further improvement of the APU in the immediate future will be restricted to the optimization of detail parts, as previously stated.

The operational time of the APU is directly controled by the aircraft operator. Cost-conscious operation will considerably reduce APU fuel consumption, for example:

- By switching off the APU, where a demand for aircraft air conditioning does not exist; and hence, by making use of available ground power sources for the direct supply of the airborne electrical power system.

- By fully utilizing the power rating of the APU for the air conditioning system, i.e. restriction of APU operation to a period of 20 - 40 minutes before boarding of passengers, where cooling-down or heating-up of the cabin is required.

- By always starting main engines on APU whenever possible (the most efficient method).

The levels of the power output demands, which control the size of the APU, are governed by the relevant aircraft system specifications (air conditioning, starting, electrical power) and their power requirements. The performance data of these aircraft systems are again defined by the aircraft manufacturer, with contributions from the customer, as the eventual operator. Possible improvements of APU design, particularly with regard to specific physical properties, may now be considered in the following paragraphs.

3.1 IN-FLIGHT AUXILIARY POWER UNIT

A current-generation APU is characterized, among other features, by effective utilization under in-flight conditions at higher altitudes; this feature seems to recommend the utilzation of the APU throughout the entire flight envelope.

The auxiliary systems of the Airbus are basically divided into two separate supply circuits, partly due to the design philosophy, but always the result of the relevant redundancy requirements. Engineering problems are, however, created by the incorporation of the APU into the entire flight envelope.

With parallel operation of main engines and APU supplying the air conditioning system, no direct coupling of the systems can be allowed, due to mutual interference. For this reason, a considerable - and unjustifiable - technical expenditure would be required, for example, the provision of a third air condition pack.

The addition of a third generator to the airborne electrical power supply system of the Airbus would mean the addition of a third supply circuit, or the synchronization of the three generators. The generation of electrical power by the APU under in-flight conditions is less economical than by main engine generation, and this factor also, does not commend full utilization of the APU under in-flight conditions.

The integration of the APU into the "in-flight auxiliary power generation" concept does not offer any engineering or economical advantages; therefore, APU design should be confined to ground operation in commercial transport aircraft. In-flight operation of the APU will remain restricted to the replacement of a main engine generator or bleed air system, in case of failure, within the limitations of the design as defined by ground operation conditions. In order to utilize the APU load-potential to the fullest extent, for in-flight operation also, the special requirements of air intake and exhaust must be complied with, as for example, a "Scope" inlet.

3.2 IMPROVED APU POWER RATING

The design philosophy for an Airbus APU has, so far, been governed by the demands of the relevant aircraft systems - air conditioning, starting system, airborne electrical power supply. This philosophy resulted in a wide variety of loads. As an example, Figure 7 (see app.) presents a graph illustrating the APU load in cabin conditioning mode as a function of ambient temperature. As shown in 2.2.3 above, full APU rating is demanded only under extreme ambient temperature conditions; that means that some 5 percent of the APU operational modes define the power rating and the dimensions of the APU, as the design is to be based on max. demand conditions.

The specific consumption of current APU types is satisfactory, even under comparatively low output demand conditions. The efficiency does, however, deteriorate with decreasing output demand, so that a design, which is based on high peak demands, will produce a negative effect on duty cycle fuel burn

When considering the definition for a 150-seater commercial transport aircraft for the nineties, the question of a new APU design philosophy, to be based with preference on the physical characteristics, was first discussed.

A commercial transport of the new generation will have comparatively high system weights in comparison with derivatives of previous aircraft types; this stems from, among other causes, the high system performance requirements - including that of the APU - for the new aircraft; therefore, the definition of a new-generation APU was influenced by a wide variety of demands, similar to those for the APU of the Airbus A310.

In practice, an improved APU design represents an improved utilization of all available APU power rating. Less efficient utilization of the APU rated power output may arise from:

. trends to cut main engine starting time.
. comparatively high air conditioning output demands at extreme ambient temperatures.
. generation of maximum bleed air output combined with full rated generator load.

Figure 8 (see app.) illustrates the influence of main engine starting time on the bleed air output available. Whereas today, starting time values of 30 sec. or less are practised on the Airbus A310 (ISA standard conditions, sea level), extended starting time values - up to 40 sec. - would considerably reduce the APU power output demand in the main engine starting mode.

The full power rating of the APU will only be demanded under high ambient temperature conditions, when air conditioning and airborne electrical power have to be supplied simultaneously. As mentioned above, for the A310, the APU design point is defined by the cabin cool-down time to be achieved. At sea level, for example, with an ambient temperature of 38°C, the time required to cool down the cabin from 38°C to 27°C is 30 minutes. If, however, this period were to be extended to 40 minutes, the APU load would be reduced by 20 percent.

Nevertheless, the design of the APU need not be based on the simultaneous supply of max. bleed air output and full rated generator load. During normal ground operation of the aircraft, there will only be a short-term peak demand for max. bleed air and electrical power to be supplied by the APU at the same time; therefore, the APU could be designed to supply max. bleed air output together with normal electrical load. Figure 7 (see app.) shows that the adaption of the APU design to such physical characteristics leads to a reduction in the APU performance requirements at extreme ambient conditions (operation under "hot-day" conditions); thus the APU design could be based on a reduced power rating, with improved fuel consumption and a lower APU weight. However, an APU designed along these lines would have to carry higher loads than the APU in the A310. It will, therefore, be important to ensure that the experience gained from the operation of the first generation of new APU's is incorporated in the design of any new APU, so that higher loads can be combined with constant (or improved) reliability and economical maintenance.

A certain reduction in convenience will result from the reduction of APU size, and customers will need to decide whether future APU designs will tend to have physical properties as the primary consideration.

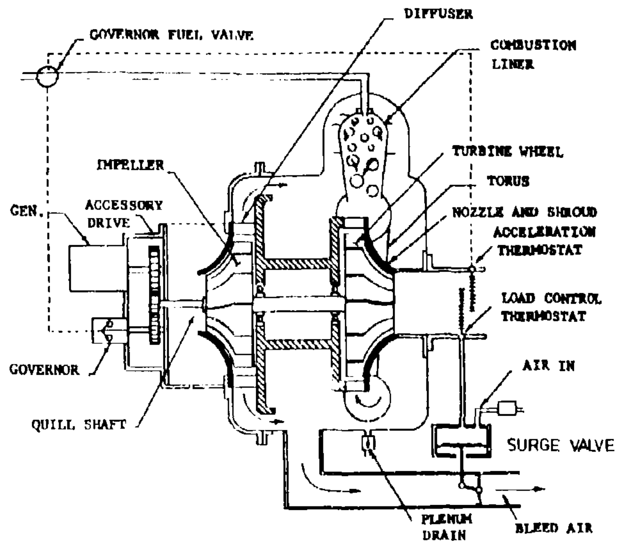
4. FUTURE DEVELOPMENT TRENDS

Design features of future APU's will be strongly influenced by fuel price trends. Even if it is assumed that there will be few major changes to the basic build-up, in the sense that the direct driven load compressor concept is retained, rising fuel costs will encourage the introduction of more economical equipment - e.g. integrated drive generator (IDG) or variable speed constant frequency system (VSCF) - into the APU concept. With VSCF or IDG, the variable speed feature is added to the single spool APU, which would provide for the further improvement of efficiency under partial load conditions.

Changing from the single shaft design to the twin-shaft design, while retaining the direct driven load compressor concept, presents another opportunity to improve APU efficiency. Figure 9 (see app.) illustrates the schematic build-up of a twin-shaft APU; this differs from the single-shaft APU in that gas generator and power turbine are mechanically separated.

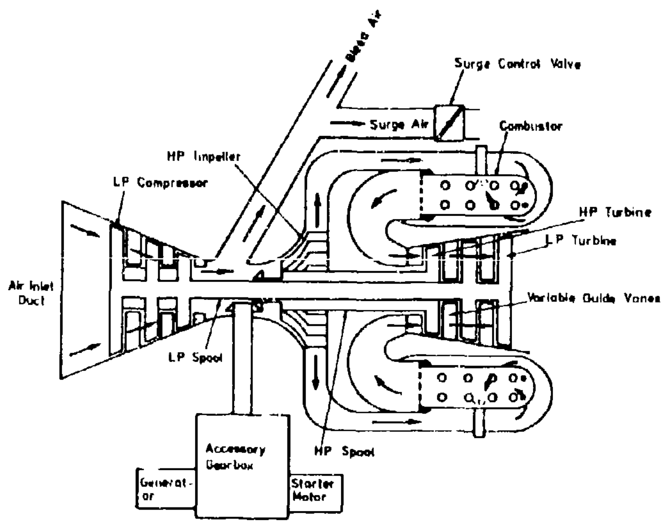
A comparison of the control features of the twin-shaft design with those of the single-shaft design does not show any fundamental differences, since the gas generator, due to its design, varies its speed as a function of the load on the constant-speed operated power turbine. Therefore, any control function is restricted to fuel injection metering with this APU type. As the rate of airflow through the twin-spool APU is a function of the output demand, operation of this APU design is more economical than that of the single-shaft APU under partial load conditions. Hence, it is the load spectrum of the future APU for commercial transport aircraft that will considerably influence APU design and construction. If the design can be based on a narrow-load spectrum, then the future APU will be of the single-spool, direct-driven load compressor type. If, however, the design of future APU's is defined by a wide load spectrum, then either the twin-spool type or the variable speed single-spool APU, with additional IDG or VSCF, will be employed.

The final decisions will thus be based on achieving a balance between the three major considerations, fuel consumption, weight and reliability.



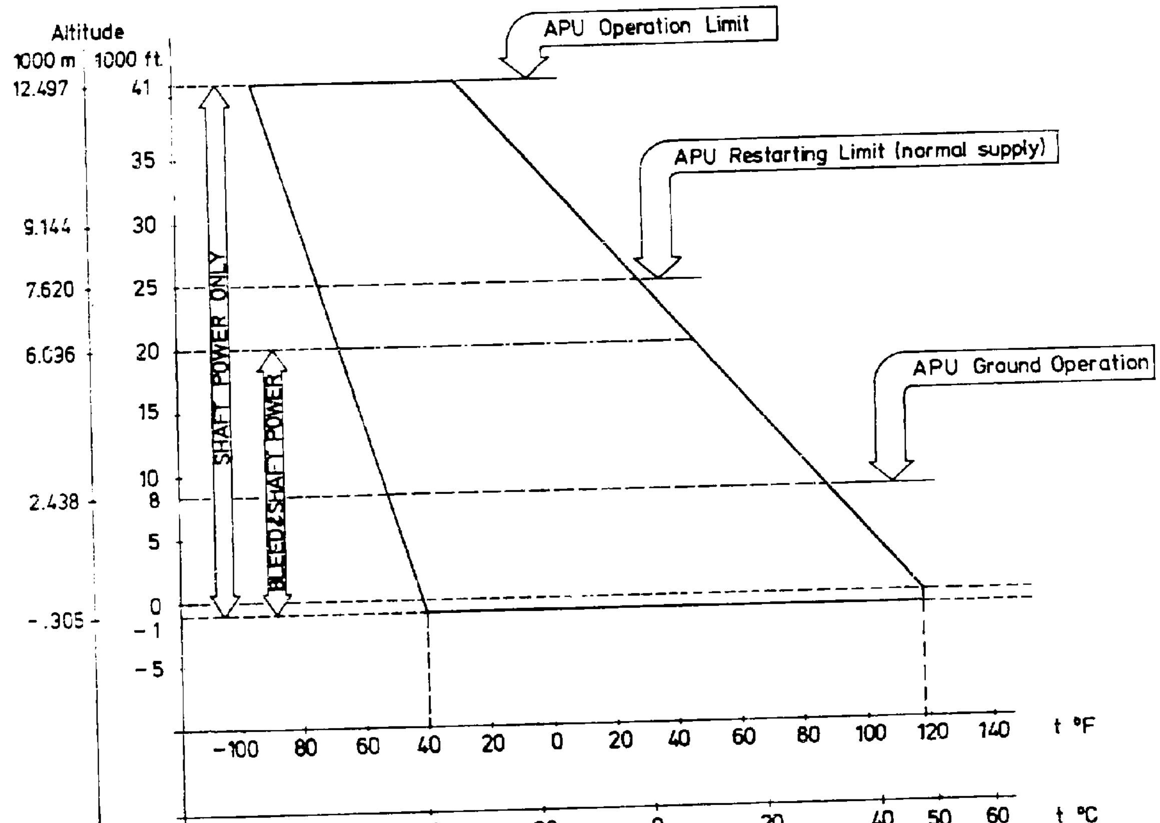
INTEGRAL BLEED APU

FIG.1



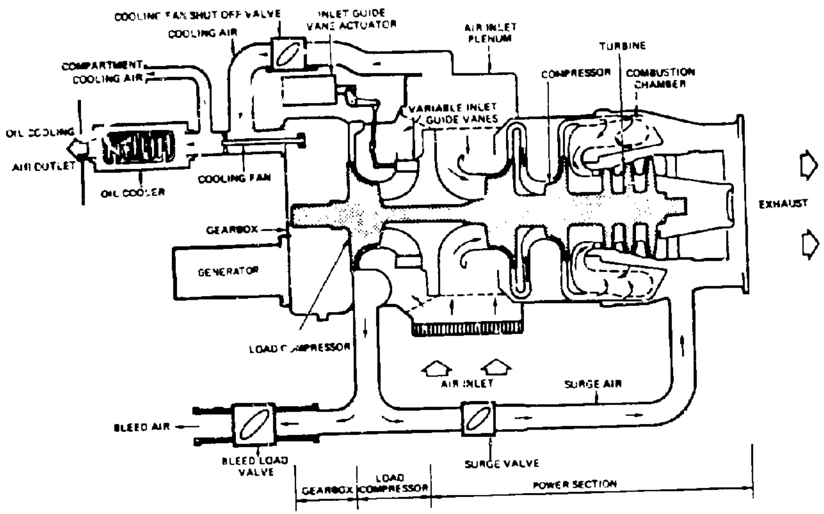
INTERSTAGE BLEED APU

Fig. 2



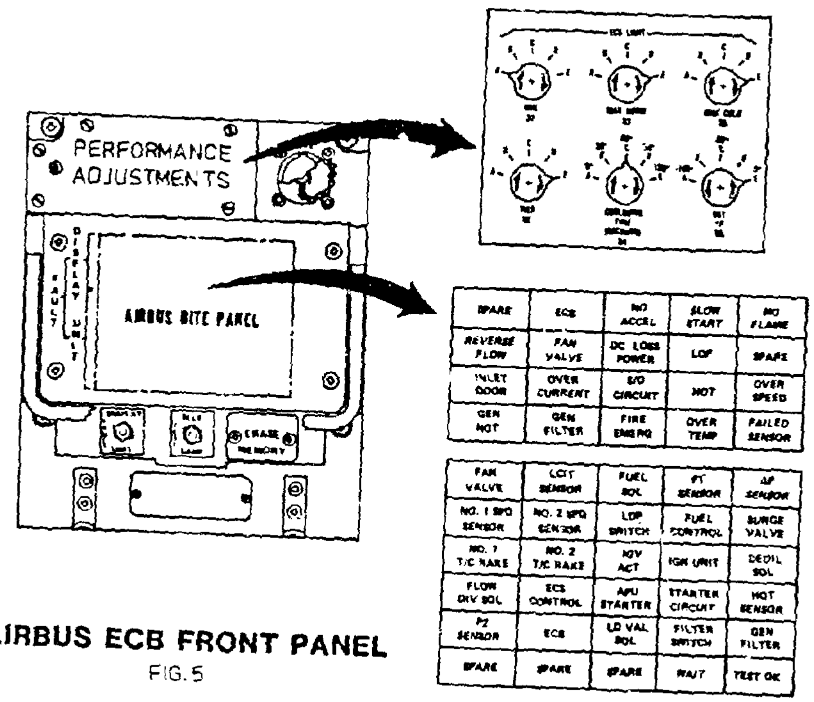
APU OPERATING ENVELOPE

Fig. 3



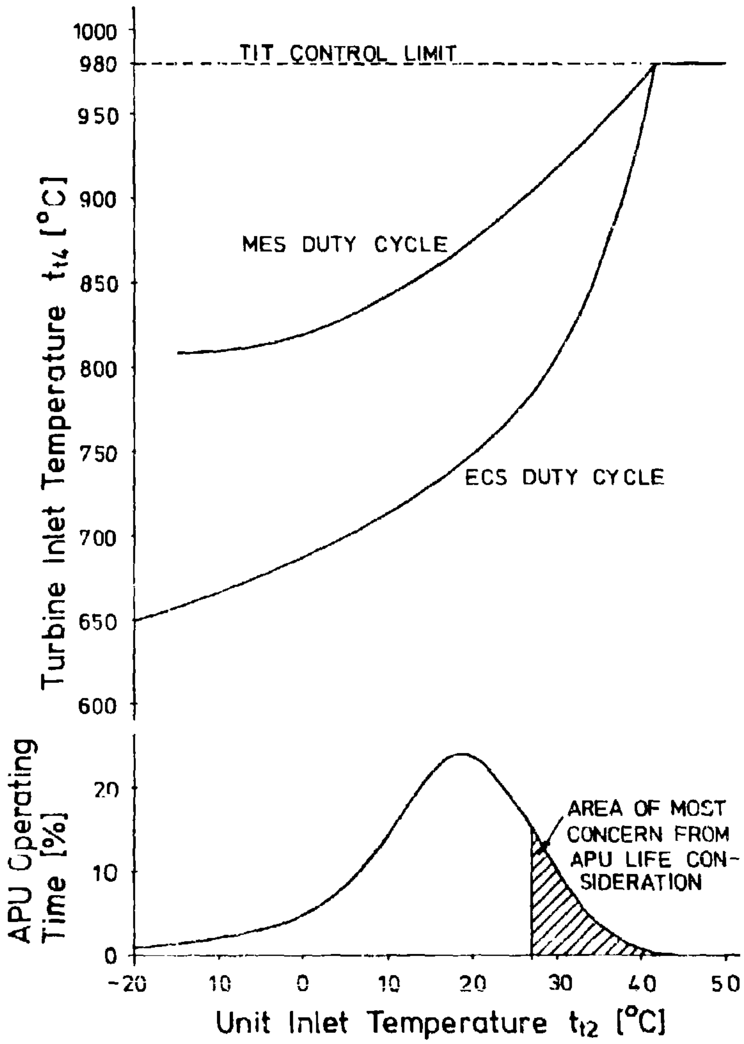
NEW GENERATION APU

FIG. 4



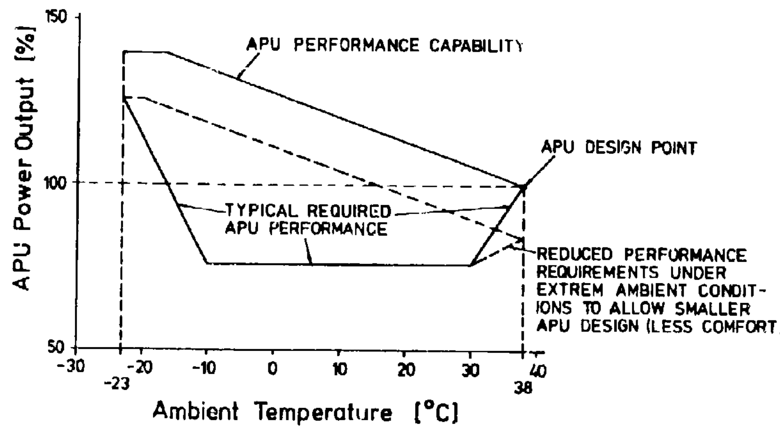
AIRBUS ECB FRONT PANEL

FIG. 5



TYPICAL APU DUTY CYCLE

Fig. 6



TYPICAL APU LOADING IN CABIN CONDITIONING MODE

Fig. 7

Starting Analysis

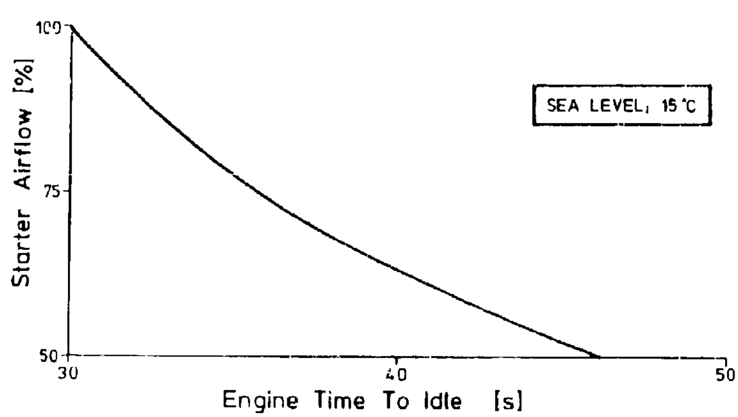
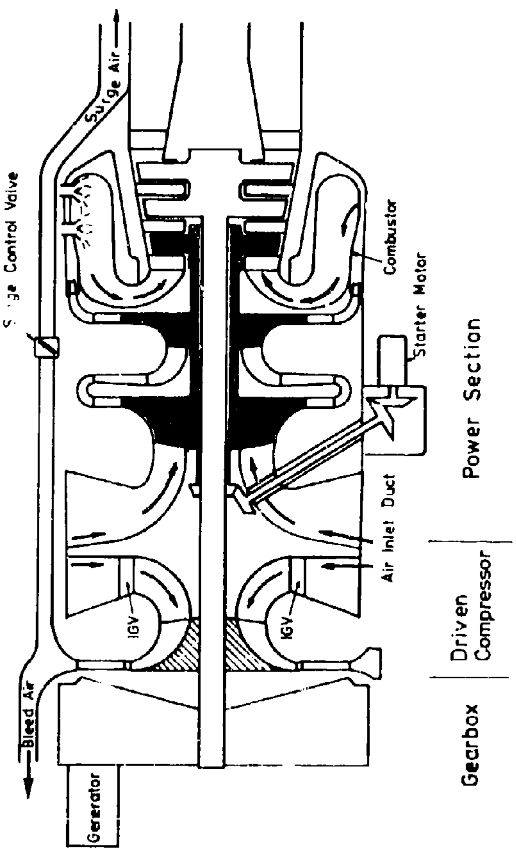


Fig. 8



NEW TWO SPOOL APU
Fig. 9

DISCUSSION

E.H.Warne, UK

What percentage change in airflow rating of the load compressor is achieved by operation of the inlet guide vanes over their full range of movement?

Author's Reply

The turn-down ratio will be approximately 3.5, that means from full open (280 lbs/min) to closed position of the IGVs (80 lbs/min). IGV angles:
θ full open = 13°
θ full closed = 48°

C.Rodgers, US

What will passenger reaction be to reduced comfort as they have small tolerance range?

Cannot you strictly trade start power with start time since starter torque and engine resisting torque are nonlinear?

Author's Reply

We think that passengers will hardly notice this if we increase passenger compartment temperature at hot day by about two degrees centigrade in combination with a higher use of recirculation air and improved compartment air ventilation.

It is correct that starter torque and engine resisting torque are not linear and therefore the benefit for the APU start period is not overwhelming.

M.Eglem, Fr

In order to reduce the APU's SFC at partial power a free turbine gas generator should be used. Electric energy then would be made by an A/C generator with VSCF to give "clean" 400 Hz and a variable frequency A/C generator for other users (frequency moving from 60 to 100%).

Can this solution be used on an aircraft?

What is the percentage of the electric power needing a fixed frequency?

Author's Reply

The reduction of the APU's SFC was mainly mentioned in connection with minimizing the APU operating costs by:
- improved single spool concepts using the electronic digital controls
- conversion/component efficiency
- increased cycle efficiency by increasing inlet temperatures and pressure ratios
- IDG or VJCF systems used on single APU's could improve the efficiency under partial load conditions.

Variable generator efficiency beyond certain limits (400 Hz ± 1% at normal steady state conditions) is not acceptable for the aircraft consumers.

GROUND AND INFLIGHT OPERATIONAL EFFECTS OF APU's

by

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Germany

SUMMARY

The Airbus-APU has to deliver well defined quantities of bleed air and electrical power under two very different conditions : on-ground and inflight. This requirement may imply a compromise. The on-ground operation is essentially governed by problems such as ~~like e.g.~~ noise-generation and -emission , ~~and~~ or ingestion of hot gas which constrains ~~constraining~~ the possible range of intake and exhaust positions. The inflight operation is dominated by the external fuselage flow conditions, intake and exhaust are exposed to, ~~and which~~ these may affect ~~effect~~ significantly the installation pressure-ratio and, with that, the inflight restart envelope and inflight performance. Depending on fuselage surface pressures and boundary layer conditions, intake and exhaust-geometries and -ducting have to be designed ~~such~~ so that a favourable IPR is provided during APU starting as well as APU-operation, keeping at the same time negative effects such as ~~on the aircraft as for instance~~ drag-increase at a minimum. The effect of these environmental conditions on the APU-performance is discussed.

1. INTRODUCTION

Since the on ground operation of an APU is very simular to that of a well known land based shaft power plant, the main attention is paid to the inflight operation. Although the APU is only an auxiliary onboard shaftpower generator and not a main engine, it is consuming fuel, demands space, reduces payload capacity and contributes to the costs of ownership. For this reasons the APU has to be sized such that it meets and only meets exactly the power demanded by the aircraft systems. Consequently a well aimed prediction of installed APU-performance, which is available under the governing fluiddynamic environmental conditions is necessary. The intention of this paper is to discuss mainly on the basis of A310-APU-characteristics some of these environmental conditions APU-intake and exhaust are exposed to essentially, when the aircraft is in-flight, and which may effect the APU inflight relight capability and the performance.

2. CONFIGURATION OF AN AIRBUS-APU

A typical Airbus-APU-configuration is presented in a cross sectional view in Fig. 1 . It is a single shaft APU that provides pneumatic and electric power for the aircraft. Pneumatic energy is provided by means of a load-compressor; electric power is provided by an airframe-furnished generator, that mounts on the APU gear box.

The APU is a modular design comprizing a power section module, load compressor module and an accessory gear box. The load compressor is driven directly by the power section with a single shaft arrangement. The load compressor and power section share a common inlet.

Centrifugal compressor impellers are used in the load section (one stage) and in the power section (two stages); the airflow through the load compressor is controlled by variable inlet guide vanes satisfying the variable bleed demand when the engine is running at constant speed due to generator constant frequency demand. A remotely (in the aft cabin) located ECB (Electronic Control Box) provides full authority digital APU control from start initiation throughout acceleration to governed speed.

3. LOCATION OF APU ON BOARD THE AIRCRAFT

Similar to most of the modern aircraft, the Airbus-APU is installed in the tailcone compartment behind fin and tailplane. Although as shown in Fig. 2 this position is rather remote with respect to the APU-supplied costumers such as e.g. main engines and aircondition-packs , there are a lot of reasons for this particular installation such as e.g. available space, noise emission, pollution by exhausted gas, the risk of uncontrolled APU fire and uncontained disc explosions.

The APU-intake position use to be on the lower side of the fuselage as shown in Fig. 3 In all Airbus installations whereas the latest aircraft of the Boeing-family have the intake on the upper side between fin and tailplane, Fig. 4 . Actually these are the two relevant positions, if the APU is decided to be installed in the tailcone.

The hot gas usually is exhausted through the tip of the tailcone.

4. SOME MAIN OPERATIONAL FEATURES

The Airbus APU's are designed to operate on ground and inflight. The characteristic operational demands are exemplified by means of Airbus A310 and summarized in Fig. 5 .

In case of a main engine generator failure on ground before take off, the A310-APU is requested to be an essential unit in order to increase the dispatch reliability of the aircraft. In this case it has to supply electrical power with a priority over bleed supply in the total range up to 30.000 ft.

Since the APU essentially is operated during turn arounds when passengers are boarding and the ground crew is servicing the aircraft, the noise emitted by the APU is an important feature. During ground operation, the installed APU should meet the noise levels as defined by ICAO, Annex 16 :

> An APU installation (when tested under certain conditions) should not exceed the noise levels at the following points, Fig. 6 :
>
> a. fixed servicing points at which ground personal normally is working for extensive periods during turn-arounds such as cargo doors, passenger doors and re- fuelling points (max. 85dB(A)) and
>
> b. any point, 1.2 m (4ft) above the ground on the outer perimeter of the rectangular pattern described in Fig. 6 (max. 90 dB(A)).

Usually the compliance with point a. imposes severe constraints on the potential positions and on the ducting of intake and exhaust and furthermore requires a considerable amount of noise-suppressing measures, which unfortunately in the most cases are compromising the aerodynamic design. Fig. 7 characterizes some contributing difficulties participating in the decision upper- or lower intake-position and Fig. 8 (a to c) illustrates some fluiddynamic round abouts which may in certain circumstances be necessary to suppress the noise emitted by the engine itself in order to comply with acoustic requirements.

Further contradictory requirements arise from the fact that on ram devices (flaps, scoops etc.) in general sharp edges are not avoidable in order to produce the necessary inflight IPR, whereas the APU- on ground operation requires a kind of flush intake having large radii and no vortex-i.e. noise-generating corners, edges etc.. Also here a compromise has to be accepted.

5. SUMMARY OF SOME RELEVANT ASPECTS OF GAS TURBINE CYCLE THERMODYNAMICS AND FLUIDDYNAMICS

In order to prepare the considerations in the next chapter, some APU-relevant thermodynamic aspects are summarized.

The gas turbine cycles may be devided into two large groups :

- shaft power cycles and
- aircraft propulsion cycles.

An important distinction between the two groups arises from the fact that the performance of aircraft (main engine) propulsion cycles depends very significantly upon forward speed and altitude. Normally these two variables do not enter into performance calculations for the land based shaft power plant.

An APU appears to be something like a hybrid between these two groups : Prevailing, the APU is operating as a land based shaft power plant, because in essence it has to provide auxiliary shaft power when the A/C is on ground.

For the remaining part of its operating time it has to serve under different conditions not on ground but inflight, profiting in some types of installation by ram pressure provided by the forward speed of the aircraft and suffering in other types of installation from adverse installation pressures due to unfavourable static pressures and boundary layer conditions APU-intake and -exhaust are exposed to . It will be the main concern of this paper to discuss this sort of inflight phenomena, exemplified by Airbus APU-installations and to relate them to the APU-cycle thermodynamics and further to inflight relight capability and performance.

5.1 ENERGY-CONVERSION IN THE APU-INTAKE

At static conditions or at very low forward aircraft speeds, i.e. when the APU is working like a land based powerplant, the intake, Fig. 9 is acting as a nozzle, in which the ambient air accelerates from zero velocity or low C_a to a velocity C_1 at the compressor inlet duct. The enthalpy-entropy (h,s)-diagram elucidates that the energy which is necessary to accelerate the ambient air from zero to $C_1 > 0$ at the engine face is extracted from the compressor. All these events are similar at the aircraft's main engine.

At normal forward speeds C_a, however, the intake face, Fig. 9 performs as a diffusor with the air decelerating from C_a to C_1 and the static pressure rising from the local ambient pressure p_a to p_1 at the intake duct via the external diffusor forming in terms of streamlines around the intake, i.e. kinetic energy from the ambient flow field on the aircraft skin (originating from the forward speed) is converted into static pressure p_1, which can be added to the compress... pressure rise and can be regarded as a benefit to the APU-cycle, as shown in the (h,s)-diagram. Opposit... main engine which has to pay for this kind of kinetik energy by a part of it's own thrust, the APU (wh... ...duces inboard shaft power only) takes profit from the airspeed and penalizes at the same time the main engine by the amount of extracted kinetik energy. In a very extreme (but of course also very unrealistic) extension of this kind of kinetic energy extraction, the APU compressor would be substituted and with that the turbine power in total would become free shaft power (This situation is given for the ram air turbine located in the lower wing root and extended for emergency supply of hydraulic systems).

Since it is the stagnation pressure p_{t1} at the compressor inlet which normally is required for cycle calculations, it is the pressure rise $(p_{t1} - p_a)$ which is of interest and which is referred to as the 'ram pressure rise'. Usually a high ram pressure rise is expected to be beneficial to the cycle performance because it contributes to the static pressure ri.e done by the compressor. But this is only true, if the stagnation (ram) device is designed such that it recovers pressure and delivers at the same time that amount of air flow which is demanded by the APU.

Fig. 10 is given to remember the energy conversion i.e. the displacement work $\int p(V)dV$ in the elementary thermodynamic cycle. The pressure in the term $p(V) \cdot dV$ is a static and not a total pressure by definition of thermodynamic laws . Fig. 11 is given to relate this fundamental thermodynamic context to the conversion of kinetic energy into static pressure (ram pressure rise), what in the subsonic flow regime only can be done by a diffusor and what has to be done by an intake ram device. In this figure, the various stagnation streamlines ahead of the ram flap, (the first line upon the fuselage skin, and the last one somewhere in the ambient fluid-volume), occur successively during the starting procedure of the APU, or to some extend also during variation in bleed demand of the load compressor, while the APU is running at constant speed.

At zero APU-RPM (lowest stagnation streamline) the full ram (stagnation) pressure is offered to the APU-inlet duct, but the volume-flow $\dot{V}$, which is taken into the inlet is zero. With increasing RPM the demanded $\dot{V}$ increases, and the stagnation streamline moves away from the skin, at low RPM still forming a diffusing stream tube and providing a rise in static pressure, which is beneficial to the thermodynamic cycle because of the external compression work supporting the engine compressor. At higher RPM, the demanded volume flow further increases, and the stagnation streamlines change over to form a nozzle flow associated with a drop in static pressure, which in turn penalizes the cycle. It is the APU inflight starting capability, and once the APU is running also the performance, which is highly dependant on these inlet characteristics.

5.2 ENERGY-CONVERSION IN THE APU EXHAUST SYSTEM

As mentioned before, the APU is designed to produce only shaft power and not a jet like the main engine of the aircraft i.e. any kinetic energy which remains in the gas exhausted by the APU is wasted.

Usually in an industrial shaft power plant, the gas immediately leaving the turbine is recovered in an exhaust diffuser which in effect increases the pressure ratio across the turbine and with that the work done by the turbine. Fig. 12 exemplifies this for an exhaust diffuser which reduces the final velocity to a negligible value, so that the diffuser discharge pressure p_D is nearly equal to the ambient pressure p_B. Ref. 1, gives evidence of the beneficial potential of an exhaust diffuser.

From this short discussion also follows that e.g. an exhaust pipe which may have a certain lenght necessary for noise suppression acts like a nozzle and raises the turbine's back pressure which in turn is followed by a degradation of turbine power.

5.3 PERFORMANCE EFFECTING PARAMETERS

In addition to the APU-intake and exhaust pressure situation, the APU-performance is affected by the intake flow swirl, which in the physical sence is an angular momentum as exemplified in Fig. 13 a, depending on the sense of rotation, this angular momentum increases or decreases the amount of kinetic energy converted in the first compressor rotor into static pressure as exemplified in Fig. 13b by means of the velocity-triangles. Thus a benefit or a penalty to the thermodynamic cycle has to be expected. The origin of the swirl is either an asymetric inlet geometry, as e.g. on the A310 inlet or any asymetric flow condition across the intake face, such as e.g. a velocity gradient or non zero crossflow component. Fig. 14 exemplifies the consequences of the slightly asymetric arrangement of the A310 intake. During on ground operation, there occurs a non zero swirl, which in this case benefits the cycle, because the APU happens to perform optimal at non zero swirl.

For completeness annother potential source of performance degradation is mentioned : the inlet distortion which describes any inhomogeneous distribution of the cross sectional flow properties (in most cases the total pressure) at the engine face und which penalizes the cycle via the mechanism demonstrated by the velocity-triangles given in Fig. 13 b.

6. FLUIDDYNAMIC ENVIRONMENTAL CONDITIONS

Based upon the relevant thermodynamic aspects of gas turbine cycles, summarized in the previous chapter the impact of fluiddynamic environmental conditions which are associated with an airborne aircraft and which affect the APU-cycle will be discussed in the following chapters.

6.1 FUSELAGE PRESSURE DISRTIBUTION

The fuselage of an airborne aircraft is similiar to the wing covered by a certain pressure field. A calculation by e.g. a fully 3-dim. panel-method on a complete aircraft, whose tail-geometry and surface panel distribution are shown in Fig.15 gives access to details of this pressure distribution, which is plotted in terms of C_p - isobars in Fig. 16 for the upper and the lower side of the rear fuselage for a typical cruise condition of M = 0.8, α = 0.6,. The C_p pattern on the upper side of the rear fuselage, where a APU intake may be located, is rather complex and is mainly dominated by the pressure field of fin and tailplane, both inducing strong pressure gradients. Steeper gradients will occure, when e.g. in the 2. segment climb after take off or in the descent a high tailplane load is required and the tailplane is set at negative trim angles. Or, in case of one engine out (which is among others one case where the APU is expected to perform satisfactorily) a distinct rudder deflection may cause steeper or even fluctuating pressure gradients. In addition, the entire flow level in the channel formed by fin, fuselage and tailplane is rather high, and with that the local static pressures are relatively low, penalizing the compressor's work due to reasons explained in the chapter before.

The situation is different and more favourable to the compressor on the lower side of the rear fuselage, where due to the upsweep of the fuselage the velocity level is considerably lower, the static pressure higher and where gradients are more moderate and less sensitive to geometrical variations of the empennage.

In order to keep the aircraft drag at it's minimum, the afterbody-aerodynamicist aims to design the aircraft tail such, that the static pressure at the base of the tailcone, where the exhaust pipe uses to be located, is as high as possible. In the theoretical case of no energy dissipation within the flow along the fuselage, the pressure at the base of the tailcone would recover to the full stagnation pressure, which also occurs at the aircraft nose.

From this isobar pattern it becomes obvious, that the APU on an aircraft inflight is expected to operate under adverse pressure conditions with the unfavourable thermodynamic consequences illustrated in Fig. 17 , followed usually if no counter measure is taken by reverse APU-windmilling and reduction in performance and relight capability.

Considering only this particular situation it appears to be usefull to turn the APU by 180^{o} and let it breathe through the exhaust pipe, taking profit from the high base pressure, the aerodynamicist attemps to verify. This 180^{o} - turn improves the thermodynamic cycle as sketched for comparison in Fig.17 . Of course it is rather obvious that there are a lot of other good reasons to abstain from a 180^{o} turn, such as e.g. exhaust noise, hot gas ingestion etc.. With the APU-intake in the normal forward facing position other well known means are necessary (and of course widely used) such as stagnation-noses, ram-flaps, scoopes etc.,Fig. 18, which are designed to catch the approaching surface flow and convert part of the kinetic energy into static pressure in order to compensate the unfavourable low static pressures originating from the flow around the airframe geometry.

Of course there is a strong limitation imposed on all pressure rising ram devices : They all have to become invisible, when the A/C is inflight and the APU is not operative in order not to increase the aircraft-drag. This requirement leads to moveable flaps, scoopes etc. which have to be controlled by the APU-start stop-switch, and whose geometrical design becomes difficult because of strong acoustic requirements during APU ground operation.

6.2 FUSELAGE BOUNDARY LAYER

As discussed in the chapter before, the afterbody of the aircraft is covered by a certain pressure distribution. Fig. 16 indicates an extended low pressure area on the lower, convex side of the fuselage, which in addition is augmented by the lower side (suction side) of the tailplane. This low pressure area attracts all the low energy boundary layer material accumulated in the flow starting from the aircraft nose and removes at the same time part of the boundary layer material from the upper side of the fuselage. A photo recorded during watertunnel investigations, Fig. 19 illustrates these events.

Boundary layer measurements in a windtunnel by means of pitot-rakes in the vicinity of potential intake positions on the upper and lower side of the fuselage, Fig. 20 , confirm this special character of 3-dim. boundary layer flow and indicate a higher deficit in total pressure on the lower side than on the upper side. Flight test results from A300 and B707 verify the energy-deficit on the lower side. Even in one meter vertical distance from the aircraft skin the total pressure ratio is still well below the value 1.0. The amount of kinetic energy, which is dissipated within the boundary layer at the location of the lower intake position and which is dependant predominately on the flight machnumber is indicated by Fig. 21 which presents A310 flight test results recorded at different machnumbers, altitudes and APU-operation modes.

7. CONSEQUENCES OF ENVIRONMENTAL CONDITIONS TO THE APU-CYCLE

The fluiddynamic environmental conditions during aircraft inflight, which are discussed in chapter 6 may be entered into the enthalpy-entropy-diagram, which is normally used for cycle considerations. To prepare this, Fig. 22 shows streamwise pressure distributions extracted from Fig.16 for a streamline on the upper and the lower side of the fuselage respectively.

A discrete fluid volume approaching the aircraft nose, and floating along the fuselage to the tip of the tail, is submitted to several displacement workes(as exemplified in Fig. 10) which are different, depending on the pressure distribution along the upper or the lower surface until the volume passes the upper or the lower intake position and finally reaches the tip of the tail.

In addition to the different pressures, the fluid volume experiences on its way a certain energy-dissipation i.e. a loss in stagnation pressure, which is usually expressed in terms of boundary layer total pressure ratio profiles or velocity profiles as exemplified in Fig. 20 . For the reasons discussed in chapter 6.2, the loss in total pressure is higher on the lower side, than on the upper side.

Since no work is done to or extracted from the fuselage flow, the flow may considered to be isoenergetic with the total enthalpy $h_{t_\infty} = h_\infty + c_\infty^2/2$ = const., determined by the speed of the aircraft c_∞, by the static temperature and by the static pressure of the ambient air at that particular flight level. Thus all changes in gasdynamic properties due to the varling pressure along the fuselage and due to the dissipation of kinetic energy within the boundary layer may be entered into the (h, s)-diagram what is done only qualitatively for exemplification in Fig. 23 with the total enthalpy h_{t_∞} as the upper limit for the total available energy. In order to maintain a rest of clearness, the energy contributed by the APU-starter motor and the fuel burned in the APU are neglected in the Fig. as well as is the kinetic energy in the exhausted gas. Thus the (h,s) -diagram comprizes only the thermodynamics of the fuselage flow.

Some special stations on the fuselage are labeled by the numbers 1 to 7 and are related to the corresponding stations in the (h,s)-diagram labeled in the same sense. Depending on the amount of the local dissipation within the boundary layer, the state-connecting lines are more or less steep, indicating a change in gasdynamic properties, which is more or less away from being isentropic and of course different between the upper and the lower side of the fuselage.

The kinetic energy c_1^2 / 2 which is plotted as a band below the line h_{t_∞} = const. is not available for conversion into static pressure, because after the ram pressure rise is completed an intake flow velocity c_1 has to be maintained in order to provide the volume flow demanded by the APU e.g. at the particular RPM's during APU-starting, as has been dicussed in chapter 5.1.

The sequential change in gasdynamic properties entered into the (h,s)-diagram indicates, that in order to provide a favourable installation pressure situation any ram device feeding an upper or an low r intake and providing an intake flow velocity c_1 at the same time has to perform with a better isentropic diffussion than is inherent in the diffusing fuselage flow moving from the upper or the lower intake position downstream to the base of the tailcone. Since the aerodynamicist are doing a tremendous effort to reduce the aircraft drag, a more efficient diffusion on the afterbody and with that a trend towards a higher base pressure has to be envisaged, what in turn requires better i.e. more carefully designed and more efficient ram devices.

Depending on the overall character of the fuselage pressure and boundary layer distribution certain constellations may occure (which easily can be verified in the (h,s)-diagram e.g. by a higher base pressure) where it is not possible any more to rise the static intake pressure by means of any ram device above or even only to the value of the base pressure and to maintain a certain velocity c_1 at the same time. In this case a reduction in relight altitude is incurred, and of course, once relight was successfull in a lower altitude, also a penalty in performance.

During the APU-starting procedure, the most critical phase occurs between start initation and ignition, because in this timespace the APU-starter motor is the only energy supplier, which has to accelerate the rotating parts of the engine and in case of an insufficientely performing intake also has to provide the energy which is necessary to accelerate the ambient air into the intake. Once ignition was successfull, for further acceleration there normally is enough energy available delivered by the fuel, which starts to be burned in the APU. Nevertheless, even if the starter cut out speed has been exceeded successfully, an unsuccessfull APU start is still possible. The series of inflight start attempts given in Fig. 24 indicates that critical start conditions occur at high altitudes. Fig. 25 displays two comparable start attemps in similar marginal conditions, where one of which failed. In this Fig. the developement of EGT, RPM and starter current versus time indicates in both cases a comparable acceleration to starter cut out speed, but then, at still similar EGT's in the unsuccessfull case, the turbine torque obviously was insufficient to further speed up the rotor and to further accelerate via the compressor the intake flow corresponding to the increasing volume demand, which in turn is associated with a decreasing static intake pressure as exemplified in Fig. 11 and discussed in chapter 5.1. The decreasing static intake pressure and with that the decreasing contribution of external flow compression work finally results in a penalty to the cycle and in an unsuccessfull start.

If an insufficient relight capability can be related to a lack in energy extraction from the external flow field via the a.m. mechanism (and of course if there are no other reasons as e.g. operating limits of the combustion chamber), the conclusion can be drawn, that in this case also a penalty in performance is incurred. On the other hand a good relight capability indicates that in this particular situation also a benefit to the inflight performance can be expected.

CONCLUSION :

On the thermodynamic-fluiddynamic grounds a variety of aspects is governing the positioning and the design of intake and exhaust of a tailcone installed APU. In the typical Boeing or Airbus input-exhaust arrangement, the natural conditions inherent in the fuselage flow on the A/C inflight create a non cycle supporting environment; the fuselage pressure field by itself tends to move a massflow into the exhaust, furtheron through the APU and finally ejects it through the intake, whereby this massflow is driven by a higher pressure drop when the intake is located on the upper side than it is the case with a lower intake. For this reason, especially the upper intake requires a highly efficient ram device, which converts high kinetic energy into static pressure to that extend, that it at least overrides the back pressure acting at the exhaust pipe. The ram device has to be designed such, that this override capability is conserved from APU-start initiation throughout acceleration until governed speed, if fuselage flow related penalties to the APU-cycle shall be avoided. If necessary, an extended ram device could extract still more energy from the fuselage flow in order to support the cycle such, that a shortfall in performance or relight capability somewhere in the APU operation envelope could be compensated without oversizing the APU and without demanding e.g. two batteries instead of one for the APU-starter supply.

Certainly the amount of energy extraction from the fuselage flow by a ram device is limited by the total enthalpy inherent in the flow and by the fact that there is a boundary layer. At the upper intake location, the unfavourable low static pressures coincide with the presence of a high energetic boundary layer, admitting to a certain extend the compensation of the static pressure deficit by the ram device. At the lower intake location however, the static pressures especially on the A310 afterbody are higher and very close to the base pressure level, thus providing nearly excellent conditions, especially because the static intake pressure is backed up by the large surrounding fluid volume having the same pressure level which is created by the diffusing flow around the A/C-afterbody. For this reason this volume carries a high and stable energy in terms of pressure times volume, thus the static pressure level is more resistant against being 'succed down' as it is the case in the volume just upstream of e.g. a ram flap at the upper intake position, where high APU suction (due to high APU-RPM's) immediately starts to cancel the diffusion-process in front of the ram flap face; depending on the size of the flap, more or less diffusion takes place, thus at higher or lower RPM's (or intake volume flows) the APU-cycle is more or less supported, which finally is an important item for the relight capability, the APU performance and of course also for intake volume flow variations due to variations in bleed demand.

If the pressure level at the lower intake position is to low in comparison to the base pressure,e.g. due to a different afterbody design (as e.g. less upsweep) a worth mentioning compensation by a ram device may become difficult because in the thick lower boundary layer only a rather small amount of kinetic energy is available for conversion into static pressure.A ram device would have to have an unacceptable lenght (because of ground clearance, when the A/C rotates)in order to penetrate into higher energy levels of the boundary layer. In this particular situation measures at the exhaust pipe should be examined, as e.g. a ring diffusor, which exhausts the gas through a circumferential slot into an area with lower pressures upstream of the base. A solution like this also facilitates acoustic damping of turbine noise in connection with exhaust noise reduction (because of lower exhaust gas velocities) and finally provides a potential to reduce exhaust pipe lenght (a certain lenght is required for engine noise damping during ground operation).In this context and also with respect to the thermodynamic cycle it should be mentioned, that in general any noise which is generated by the intake- or the exhaust-flow strongly indicates an origin of energy-dissipation and with that indicates also a poor aerodynamic design. Thus there should be a premise to avoid the noise already in'status nascendi' by a carefull design and not let it arise and then damp it.

These concluding remarks should contribute some support to the decision upper or lower intake position. Besides the constraints imposed by accoustic requirements during ground operation and besides constructual and maintenance aspects, the thermodynamics and fluiddynamics of the fuselage flow and their relation to the APU cycle-thermodynamics gives a clear answer to this question.

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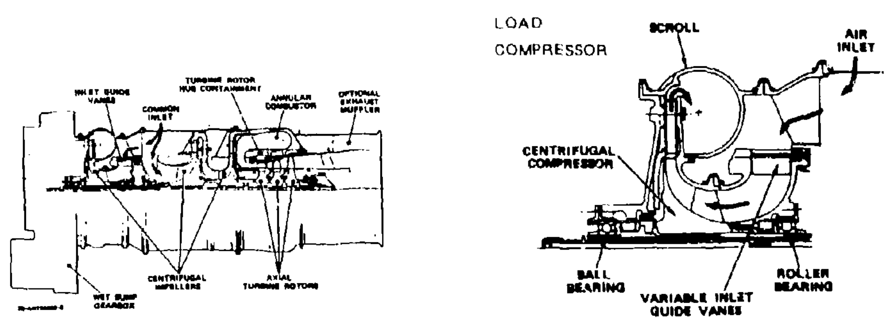

Fig. 1 TYPICAL AIRBUS APU-CONFIGURATION

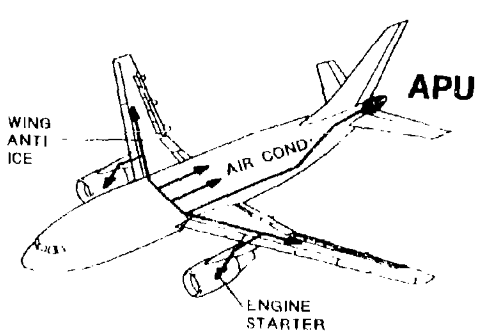

Fig. 2 LOCATION OF APU ON BOARD
THE AIRCRAFT

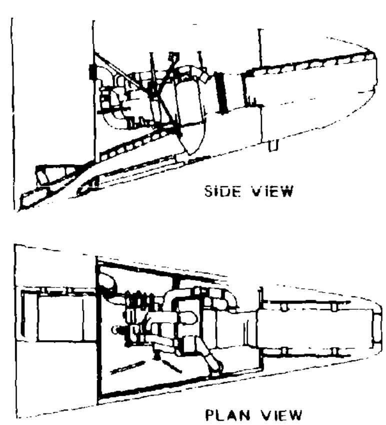

Fig. 3 AIRBUS APU INSTALLATION

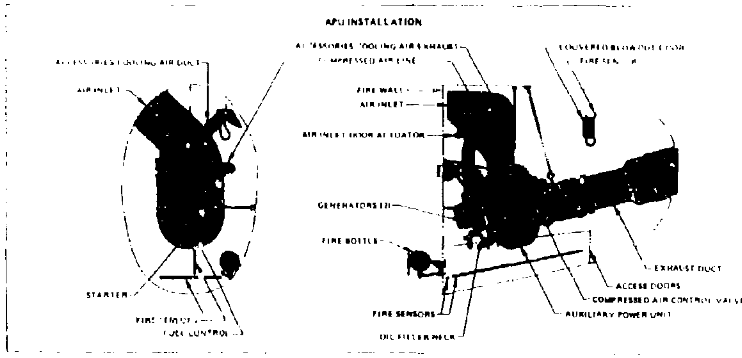

Fig. 4 BOEING 747 APU-INSTALLATION

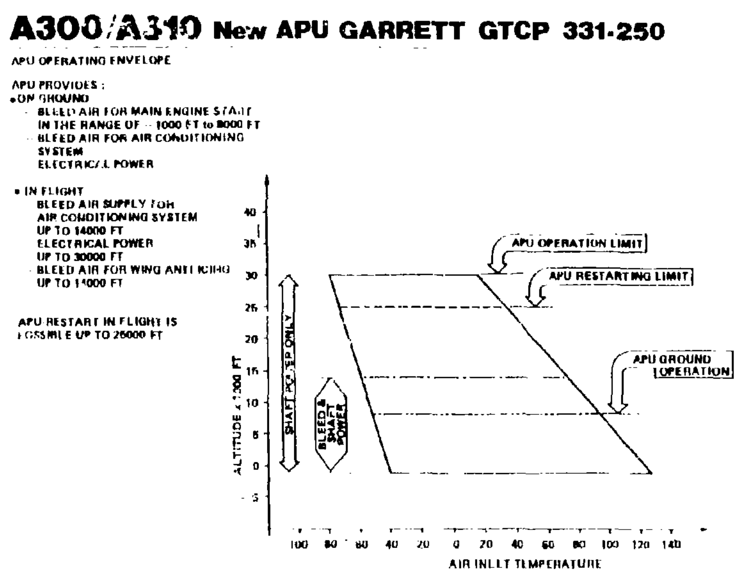

Fig. 5 CHARACTERISTIC OPERATIONAL FEATURES OF AIRBUS APU

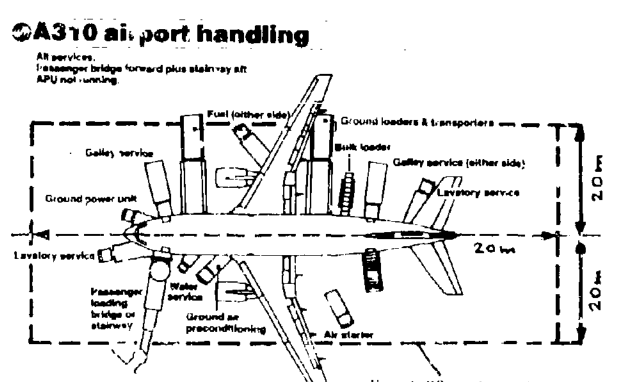

Fig. 6 APU-RELEVANT NOISE LEVEL DEFINITION POINTS

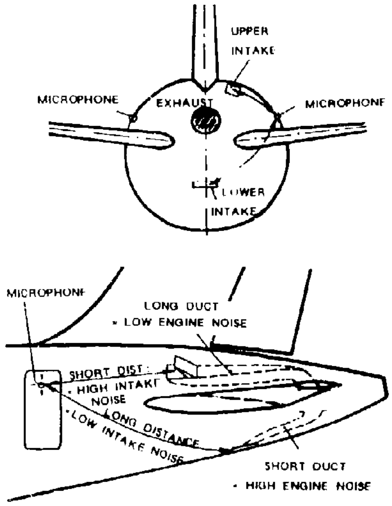

Fig. 7 INTAKE POSITIONS AND NOISE IMMISSION

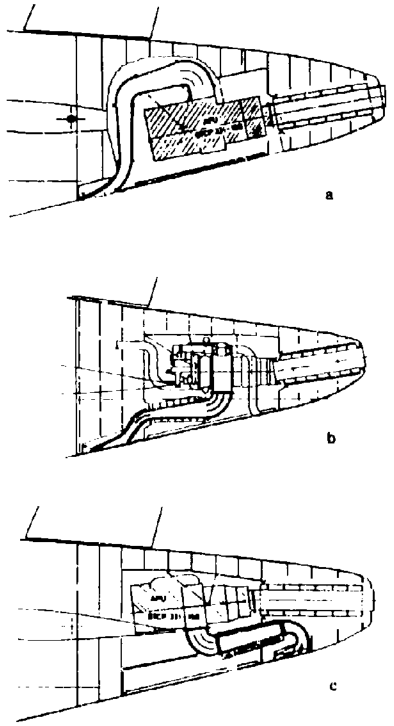

Fig. 8 ACOUSTIC REQUIREMENTS COMPROMIZING THE AERODYNAMIC INTAKE-DUCT DESIGN

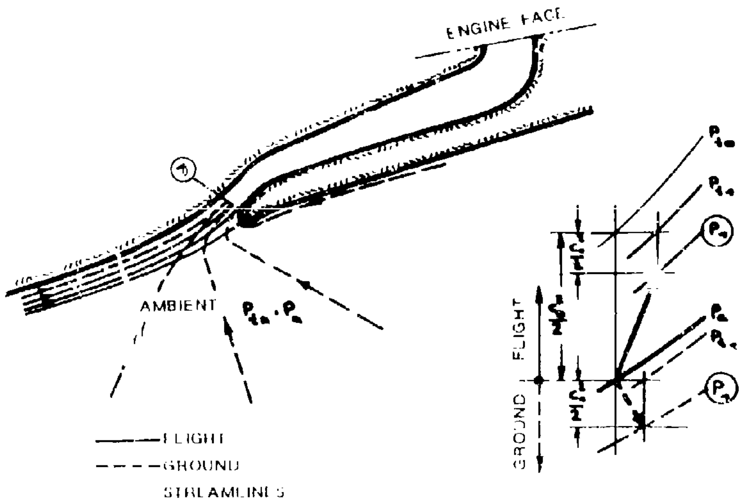

Fig. 9 ENERGY CONVERSION IN THE APU INTAKE ON GROUND AND IN FLIGHT

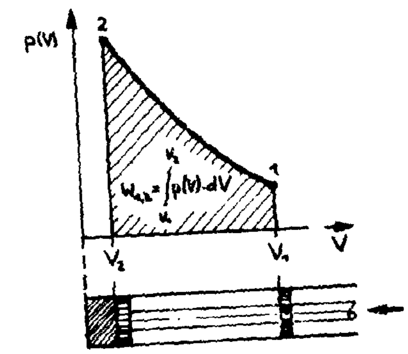

Fig. 10 DISPLACEMENT WORK IN THE ELEMENTARY THERMODYNAMIC CYCLE

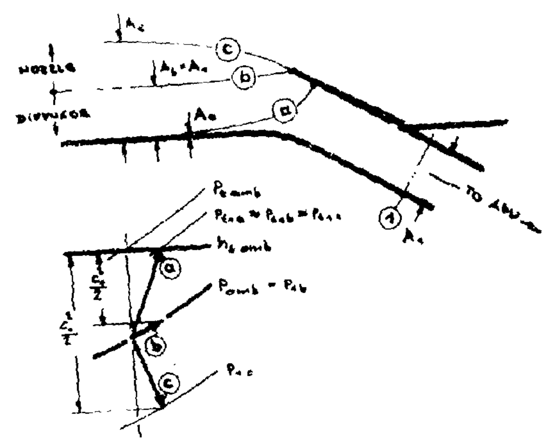

Fig. 11 CONVERSION OF KINETIC ENERGY INTO STATIC PRESSURE BY THE APU-INTAKE

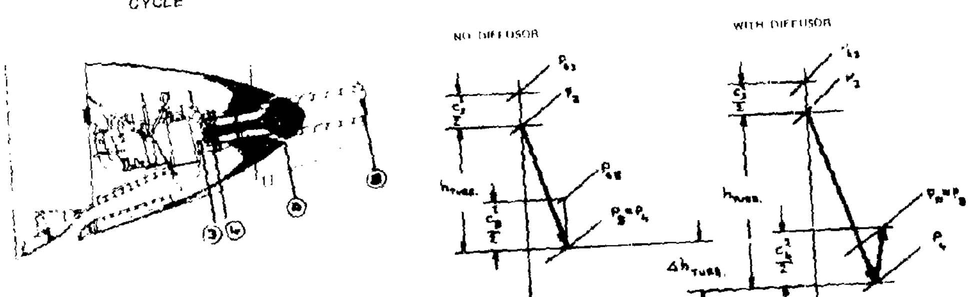

Fig. 12 BENEFIT OF EXHAUST-GAS DIFFUSER TO THE TURBINE's WORK

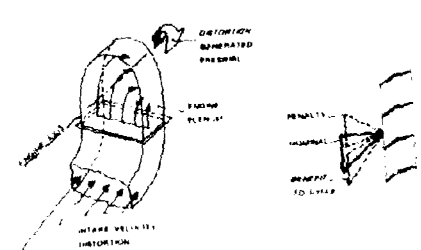

Fig. 13 GENERATION OF PRESWIRL AND CONSEQUENCES TO THE APU CYCLE

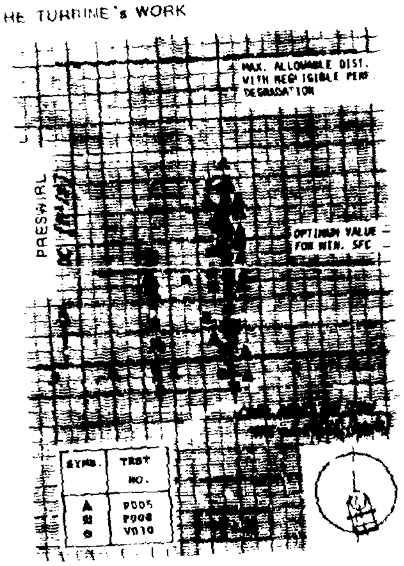

Fig. 14 BENEFICIAL IMPACT OF PRESWIRL TO A310-APU PERFORMANCE (Flight Test Results)

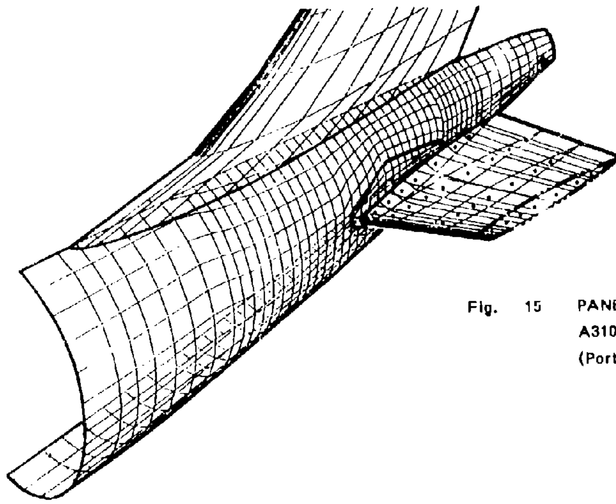

Fig. 15 PANEL DISTRIBUTION ON A310 REAR FUSELAGE (Port side)

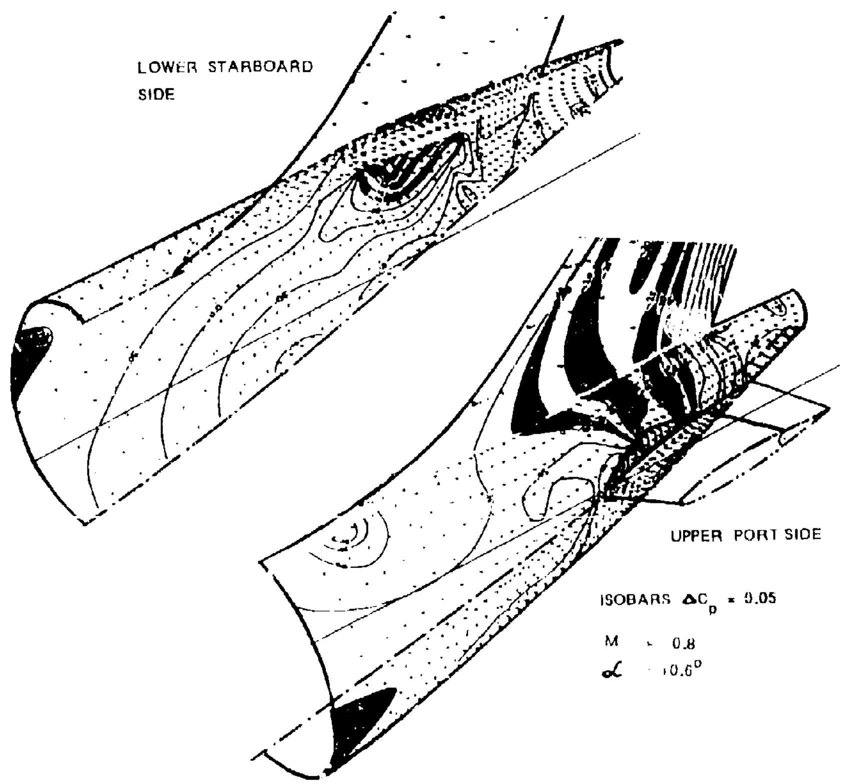

Fig. 16 ISOBARS ON THE REAR FUSELAGE OF A310 (3-D POTENTIAL FLOW CALCULATION)

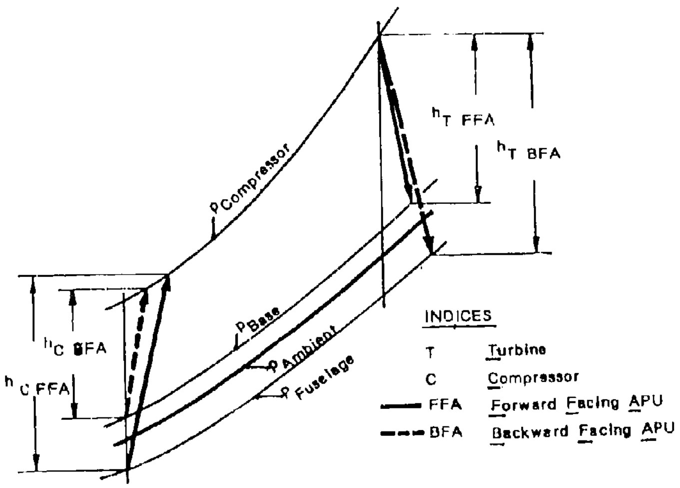

Fig. 17 CYCLES FOR FORWARD AND BACKWARD FACING APU

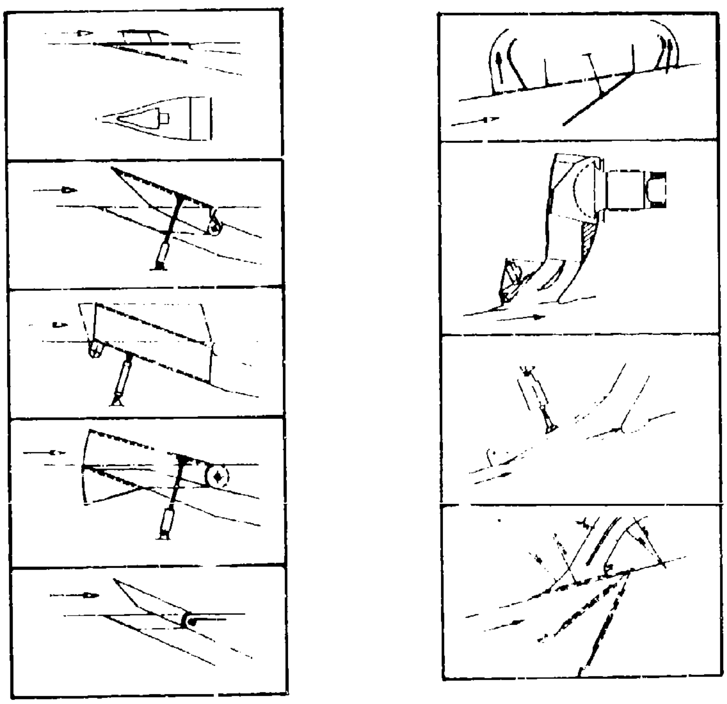

Fig. 18 VARIOUS TYPES OF RAM DEVICES

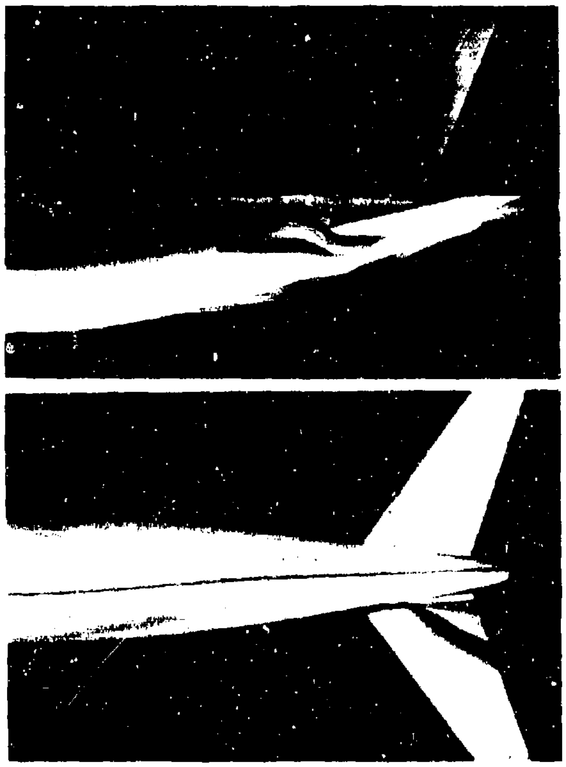

Fig. 19
FLOW FIELD AROUND
A REAR FUSELAGE
(Watertunnel Investigations)

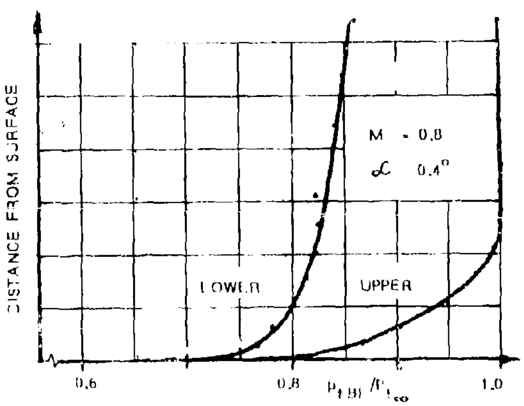

Fig. 20 BOUNDARY LAYER PROFILES AT POTENTIAL INTAKE POSITIONS
(Windtunnel Results)

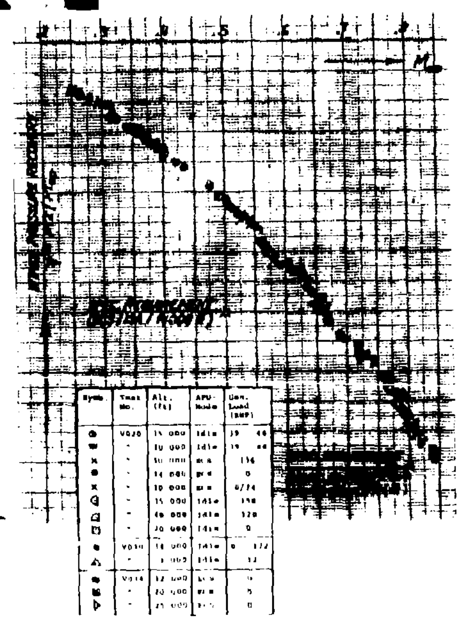

Fig. 21
IMPACT OF BOUNDARY LAYER DEVELOPMENT ON LOWER INTAKE PRESSURE RECOVERY
(A310 Flight Test Results)

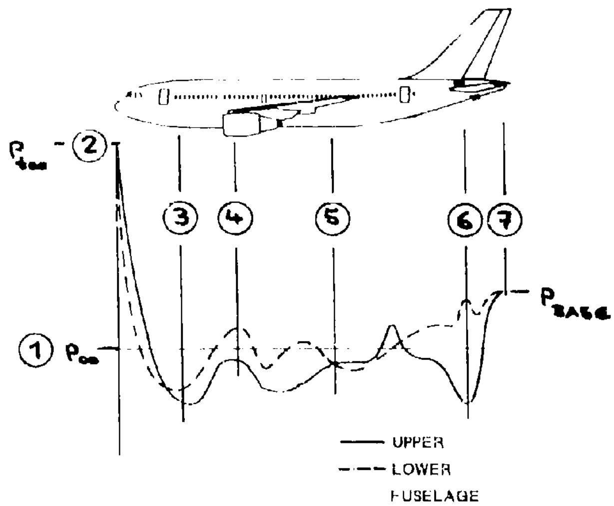

Fig. 22 STREAMWISE PRESSURES ON UPPER AND LOWER SIDE OF THE FUSELAGE

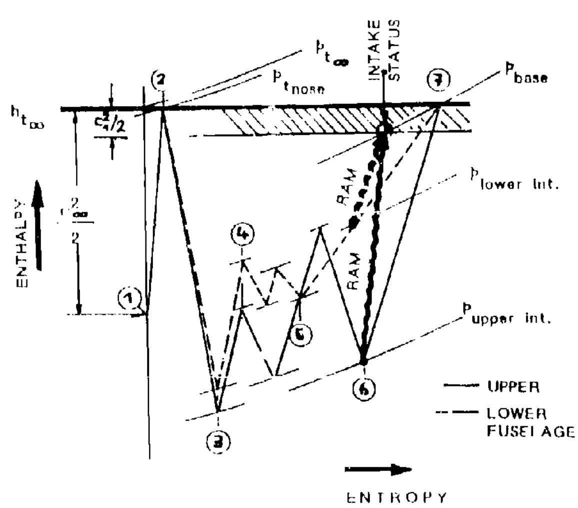

Fig. 23 STREAMWISE GASDYNAMIC PROPERTIES OF FUSELAGE FLOW IN THE ENTHALPY-ENTROPY DIAGRAM

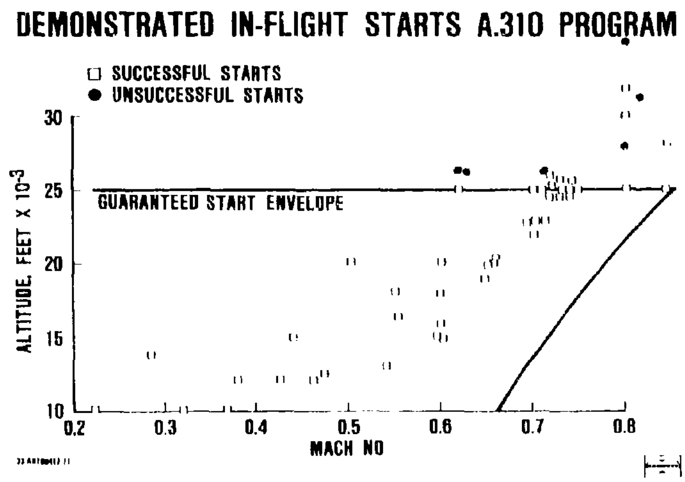

Fig. 24 APU- STARTS AS A FUNCTION OF FLIGHT-MACHNUMBER AND ALTITUDE

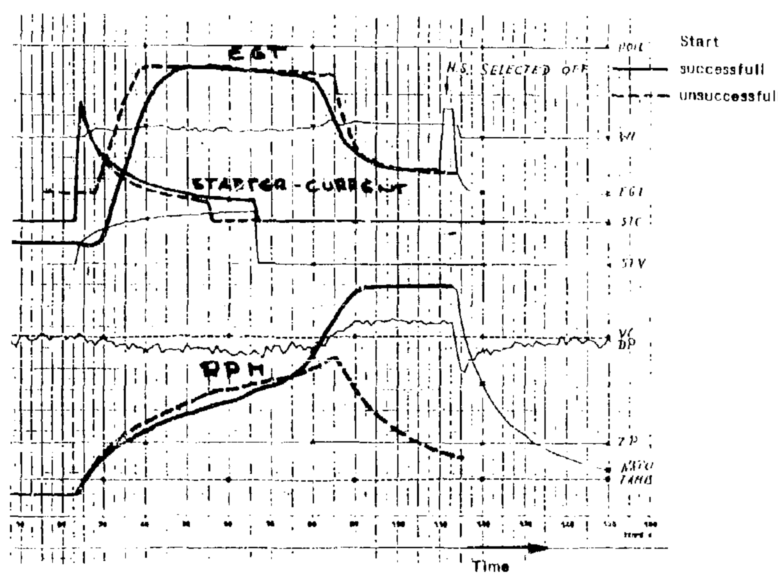

Fig. 25 A SUCCESSFULL AND AN UNSUCCESSFULL APU START IN MARGINAL CONDITIONS

DISCUSSION

I.C.M.Hogenvorst, Ne

1. What is your definition of Relight capability: is this restart after an inflight shut down or an inflight start after unspecified cruise with an APU not operating?
2. Where does this requirement originate (to start the APU at high altitude)?

Author's Reply

1. The relight capability is defined as the ability of an APU to be relit after unspecified cruise within the aircraft-flight-envelope at arbitrary altitude, Mach number and aircraft altitude with an APU not operating.
2. The relight capability and with that the operational readiness of the APU is required to be consistent with the aircraft flight-envelope, such that in a case of emergency, provision of bleed air and electrical power is covered by the APU, an important item with respect to the increasing demand of electrical power for e.g. the avionics and other onboard consumers, and also with respect to the dispatch reliability of the aircraft.

SUPER INTEGRATED POWER UNIT FOR FIGHTER AIRCRAFT

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SUMMARY

An SIPU is a multifunctional aircraft power unit capable of providing: (1) electrical, pneumatic, and hydraulic power for ground maintenance and standby operations; (2) normal and emergency main engine start power; and (3) emergency electrical and hydraulic power. Power is obtained either from jet fuel combustion with air for ground operations and normal engine starts, from a gas generator system using onboard stored propellants for emergency functions, or from the aircraft main engine compressor air for the emergency electrical and hydraulic power function. Rockwell International's Rocketdyne Division conducted a program, sponsored by the Aeropropulsion Laboratory at Wright-Patterson Air Force Base, to design, fabricate, assemble, and test the SIPU concept by operating a demonstrator in all modes. This paper provides a description of the SIPU concept, the designs of the demonstrator, and each of its subsystems. ~~The~~ benefits resulting from ~~the~~ use of an SIPU are presented.

INTRODUCTION

Aircraft secondary power systems, e.g., normal electrical and hydraulic power sources, environmental control systems, engine starters, emergency power units, etc., characteristically consist of individual devices for performing each function. Not all of these devices are aircraft mounted, but rather some are parts of cumbersome ground support equipment. Future military aircraft will demand more and more self sufficiency, requiring more compact, lighter weight, totally aircraft-mounted secondary power equipment that may best be realized by using an integrated multifunctional device. Furthermore, increasing demands for secondary power system operation at high altitudes will require performances beyond those of the current strictly air-breathing combustion types. These needs have led to the concept of the Super Integrated Power Unit (SIPU).

An SIPU is a single aircraft power unit capable of providing (1) electrical, pneumatic, and hydraulic power for ground maintenance and standby operations, (2) main engine ground start power, (3) emergency in-flight electrical and hydraulic power, and (4) in-flight emergency main engine restart power. Power can be obtained from jet fuel combustion with ambient air for the ground operations, engine ground starts, and low altitude emergency engine restarts; from a gas generator system using onboard stored propellants for the emergency power and high altitude emergency engine restart functions; and from the aircraft main engine as a second source for the emergency electrical and hydraulic power function.

An SIPU concept demonstration program is being conducted at Rockwell International/Rocketdyne Division, sponsored by the Aeropropulsion Laboratory at Wright-Patterson Air Force Base. The program is for design, fabrication, assembly, and test of a liquid oxygen/jet fuel-based demonstrator SIPU capable of accomplishing all the secondary power functions listed above except the pneumatic ground power function.

Studies and experiments at Rocketdyne show that the SIPU concept is feasible. The aircraft can be made self-sufficient, i.e., independent of most ground support equipment, and can be provided with more emergency power options than are now available, all with small weight penalties. Furthermore, since the SIPU power turbine is driven by an oxygen/jet fuel gas generator, it uses available propellants and can operate in the cold environments and high altitudes encountered by the aircraft--features not presently possible with existing secondary power systems.

Several SIPU concepts were studied, two concepts being preferred for reasons of light weight and technological feasibility. One is a single unit SIPU with a single power turbine in the Auxiliary Power Unit (APU). The APU has a clutched shaft that must be engageable at full speed. The other SIPU has a separate Emergency Power Unit (EPU). The separate EPU has the advantage of an independent power turbine for the emergency power generator and hydraulic pump. The APU can then employ a simple clutched shaft that does not engage under load.

Incorporation of an SIPU will improve the operational capabilities of an aircraft by minimizing logistics support requirements, increasing availability, improving servability, and reducing maintenance.

DISCUSSION

The SIPU concept is applicable to both single-engine and multi-engine aircraft. It is presented here as a single-engine aircraft device.

This discussion presents what is considered to be the optimum SIPU concept. That is, all the secondary power needs of an aircraft are available. A specific aircraft design may include all or any of the available capabilities and be tailored to meet the mission duties of the aircraft. The capabilities of the SIPU secondary power functions are:

1. Generation of electrical and hydraulic power plus environmental control air power for aircraft ground checkout or "start standby" operations. This eliminates the need for aircraft main engine operation or ground support equipment (GSE) for these operations.

2. Aircraft main engine ground start. Torque is applied to the main engine to accelerate the engine rotor to i light-off speed and further to an engine self-sustaining speed.

3. Generation within 3 seconds of in-flight emergency electrical and hydraulic power upon loss of normal electrical and/or hydraulic power.

4. Main engine in-flight restart assist upon loss of main engine operation at any altitude. Torque is applied to the main engine rotor to attain and/or maintain speeds required for reignition and self-sustained operation.

To perform there functions, the SIPU operates in three different modes:

1. Air-breathing mode. Main engine jet fuel from the aircraft fuel tank(s) is burned with ambient air to drive the SIPU turbine.

2. Gas generator mode. On-board stored propellants (fuel and oxidizer) react in a chamber; the exhaust gases drive the turbine.

3. Main engine bleed air mode. Air from one of the main engine compressor stages drives the turbine.

These modes are used for the various functions in Table 1. Generation of ground power plus environmental control system (ECS) air in the air-breathing mode provides for long duration operation with minimal use of stored aircraft fuel. Main engine ground start is also accomplished in the air-breathing mode either as a transition from the ground power function or as a transition from an aircraft quiescent state. In-flight emergency electrical and hydraulic power generation can be accomplished in three modes as indicated in Table 1. The gas generator mode is independent of both altitude and the operational capability of the main engine, but is limited in duration by the quantity of stored propellants. Below certain altitudes the air-breathing mode can be used to generate at least part of the required emergency secondary power. The air-breathing mode is also independent of the operational capability of the main engine and is capable of long duration operation. It is, however, altitude power-limited. With the main engine operating, emergency power can also be provided in the SIPU bleed air mode. This mode, however, is also altitude power-limited.

Table 1. SIPU Functions - Operational Modes Relationship

Function	Operational Mode
Ground Power + Environmental Control System Air	Air Breathing
Main Engine Ground Start	Air-Breathing
In-Flight Emergency Secondary Power	Gas Generator Air Breathing Main Engine Bleed Air
Main Engine In-Flight Restart Assist	Gas Generator Air-Breathing

Finally, in-flight main engine restart assist is accomplished primarily in the gas generator mode (at high altitudes), but can be supplemented by the air-breathing mode at certain (lower) altitudes. The gas generator restart power is independent of altitude, but is limited in the number of restart attempts by the quantity of stored propellants. The air-breathing mode does not have the number-of-starts limitation, but is limited in power level by altitude effects.

A schematic of the SIPU system concept is shown in Fig. 1. The major components of this system are: (1) the auxiliary power unit (APU), (2) the gas generator (GG) subsystem, (3) the accessory drive gearbox (ADG), (4) the electrical generators and hydraulic pumps, and (5) the control subsystem (not indicated). Although the SIPU proper consists of just the APU and the GG subsystem, the total system design concept is presented here for completeness of understanding.

The APU is a gas turbine engine that delivers shaft power to the accessory drive gearbox in the air-breathing, gas generator, and bleed air modes. Three separate sets of internal nozzles are used to deliver to the turbine the working fluids generated by the APU burner, the externally mounted gas generator system, or the main engine compressor (bleed air). The APU compressor also provides air to the aircraft environmental control system (ECS).

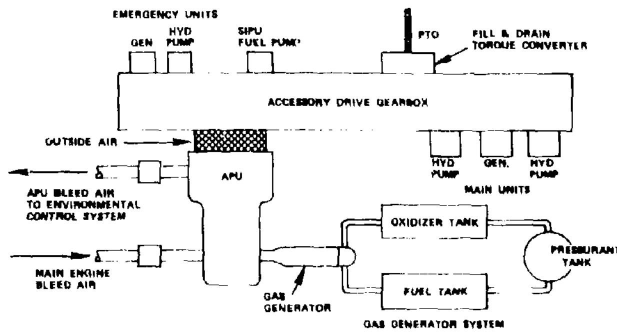

Figure 1. SIPU System Concept Schematic

The gas generator subsystem consists of an APU mounted gas generator (combustion chamber, propellant injector, propellant control valve(s) and ignition system), and an aircraft-mounted propellant feed system. Two types of propellant feed systems are available: (1) separate dedicated aircraft mounted oxidizer and fuel tanks (one tank for a monopropellant system and two tanks for a bipropellant system) and (2) dedicated tank for the oxidizer only--the fuel being supplied by a pump from the aircraft main fuel tank. The bipropellant arrangement for the first type subsystem is illustrated in Fig. 1 and is called the "pressure-fed fuel subsystem"; the second type system is called the "pump-fed fuel subsystem." The propellants may be either gaseous or liquid. Dedicated liquid propellant tanks require pressurant sources, as illustrated.

The accessory drive gearbox is the power transmission device of the SIPU system. During performance of ground functions, auxiliary power unit input shaft power is transmitted to the main hydraulic pumps and electrical generator. During emergency power generation, APU input shaft power is delivered to the emergency hydraulic pump and electrical generator. For ground and emergency main engine starts, APU start power is transmitted to the power takeoff (PTO) shaft. Finally, APU fuel for air-breathing mode operation is supplied by the ADG mounted fuel pump. This pump also supplies aircraft fuel for a "fuel pump-fed" gas generator system. The fill and drain torque converter shown is one of a number of methods for: (1) separating the ADG from the aircraft main engine during performance of ground functions (torque converter drained) and (2) connecting the ADG to the main engine during APU assisted ground and emergency starts (torque converter filled).

The ADG design segregates the main hydraulic pumps and generator from those used for emergency purposes to prevent a single point gearbox or main generator or pump failure from causing a total loss of electrical hydraulic power to the aircraft.

The SIPU electrical generators are of two types. The emergency generator is used to meet only the smaller emergency electrical needs (~5 kVA) whereas the larger main generator must supply electrical power for ground checkout or standby functions (~50 kVA).

The SIPU hydraulic pumps are also of two types. The emergency hydraulic pump is used for the smaller emergency power (~50 hp). The main hydraulic pump power is higher for ground power needs (two 90 hp pumps).

The SIPU system control subsystem is multifunctional. Not only does it execute the discrete events necessary to initiate and terminate the air-breathing, gas generator, and main engine bleed air modes of operation, it controls the performance of these modes over the required wide power ranges. Such control is based on maintaining a constant APU speed. For the air-breathing mode, constant speed is maintained by modulation of the APU fuel control valve; for the gas generator mode, constant speed results from control of the gas generator propellants valves; and main engine bleed air mode speed control is achieved through modulation of the throttle valve at the APU bleed air inlet. Gas Generator propellant valve control may mean flow modulation of open/close pulse operation. By simultaneous control of the two valves of a bipropellant gas generator, the ratio of oxidizer to fuel is maintained within allowable limits. Further, due to a hierarchy of operational modes for executing the emergency power function, the air mode is selected as the primary mode and is augmented with the gas generator mode only to the extent necessary to maintain constant APU speed. Another modal transition performed is from gas generator to air-breathing mode, when appropriate, during execution of the emergency power function. Finally, functional transitions are performed such as ground check-to-main engine start, as well as emergency power-to-engine in flight restart. These multifunctional demands, plus the desirable maintenance feature of built-in-test capability, dictate that the SIPU system control subsystem be digital and microprocessor based.

All functions of the SIPU are reached by first operating the gas generator. Once this has occurred any other function can be attained. Figure 2 shows this schematically. The gas generator is used either to start the APU (air-breathing start) from which ground power and ground engine start may be achieved or to enter an emergency mode of operation. The emergency mode provides power either from gas generator operation or, if available, from main engine bleed air. In-flight engine restart is also achieved through the emergency mode.

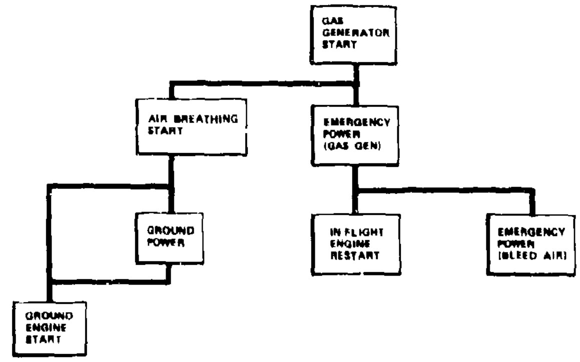

Figure 2. Demonstrator SIPU Operational Requirements

Achieving all functions through the gas generator provides several advantages. Sufficient power is available at the full range of operational temperatures to start the APU. The temperature sensitive hydraulic start of the APU is thus eliminated. Emergency power is available immediately when needed and transfers to the slower starting bleed operation only when full power is achieved and sufficient bleed air is available. A key feature of the SIPU is that emergency in-flight engine starting can be attempted while still providing emergency hydraulic and electrical power.

The advantages of the SIPU include: (1) ground support equipment independence, (2) altitude independent emergency and main engine in-flight restart power, (3) temperature insensitive APU start capability, and (4) a minimum number of secondary power system components resulting in low weight, minimum envelope package.

These advantages result in an aircraft with: (1) a high degree of disbursability due to the reduced need for support equipment, (2) an increased survivability from the capability to restart the engine(s) at any altitude or attitude and from the emergency function, (3) a greater availability because of insensitivity to starting temperatures and higher reliabilities, and (5) easier maintainability and higher reliability due to the fewer components needed to perform the SIPU functions.

Some of the current air-breathing type auxiliary power units and jet engine starters are started by hydraulic motor systems. Within the -40 to -65 F ambient temperature range, hydraulic start motor systems are very unreliable, thereby limiting APU or engine starter usefulness. This problem does not exist with the SIPU because the relatively temperature insensitive gas generator system is used to initiate the air-breathing mode. This is done by accelerating the APU rotor to full speed with the gas generator, terminating gas generator operation and igniting the APU during spooldown.

The multifunctional component design approach of the single-unit SIPU minimized the number of components required in the advanced secondary power system. The result is higher reliability than for multiunit systems. In addition, there is little weight penalty, if any, for the many advantages of the SIPU. Table 2 provides a weight comparison for a pump-fed SIPU compared to an F-16 type aircraft containing a jet fuel starter (JFS) and a hydrazine emergency power unit (EPU). As shown, the change in weight ranges from 0 to 65 pounds, a small amount compared to the advantages gained. The advantages of an SIPU translate into higher sortie rates by minimizing logistic support ground equipment needs and reduced maintenance. These features also allow aircraft disbursement because of its increased self sufficiency.

Table 2. SIPU Weight Analysis Results - Gas Generator Pump-Fed Fuel Configurations

	System Configuration	Weight, pounds	Δ Weight With Current System, pounds
Single Unit	Solid Shaft APU With Compressor Flow Shutoff	617	65
	Solid Shaft APU With Compressor Variable Diffuser Vanes	617	65
	Clutched Shaft APU	552	0
	Free Power Turbine, Two Gas Generators	571	19
	Free Power Turbine, Two Hot Valves	569	17
	Free Power Turbine With Gas Producer Starter Clutch	571	19
Two Unit	Solid Shaft APU, Separate EPU	592	40
	Solid Shaft APU, Separate EPU With Integrated Tanks	581	31
	Clutched Shaft APU, Separate EPU With Integrated Tanks	580	34
	Current JFS + Hydrazine EPU System	552	0

The present status of the SIPU program is that demonstration tests are underway. The program is for design, fabrication, assembly, and test of a demonstrator capable of accomplishing the following functions while operating in the modes indicates:

Function	Operational Mode
Ground Power	Air-Breathing
Engine Ground Start	Air-Breathing
Emergency Power	Gas Generator and Main Engine Bleed Air
Engine In-Flight Restart	Gas Generator

The nature of the current program is such that the low cost available hardware be used whenever possible to demonstrate proof-of-concept. As a result, the demonstrator in total is not flight weight.

The current program is expected to be followed by a flight demonstrator program tentatively scheduled to begin in early 1984. This program will require approximately 3 years to complete including an in-flight demonstration test series. On this schedule, it is anticipated that production versions of an SIPU can be available for aircraft starting in 1987.

CONCLUSIONS

1. The SIPU concept is feasible for both single- and multiple-engine military aircraft.
2. The SIPU can provide significantly increased aircraft capability in temperature and altitude operations with little or no weight penalty.
3. The SIPU can provide significant independence from ground support equipment.
4. The SIPU allows an increase in sortie rate.
5. The SIPU allows wide disbursement of aircraft because of the increase self sufficiency.

DISCUSSION

Ph. Ramette, Fr

Dans le cas d'un groupe de puissance superintégré, le nombre de redémarrages est limité par la quantité de combustibles stockés. Quel a été le nombre de redémarrages possibles lors des essais que vous avez effectués?

Author's Reply

The number of main engine restarts required to be demonstrated was three. This is in addition to providing emergency power (electrical and hydraulic for 10 minutes). In an actual aircraft application the amount of propellant can be varied to satisfy the requirements of the user. The amount of propellant needed is dominated by the emergency power requirements. Engine starts require much less propellant and therefore do not affect weight or volume very much. APU starts require very little propellant. This function has essentially no effect on weight or volume.

'ALL-ELECTRIC' ACCESSORY DRIVE SYSTEMS: IMPLICATIONS ON ENGINE DESIGN AND PERFORMANCE

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SUMMARY

Engine studies had been conducted on 'all-electric' accessory power systems in response to questions from aircraft manufacturers, particularly in the civil field. It was felt appropriate to extend the studies to military applications, and to clarify the effects on engine design and performance.

In the 'all-electric' proposal, all accessory power requirements are generated and distributed electrically. A major feature is that air bleeds for environmental control systems are replaced by increased shaft power offtake.

It is concluded that the concept is worthy of consideration for large subsonic transport aircraft, since it offers the prospect of simplification and weight reduction in the engines. For supersonic combat aircraft, the concept is not recommended; with relatively higher levels of shaft power offtake, additional handling problems will be created.

In both cases, it is the effects on the engine and aircraft weight and systems as a whole which need to be considered.

1.0 INTRODUCTION

In the 'all electric' systems concept, all aircraft accessory power offtake from the engines is by mechanically driven electrical generator. Within the aircraft, electric motors drive air compressors for the environmental control system. Aircraft control surface actuators are electrically powered, either directly or through an electrically driven hydraulic pump.

Interest in this concept has been stimulated by the improved power/weight ratio available from electrical machines using samarium/cobalt magnetic material. Interest has been particularly noticeable in the civil field (eg Ref 1), where there is a continuing search to reduce operating costs.

The present paper examines some of the implications, particularly those affecting the engines. Application of the principle to a subsonic transport aircraft is examined and its relevance to a supersonic combat aircraft also considered.

2.0 SUBSONIC TRANSPORT AIRCRAFT

2.1 Power offtake requirements

Table 1 illustrates the order of power offtake needed at 35 000 ft cruise. The figures are for each engine in a twin-engined 150 seater application;

TABLE 1: POWER OFFTAKE (SUBSONIC TRANSPORT)

	CONVENTIONAL SYSTEM	ALL-ELECTRIC SYSTEM
Environmental control system	0.37 kg/s	99 kW
Flying controls	11 kW	11 kW
Electrical services	69 kW	69 kW
Engine fuel pumps	37 kW	50 kW

At 35 000 ft, there is no anti-icing requirement. At altitudes where anti-icing is required (eg 22 000 ft), heater mats to protect airframe and engine cowl would require 63 kW. The total power listed above would not rise by the full amount, since the environmental control system would require less power than at 35 000 ft. So far as engine performance effects are concerned, the 35 000 ft cruise condition is the more significant.

2.2 Environmental control systems

The 'all electric' aircraft environmental control system will use an electrically driven air compressor. A potential attraction of this concept is its flexibility. If the auxiliary compressor is driven only as fast as is necessary to match the environmental control system pressure requirement, the power required to drive it is minimised over the full range of engine/aircraft operating conditions. At low altitude (high pressure) conditions, the device would operate at low pressure ratio, low $W\sqrt{T}/Pd$. At high altitude, higher pressure ratio and higher $W\sqrt{T}/Pd$ are required. Thus the speed of the compressor could be programmed to increase with altitude (or similar parameter), and so match the aircraft requirements. The pressure required in the aircraft cabin is typically that shown by the lower line on Figure 1 (line A).

If this cabin pressure is to be provided by a conventional (ie engine air bleed) environmental control system, the engine bleed pressure will need to be at least that shown by line B in Figure 1. The difference between lines A and B is characteristic of a modern environmental control system in setting up the required cabin air pressure and temperature. The pressure available from various compressor stages at selected operating conditions is also shown on Figure 1. Current aircraft may tap air from a particular compressor stage, switching to compressor delivery for the descent phase. At low altitude (including take-off), this is wasteful of pressure. Hence the electrically driven system is able to show a modest fuel saving (see Appendix). This saving is limited, partly because the overall efficiency of the electrical power transmission system will never exceed 50% (see Figure 2).

The direct air bleed system can be improved by providing a further bleed point on the compressor, so that the bleed can be taken from the most economical point. By this means, the fuel penalty can be made comparable to that of the 'all-electric' offtake system. However, this multi-bleed system requires additional ducts and control valves.

The data in Figure 2 also shows that the electrical transmission will reject 39 kW of heat at cruise, plus a further 13 kW if the engine fuel pumps are electrically driven. The heat sink capacity of the fuel is already likely to be fully committed, even allowing for reduced heat dissipation from the simplified engine gearbox. However, the cooling load of the aircraft environmental control system will also be changed. An environmental control system trade-off study is needed to establish the revised cooling needs, but that is outside the scope of this paper. The particular problems of the 'one engine out' and other failure cases need to be included.

2.3 Starting

It can be shown that a 150 KVA generator operating in the starting mode and operated by a 400 Hz 115 volt 3 phase power supply regulated by the variable speed constant frequency system is capable of starting most large civil engines in reasonable time. This implies that a suitable APU is available to provide the electrical power.

2.4 Engine design

The 'all-electric' concept offers considerable scope for engine simplification and weight reduction. The following points are relevant:

a) Air bleed system: The manifolds needed to bleed off substantial air flow for the aircraft environmental control system would no longer be needed. An efficient conventional subsonic transport aircraft would require the facility to bleed probably three stages of the HP compressor, together with appropriate ducting and control valves. The only air bleeds remaining would be those required by the engine itself, including starting and handling capability.

b) Engine gearbox design. This could be greatly simplified if it was only required to drive one accessory, ie an electrical generator. (The ultimate step would be to integrate the generator into the engine core and hence dispense with the gearbox and cross drive altogether, but consideration of this is not included in the current paper). A modern 150 KVA, 21 000 rpm generator could be 280 mm diameter x 450 mm long. It will be accompanied by a variable speed constant frequency controller of at least equal volume.

c) Engine installation. The new concept would reduce the number of pipe connections affected when removing an engine. In particular, major air and hydraulic pipes between engine and aircraft would be avoided, although the electrical connectors will have to carry additional load.

d) Fuel pumps. Drives for the engine fuel pump would be redesigned. Electrically driven fuel pumps are widely used in aircraft systems, so a precedent exists for considering that option. Integrity and failure analysis aspects require further investigation.

e) The direct effect on engine weight will be a saving of approximately 40 kg on a 100 kN thrust engine. This does not include changes in the weight of the accessories themselves, or changes to system components in the aircraft.

3.0 SUPERSONIC COMBAT AIRCRAFT

3.1 Power offtake requirements

This section relates to a medium size turbofan engine, with afterburning, applied to a twin-engined aircraft.

Table 2 illustrates the order of power offtake needed with a conventional accessory power system. The two sets of figures given are for: a) a normal cruise condition and b) short duration peak demands. The latter relates to single-engined operation including reheat: the listed demands may not occur concurrently.

TABLE 2: POWER OFFTAKE FOR CONVENTIONAL SYSTEM

	CRUISE	PEAK DEMAND (SINGLE ENGINE)
Environmental control system	0.20 kg/s	0.40 kg/s
Flying controls	20 kW	120 kW
Electrical services	40 kW	80 kW
Engine fuel pumps	20 kW	180 kW
Engine nozzle actuation	0	10 kW

Compared to the subsonic transport engine considered in section 2.0, the following points are of interest:

a) The peak power for fuel pumping is particularly high. This is due to the after-burner fuel pump.

b) The total power offtake of the military engine in cruise is greater relative to the spool power: the effect is even more significant when the afterburner is in use.

c) The single engine failure case (ie all services supplied by one engine) is more significant because of the relatively high accessory power demands.

Table 3 illustrates the order of power offtake needed with an all-electric drive system.

TABLE 3: POWER OFFTAKE FOR 'ALL-ELECTRIC' DRIVE SYSTEM

	CRUISE (BOTH ENGINES)	CRUISE (SINGLE ENGINE)	PEAK (SINGLE ENGINE)
Environmental control system	25 kW	50 kW	50 kW
Flying controls	25 kW	50 kW	160 kW
Electrical services	40 kW	80 kW	80 kW
Engine fuel pumps	30 kW	30 kW	220 kW
Engine nozzle actuation	0	0	10 kW

One immediate conclusion is that interposing an electric generator/motor link in the drive to the afterburner fuel pump is unacceptable: the power wasted is too great.

3.2 Environmental control system

In the supersonic combat aircraft, the air bleed for environmental control system purposes is a lower percentage of the total accessory power load. Thus any saving in fuel consumption by introducing an 'all-electric' drive system is minor in comparison with other considerations.

3.3 Engine handling

As described in the Appendix, the effect of taking shaft power is to move the HP compressor working line towards surge, while taking bleed moves the running line away from surge. Thus the conventional environmental control system air bleed system has the advantage that it helps to offset working line movement caused by shaft power demands. If the 'all-electric' drive system is adopted, working line movement will be increased. While it is possible to match the design of an engine to cater for a fixed quantity of shaft power offtake, this is difficult in the case of the supersonic combat application because the range of power offtake is so great (ie 5:1). Hence if the engine is designed to provide the substantial surge margin needed for rapid engine acceleration, at other conditions the compressor will be operated at a point where significant loss in efficiency occurs.

An alternative approach is to measure the shaft power offtake, and allow for this in the control of the engine. Provided the necessary degree of variable geometry exists (eg variable inlet guide vanes), modern electronic control systems are ideally suited to performing this task.

3.4 Engine design

Although some saving in engine weight could be expected if electrically driven accessories were adopted, the potential saving is substantially less in the supersonic combat engine than in the case of the subsonic transport engine. The reasons are as follows:

a) Air bleed system. On the supersonic combat, there is not such a strong case for complex air bleed systems. There is therefore less to be gained by deleting the relatively simple air bleed systems used on current supersonic military engines. In any case, existing starting and handling bleeds would have to be retained.

b) Engine gearbox design. Since it is recommended that the afterburner fuel pump should continue to be engine driven, there is less scope for simplifying the engine gearbox. Also in some supersonic combat aircraft, certain aircraft accessories are mounted on a separate gearbox, ie not on the engine gearbox itself. Consequently there is less to be gained from the new concept.

4.0 DISCUSSION

In considering the application of 'all-electric' accessory power systems, the most significant change concerns the provision of air for the aircraft environmental control system.

For comparison, it is convenient to use a conventional air bleed system as datum, ie bleed is from one compressor stage, probably switching to HP compressor delivery for idling descent. This bleed has an effect on engine fuel consumption. By choosing a more efficient environmental control system and providing a further air bleed point on the compressor, the fuel consumption penalty can be improved. A similar fuel saving can be obtained by choosing the electrically driven environmental control system, and the latter has the advantages that a) complex air bleed manifolds and control valves are not required and b) engine gearbox design can be simplified. Thus for the large subsonic transport application, the electrically driven approach has merit provided the weight saved on the engine is not negated by additional weight in the associated aircraft systems. The fairly modest items of weight saving have a cumulative effect when considering aircraft fuel consumption and payload.

On an engine for a supersonic combat aircraft, additional considerations arise. Accessory power requirements are higher relative to the size of the engine core. Consequently, there are additional handling problems, and replacing air bleed by additional shaft offtake would worsen the difficulties. The scope for simplifying engine design is also more restricted, particularly since it is concluded that the afterburner fuel pump should not be powered by electrical transmission. Thus in the supersonic combat application, engine considerations indicate that the 'all-electric' approach should not be adopted. It may be appropriate to ask whether the fairly complex mechanical/pneumatic transmissions often used in this type of aircraft could be replaced by electrical power transmission. However, consideration of the special needs of the aircraft, and accompanying integrity and operation features go beyond the scope of the present paper.

APPENDIX

ENGINE PERFORMANCE

General

Figure 3 shows a HP compressor characteristic, and the steady 'working line'. This line is produced by metering the appropriate fuel flow. To accelerate the engine, fuel flow is increased. During the subsequent transient, the compressor delivery pressure rises above the steady state working line, ie the HP compressor runs closer to surge.

Taking shaft driven accessory power also raises the running line as indicated on Figure 3, ie it reduces the surge margin. Taking air bleed from the HP compressor has the opposite effect. Movement of the working line can also affect the efficiency at which the compressor operates.

A fair comparison between the engine performance with a conventional environmental control system and the all electric system requires the engines to be matched so that at the chosen level of bleed or power offtake, the installed working line is identical.

Large subsonic transport

The engine performance studies are based on an advanced turbofan with a nominal bypass ratio of 7, an overall pressure ratio of 32:1 and a hot day take-off stator outlet temperature of 1600 K. The engine is sized to meet the thrust requirements of a 150 seat twin-engine aircraft.

Table 4 illustrates the effect of 0.37 kg/sec bleed on engine specific fuel consumption. This should be examined in conjunction with section 2.2 and Figure 1.

TABLE 4: EFFECT OF AIR BLEED

TAPPING POINT	SFC (5000 ft climb)	SFC (36 000 ft cruise)
Cabin pressure	Datum	Datum
3rd stage	+0.43%	+0.32%
4th stage	+0.55%	+0.54%
5th stage	+0.67%	+0.79%

Comparisons between the block fuel used on a typical 500 nautical mile mission are given in Table 5 for different standards of environmental control system.

TABLE 5: BLOCK FUEL USAGE

DESCRIPTION	LOCATION	BLOCK FUEL
Conventional environmental control system	5th and 10th stage tappings	Datum
Improved environmental control system	3rd, 5th and 10th stage tappings	-0.47%
All electric	Shaft from HP spool	-0.44%

A constant fan diameter was assumed, with core size allowed to float to maintain a constant hot day take-off HP turbine rotor blade metal temperature. The aircraft thrust requirements were assumed to be the same in each case. The table shows that the modest fuel saving is similar for both the improved environmental control system and the 'all-electric' system. The fuel saving is equivalent to about 28 kg per aircraft per mission.

The descent phase was not included in the above tabulation, since it represents only a small proportion of the total fuel consumed. There could be slight advantage to the 'all-electric' system during that phase, because with an air bleed system, engine rpm during descent is being held unnecessarily high specifically to meet environmental control system needs.

Although a relatively small hot core was used for the base engine, its precise detail is not important to the result since the effect of power offtake or bleed is a function primarily of core power, not core engine flow.

There will also be a cumulative effect arising from adjustments in aircraft weight due to changes in fuel load, engine weight, and accessory systems weight. The tabulations do not include these features.

Supersonic combat aircraft

The principles are similar to those described above. However, the implications are different for reasons described in section 3.3.

ENVIRONMENTAL CONTROL SYSTEM AND ENGINE AIR PRESSURES

(SUBSONIC TRANSPORT)

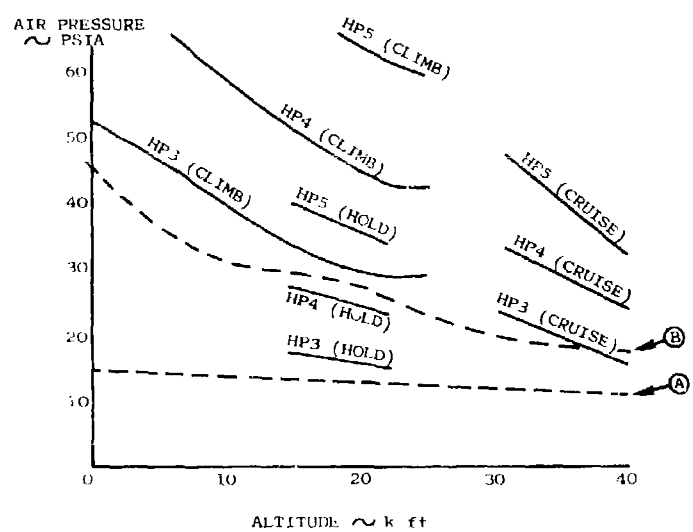

FIGURE 1

POWER CONSUMPTION FOR ELECTRICAL ENVIRONMENTAL CONTROL SYSTEMS

(SUBSONIC TRANSPORT)

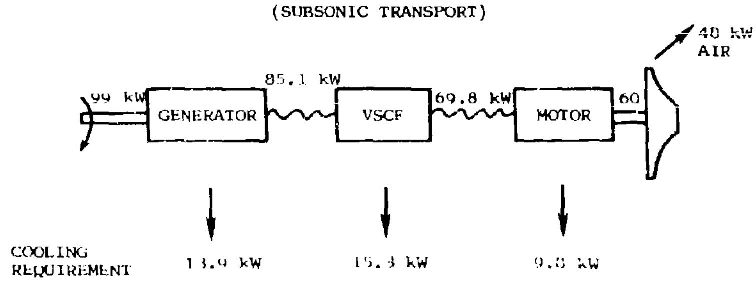

FIGURE 2

COMPRESSOR WORKING LINES

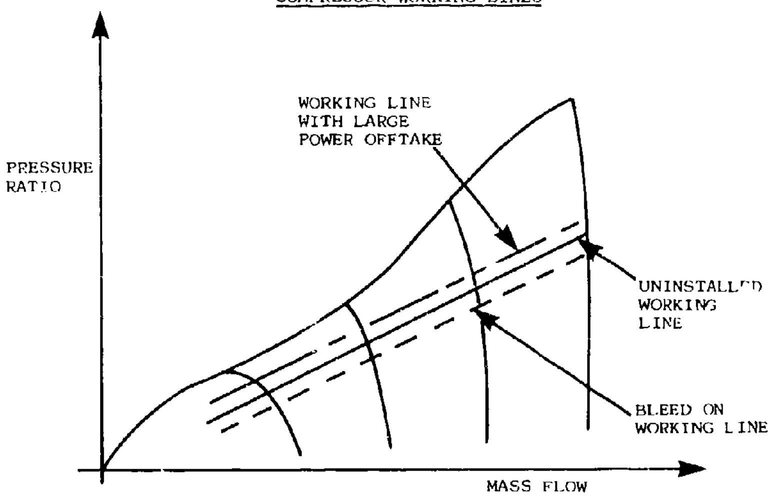

FIGURE 3

REFERENCE

Ref 1 The All-Electric Airplane as an Energy Efficient Transport
M J Cronin (Lockheed)
SAE Paper 801131, October 1980

DISCUSSION

E.W.Eckert, Ge

The conversion of secondary power system to an all electric system may have advantages in terms of better control, high overload possibility etc. The distribution of high loads means high current and affects the cabling system, as either the wires cross sections have to be increased (weight implication!) or a higher voltage has to be chosen. 270 V DC have been mentioned in this context. Are the data given in fig.2 based on such a higher voltage?

Remark: Introduction of higher voltage will have considerable implications on logistics and standardization.

Author's Reply

The figures and efficiencies quoted in the paper were based on present-day voltages. You imply that the weight of copper could become serious: this could be so. Against this, the "all-electric" system will alow us to throw away some major connections between engine and aircraft including oil and large air pipes. This again shows the need for the aircraft, engine and accessory specialists to work closely together to fully assess the difficulties at an early stage.

Ph. Ramette, Fr

Entre un système conventionel et un système "tout électrique" vous avez présenté des comparaisons sur les prélèvements de puissance nécessaires et sur les poids. Avez vous effectué des comparaisons concernant la maintenance du système.

Author's Reply

We have not yet studied maintainability issues. Solely from the engine viewpoint, there should be some improvement since there are no longer, for example, hydraulic connections to the engine.

A.L.Romanin, US

What are the types of main engine generators, any high speed P.M.G.?

Author's Reply

The efficiency figures used were those associated with Samarium cobalt V.S.C.F. machines.

Implementing Microprocessor Technology in Aircraft Electrical Power Generating System Control

by

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ABSTRACT

Today's advanced aircraft electrical power generating systems rely on microprocessor technology for the implementation of most control, protection, and built-in test functions. Microprocessors offer distinct advantages over discrete logic devices in system design and performance. The first half of this paper highlights these advantages by illustrating the design implementation process used in current systems.

The second half of the paper expands on these advantages. By adapting more advanced microprocessor systems in the next generation of aircraft electric systems, additional functions can be implemented. Microprocessor control of generator paralleling and voltage regulation coupled with more effective built-in test capabilities will result in significant improvements in system performance.

INTRODUCTION

Microprocessors are operating in a variety of advanced commercial and military aircraft electric power generating systems. Each application is unique in its power rating, bus structure, control and protection logic, and built-in test capabilities. Yet, they all maintain a substantial amount of commonality. This is due largely to the use of microprocessors and a building block design approach to create built in system design efficiency and performance advantages.

The building block analogy starts with primary blocks representing those components making up the electrical power generating system. Blocks corresponding to system control units will in turn be composed of many smaller blocks representing individual system functions. Some of the smaller blocks will be circuit blocks. Others will be software blocks. The selection and combination of these smaller detailed blocks produce the characteristics of the control units and ultimately the system operation.

Examining the design process provides an insight into the advantages of microprocessors in today's advanced systems. In the future, new building blocks will further inhance system performance. These new blocks will become available as microprocessors are implemented in more system functions.

I. ELECTRICAL POWER GENERATING SYSTEM (EPGS)

This discussion is based on systems powered by 115/200 Vac, 3 phase, 400 Hz generators. Primary functions of the EPGS are to generate electric power and distribute it to the electrical loads on the aircraft.

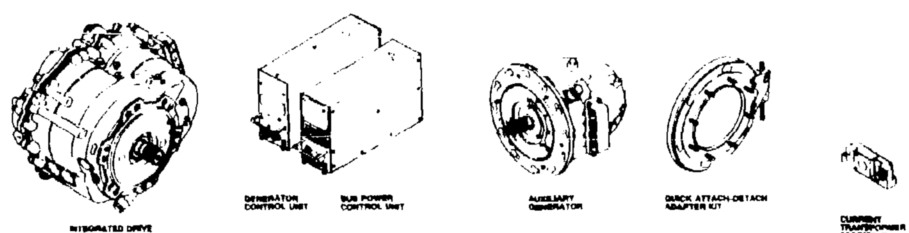

Figure 1 Electrical Power Generating System

A. Electrical Power Generating System Devices

Major components (primary blocks) of the typical EPGS include the following devices shown in Figure 1:

-Aircraft engine driven integrated drive generators (IDG)
-Auxiliary generator driven by the auxiliary power unit (APU)
-Current transformer assemblies (CTA)
-Generator control units associated with each generator (GCU)
-Bus (or ground) power control unit (BPCU or GPCU)

The number of CTAs is dictated by the number of system power sources and the extent of the overload and feeder fault protection provided.

B. System Configuration

Figure 2 is a line diagram of the EPGS. This example system is nonparallel but takeover of one main source's loads by the other is allowed. The external power and APU sources can assume one or both main loads depending on availability of the main sources. Specific system responsibilities are as follows:

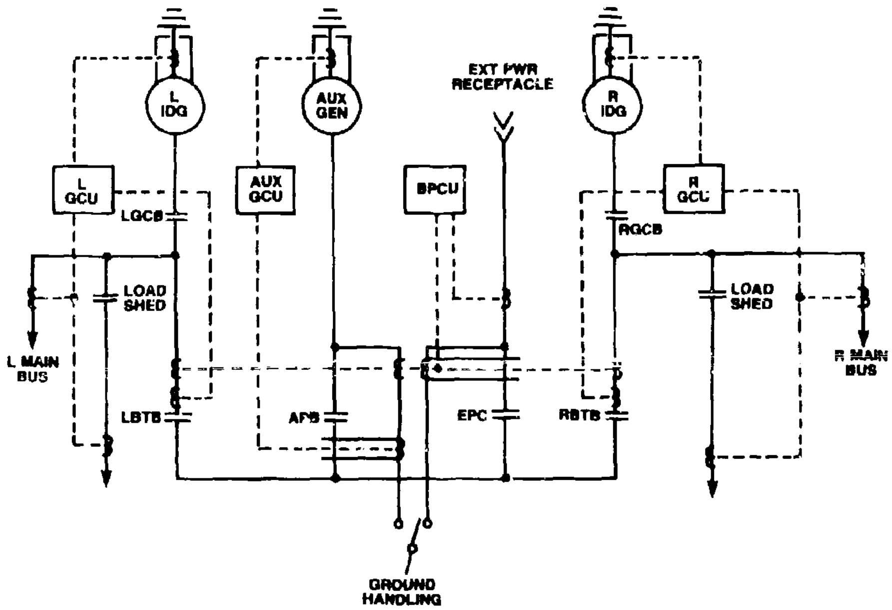

Figure 2 Typical EPGS Bus Structure

GCU - The GCU's primary function is to monitor and regulate the generator. Its secondary responsibility is to control and protect the main bus. This is accomplished by manipulating breakers in response to fault conditions and system configuration. The GCU also has built-in test equipment (BIT) capabilities for monitoring the integrated drive generator, main and auxiliary bus breakers, and its own internal circuitry.
BPCU - The BPCU controls and protects the external power bus, tie bus, and ground handling bus. Communication and BIT interrogation for system controls are initiated by the BPCU. The display and switches for BIT interrogation are located either on the BPCU front panel or on a remote panel connected to the BPCU by data link.
CTA - The CTAs monitor system current for overload and feeder fault protection.

II. ELECTRICAL POWER GENERATING SYSTEM DESIGN

Design of the microprocessor-based EPGS control units typically includes the following steps in the design cycle (1):

SYSTEM REQUIREMENT ANALYSIS
SPECIFICATION
ARCHITECTURAL DESIGN
DETAILED DESIGN
IMPLEMENTATION
MAINTENANCE

A. System Requirement Analysis

As with any design function, the first step is to analyze the needs of the system. Examples used here are based on systems installed on the Boeing 757/ 767, Airbus A310, Air Force (Boeing) KC-135R, and their derivatives. The primary needs of each of these aircraft are a high quality/reliability power output and a high degree of automation. The high quality/reliability output requirement is dictated by the increased use of electronics for flight control. Increased automation to reduces crew work load and also plays an important role in advanced power transfer and system protection schemes.

Secondary system requirements include weight reduction and decreased maintenance downtime. In light of fuel cost increases, the inverse relationship of fuel efficiency to aircraft weight is a critical design consideration. High costs of lost revenue and labor make it mandatory that nonproductive maintenance downtime be minimized.

B. Specifying System Functions and Constraints

Based on the requirements, a functional description and operating constraints of the system are developed. Functions of the system will be categorized as control, protection, and built in test. Each functon will have its corresponding constraints. For example, routing power to a load channel may be inhibited by a control constraint under single source conditions. A typical protection constraint would be the operating voltage limits for the system. The built in test functions would be constrained by the type of crew interaction proposed.

The end product of this exercise is a specification document describing system performance.

Dwelling on the apparently obvious steps of requirement analysis and specification seems to belabor these points. However, experience has shown that when emphasis is placed on these efforts, two important gains are made. First, the customer and designer will be assured that their interpretations of the system functions will be based on the same requirements. Second, the designer will have a clearer picture of what the intended results are to be. These advantages will become more apparent later in the design process.

C. Architectural System Design

The architectural system design effort consists of identifying the input and output signal requirements and the logic expressions required to perform the specified system functions. Since this system is to be microprocessor-based, this is also the time to select the processor to be used and estimate the memory and peripheral requirements.

Processor selection is based on two criteria. The first will be considered "design" requirements. In this category, the following parameters are studied:

Speed--How fast the microprocessor handles data
Capacity--The amount of information that can be processed
Versatility--The types of peripheral devices available
Instruction Set--Instructions for operating the processor
Functions--Features of the processor that would be useful in the control unit design.

The second requirement is "practicality". This includes:

Cost--Price of the processor and its peripherals
Availability--Lead times and second sources
Support--Logic analyzers, development systems, emulators, and programmers necessary to design a microprocessor-based system
Experience--Personnel familiarity with the processor

Based on the above criteria, the 8085 was chosen as the microprocessor to be used in these systems.

D. Detailed System Design

(1) The Microprocessor Circuit

Once the decision is made to use the 8085, the microprocessor circuit is designed. Stressing commonality, the circuit design is configured for the most complex control unit requirements. Other control unit applications will use a scaled down version of this design.

In the microprocessor circuit, there are provisions for a variety of functional circuits and devices. Figure 3 is a diagram illustrating those circuits and devices.

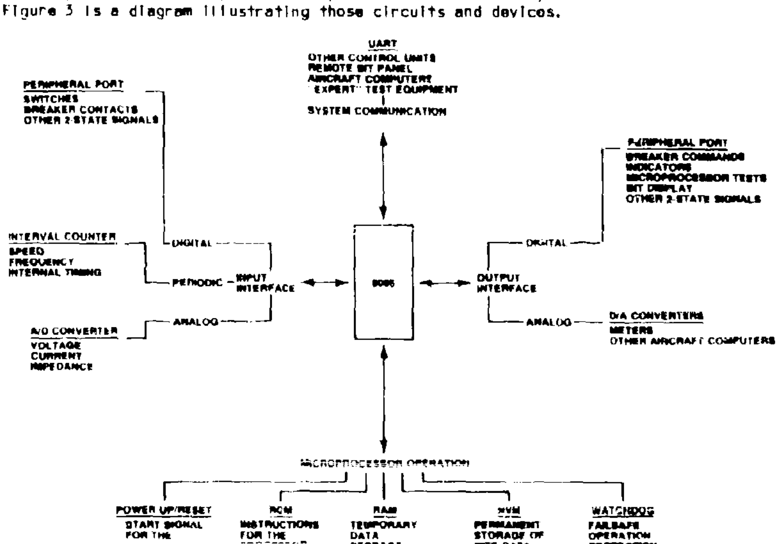

Figure 3 The Microprocessor Circuit

-The POWER UP/RESET circuit is the GO signal that starts the microprocessor reading instructions from the read-only memory.

-READ-ONLY MEMORY (ROM) contains the step-by-step instructions used by the microprocessor to perform the control unit functions. Two technologies of ROM are used; Erasable/Programmable Read-Only Memory (EPROM) and Electrically Erasable/Programmable Read-Only Memory (EEPROM).

-The WATCHDOG circuit initiates a fail safe shutdown of the control unit if a microprocessor related failure occurs that could potentially result in incorrect operation.

-NONVOLATILE MEMORY (NVM) is the storage location for BIT messages identifying system failures. Nonvolatile indicates that the memory is retained when power is removed. Two technologies are also used here. Electrically Alterable Read-Only Memory (EAROM) is similar to and has been replaced by the EEPROM in later designs.

-RANDOM ACCESS MEMORY (RAM) locations temporarily store data used for timing, sensing, and controlling.

-PERIPHERAL PORTS interface the microprocessor with digital inputs and outputs.

-INTERVAL COUNTERS are used to determine frequency of an input and provide periodic timing functions asynchronous to the microprocessor.

-ANALOG-TO-DIGITAL (A/D) and DIGITAL-TO-ANALOG (D/A) CONVERTERS interface the microprocessor with analog inputs and outputs.

-A UNIVERSAL ASYNCHRONOUS RECEIVER/TRANSMITTER (UART) translates the 8 bit parallel information from the microprocessor to serial data that is communicated between control units.

(2) Software Design

The design of software building blocks begins with the abstraction of system functions that outlines the intended purpose of the software. From the outline, a written description of software operation is created. Writing descriptions of the software programs serves two useful purposes. First, it reinforces the goal of bringing outside groups (other than the designers and programmers) into the review process by providing a narrative of the software function. Second, the description provides details required by the programmer. With the written description in hand, the structured design phase begins.

Structured program design (2), as used in EPGS software programming, is primarily concerned with the concepts of modularity, cohesion, and coupling. Beginning with a hierarchy of executive routines or modules, the software is broken down into specific functional modules of ever increasing detail. At the lowest level, the ideal module deals with a single system function. Cohesion is the measure of the degree to which a module's responsibility is restricted to one function. The other concept, coupling, refers to the interdependence of modules. Efficiency in design is improved by maximizing cohesion and minimizing coupling.

In creating the software module, three design aids are used: logic flow charts, pseudocode or program design language (PDL), and data flow charts. The logic flow chart provides a convenient method of representing module operation. It is, however, open to interpretation by the programmer and allows for a choice of programming techniques. The ideal pseudocode restricts programmer interpretation. Also, when used with a system of strict format and syntax rules, pseudocode dictates the instructions to be programmed. This efficiency is offset by the need to study it in detail to understand the function of the module. To obtain the benefits of both methods, utility programs have been created to automatically generate logic flow charts from the pseudocode. The third tool used in software design is the data flow chart. This maps out the travels of data from inputs to memory storage areas to the manipulating modules and ultimately to the interfacing peripheral device. Its main function is to assist in hardware/software integration.

Integration of hardware and software need not take place in the early phases of software design. Simulator programs are available that can be configured to imitate hardware and system functions. Software development systems provide flexibility during development by replacing the read-only memory with external memory such as a disk drive. By using simulators and development systems, the time required for circuit modification and memory reprogramming is greatly reduced.

Beyond the requirements of a particular application, standardization in module design promotes reusability in other systems. This requires that definitions and representations of data and parameters be used consistently. Consideration must also be given to frequently used software functions, such as timing and memory storage, to ensure portability between systems.

(3) Hardware Design

While software development continues, hardware design proceeds with virtual independence.

Interfacing circuits between the inputs and outputs and the microprocessor circuit are developed according to the requirements defined in the architectural system design. Peripheral requirements of the microprocessor circuit are dictated by the following:

1. Digital inputs to the peripheral ports
2. Digital outputs from the peripheral ports
3. Analog inputs to the A/D converters
4. Analog outputs from the D/A converters
5. Periodic inputs to the interval counters
6. Communications channels

These will become the hardware building blocks. It is important to stress that the same reusability philosophy promoted in software design carries over to hardware circuit design. Microprocessors make this goal more attainable. Rather than creating large self contained circuits with inputs, logic, and outputs, as in a discrete design, input and output circuits stand alone. Similar input and output circuits will interface with the same types of microprocessor circuit peripheral devices. Thus, a measure of commonality has been achieved within the control unit and with other control unit applications.

E. Design Implementation

Implementation of the design is the exercise of combining software and hardware blocks to produce the desired control unit functions. It is a series of analyses, tests, modifications, and recombinations.

(1) Software Analysis and Testing

Hardware circuit analysis has the software correlation of module testing and integration testing. Just as the individual circuit is analyzed for response, the software module is exercised to verify that each logical path in the module produces the desired outputs for specific inputs. In integration testing, a software input or combination of inputs is tracked through the software to verify that the resulting system response is correct. These tests and others used to establish software integrity are outlined in military standard MIL-S-52779A, Software Quality Assurance Program Requirements. (3)

The development of software is a labor intensive effort with time equally divided between design and testing. By promoting the reusability of software modules, the testing can be greatly reduced by testing through similarity. (4)

(2) Integration of Hardware and Software

Once integration of hardware and software has taken place, functional testing is performed on the bench and in system mockups to confirm control unit and system operation. Any discrepancies in the hardware or software operation are corrected and the process is repeated. Modularity in the software reduces the impact of repeated analysis and testing by limiting the amount of software affected by any change.

When testing is completed and production of the control unit begins, an important characteristic of a microprocessor system becomes evident. With the exception of the programming of the EPROMs and the assembly of the microprocessor circuit, all costs of software-incorporated functions are nonrecurring. A comparable discrete logic control unit will continue to have the recurring costs associated with assembly. As the complexity of the control unit and the number of units produced increase, the nonrecurring cost impact of software development is reduced.

F. Maintenance

Maintenance of the design, as defined here, pertains to control unit modificaton and configuration control. Experience has shown that even the most thought out systems are prone to change in the life of the program. This is especially true in the aviation industry following delivery of the aircraft to the customer. Product improvements and design changes can have impact on hardware, software, or both. It is advantageous to incorporate any changes in software whenever possible. From a hardware perspective, software changes affect only the EPROMs. This reduces the impact on printed wiring board layout, parts selection, assembly, and testing associated with hardware changes. From the customer's point of view, the effect on spare parts procurement and down time required for modification are minimized. Changes implemented in software are further facilitated by techniques such as the use of easily removable EPROM board subassemblies that permit erasing and reprogramming of the EPROMs without component removal. Configuration control in a software implemented change is accomplished through written descriptions detailing differences between configurations. Impact to illustrated parts lists and schematics are minimal.

III. THE MICROPROCESSOR ADVANTAGE

The decision to implement microprocessor systems in EPGS design is based on two major advantages:

1. Complex Logic and Timing Capabilities
2. Flexibility

These advantages are manifested in features found in existing EPGS systems. The three most significant features are built in test, complex control and protection schemes, and design flexibility.

A. Built-In Test

A major advantage of the microprocessor based controls is the incorporation of built-in test (BIT) functions that significantly impact aircraft electrical system maintenance by reducing checkout and troubleshooting time. These functions are implemented on a level of complexity that would not be feasible with discrete logic. This level of complexity results in fault isolation accuracy calculated at 95%.

The intended purpose of BIT in the EPGS is to detect both active and passive failures in the major system devices and in the internal circuits of the control units. Intermittent faults are also detected. Results of the failure isolation are displayed in codes and messages that reduce the amount of maintenance troubleshooting. Finally, BIT maintains an aloofness with respect to control and protection functions.

BIT's operational mode is designed to isolate active and intermittent faults in the system. It is composed of a collection of routines that are either event-driven or periodic. Those event-driven routines are called in response to a change such as a breaker trip or a command to close a breaker. Other operational BIT routines are called periodically to monitor the status of system functions.

Following a protective action, an operational BIT routine is called to determine the cause of the action. The routine first identifies the specific cause of the action. It then exercises the control unit's corresponding protective circuit to verify proper operation. If the circuit responds, then the protective action is stored in nonvolatile memory. In some circumstances, enough additional information is available to identify another EPGS component as the source of the fault. That information will also be stored. If the circuit did not respond, the protective action is stored in memory with a failure code identifying the circuit fault. Figure 4 is a generalized flow chart depicting a typical isolation procedure.

The other operational BIT functions are the status monitor routines that are called periodically. Each routine monitors an EPGS function such as a control switch or breaker and compares its status to information from other software routines and other control units. Any disagreements are resolved by exercising the circuit in question. If it responds, the system function discrepancy is identified and stored in nonvolatile memory. If it does not respond, the sensing circuit is identified as the failure. Figure 5 is an example of this type of routine.

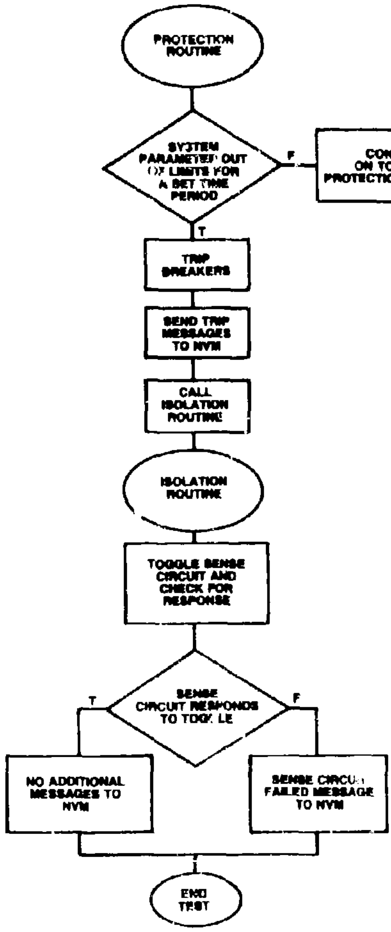

Figure 4 Operational BIT Function Responding to a Protective Trip

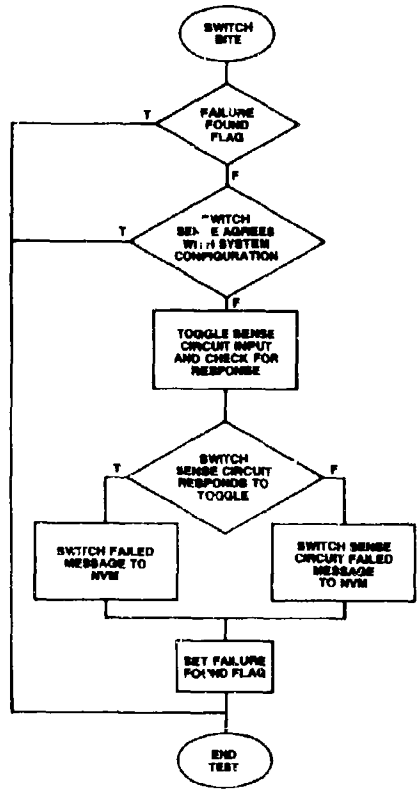

Figure 5 Operational BIT Function Routinely Monitoring Switch Status

Maintenance BIT routines are performed automatically or on request. Both modes test control unit sensing circuits to detect passive faults not otherwise detected because of aircraft operating configuration. In general, maintenance BIT serves as a confidence check and is typically performed following an EPGS maintenance action. It may also be requested as part of a scheduled EPGS check.

The nonvolatile memory is the storage area for all fault messages. Unlike earlier BIT systems that rely on crew response to indicator lamps, nonvolatile memory allows for detection and storage of fault information until it is convenient or desirable for BIT interrogation to be performed. Because the memory is partitioned by flight, an intermittent fault can be identified along with the time of occurence. In existing systems, the BIT information from the current flight and up to seven previous flights can be reviewed during interrogation. Another feature of nonvolatile memory is the reset command. Following a maintenance action, it is desirable to clear the memory prior to performing a confidence check. The reset command makes all previous stored information inaccessible.

Interrogation of BIT is performed by pressing pushbuttons either on the front panel of the BPCU (GPCU) or on a remote panel. A 24 character display, also on the panel, lists BIT information as shown in the following sequence:

EXTERNAL POWER SYSTEM IDENTIFICATION

BPCU or EXTERNAL POWER FAULTS (or OK)

LEFT GEN POWER SYSTEM IDENTIFICATION

LEFT GCU or GENERATOR FAULTS (or OK)

RIGHT GEN POWER SYSTEM IDENTIFICATION

RIGHT GCU or GENERATOR FAULTS (or OK)

APU GEN POWER SYSTEM IDENTIFICATION

APU GCU or AUXILIARY GENERATOR FAULTS (or OK)

FLIGHT IDENTIFICATION

PROMPT FOR PREVIOUS DATA REQUEST

The display first lists the channel being interrogated followed by a fault message (if any) identifying the event and/or failure. If the fault is in a control unit, the unit will be identified and then a code will identify the functional circuit that failed. For example:

EXTERNAL POWER SYSTEM

TIE BUS DP TRIP

BPCU FAILED CODE 60

In this instance, a tie bus differential protection (feeder fault) trip of the external power contactor has occured. The trip is due to a failure in the BPCU differential current sensing circuit. The channel, event, and cause have been identified. Replacement of the BPCU will allow the aircraft to return to service. Furthermore, when the BPCU reaches the repair facility, the technician will be able to immediately go to the differential current sensing circuit and begin troubleshooting there.

BIT is mandated to perform all of the above functions without interfering with the control and protection functions of the control unit. This is accomplished in the following manner. The BIT routine will monitor the circuit by reading data from control and protection RAM locations. If a fault condition is sensed, the routine will exercise the circuit and make the fault isolation. No tests initiated by BIT will result in any protective action. A BIT routine will in no way interfere with the contents of control and protection RAM. If temporary storage of data is required, separate RAM locations for BIT have been designated. Reading from or writing to the nonvolatile memory, BIT interrogation, and BIT display operate independent of all control and protection functions. Finally, any software related fault in the microprocessor circuit will initially result in the bypass of all BIT routines. In this way, if the fault is located in a BIT module or circuit, the primary functions of control and protection can continue to be performed.

B. Complex Control and Protection Schemes

In the system requirement analysis, it was determined that automation and weight reduction were to be major design considerations. To meet those needs, these advanced systems removed a significant amount of control switches and hardwired relay logic found in older systems. The relatively simple logic of switches and relays was replaced by complex logic expressions demanded by automated operation. The following example shows the advantage of microprocessor-based controls in the design of complex systems. Figure 6 illustrates this example.

First assume that both the discrete and microprocessor versions have M inputs and N outputs. All inputs and outputs are independent so the discrete version will require N logic processing circuits to produce N outputs. The microprocessor version has one logic processing circuit with the number of peripheral devices determined by the number of inputs and outputs. However, because of multiplexing, the peripheral number is not proportional.

The size (component count) in the discrete version's logic processing circuit will be proportional to the complexity of the control function. The memory requirements (RAM and ROM) of the microprocessor version increase with complexity but the memory devices can hold many control functions. The rest of the logic processing circuit remains the same.

Each discrete version control function with timing considerations requires a separate timing circuit. For most timing functions in the microprocessor version, software loops are adequate. Likewise, the discrete design relies on hardware to establish setpoints. Accuracy is dependent on the availability of desired component values, the precision of those values, and the component's environmental stability. A software setpoint can be adapted to a range of scaling factors. In addition, software limits can be tailored to take into account circuit variations due to component tolerances and stability.

In a simple system with few control functions, the discrete version has the definite advantage. The microprocessor version is saddled with the minimum part count overhead of the microprocessor circuit (outlined in the Detailed System Design). But, as the number of inputs, outputs, and variables in control functions increases, the microprocessor version has the advantage.

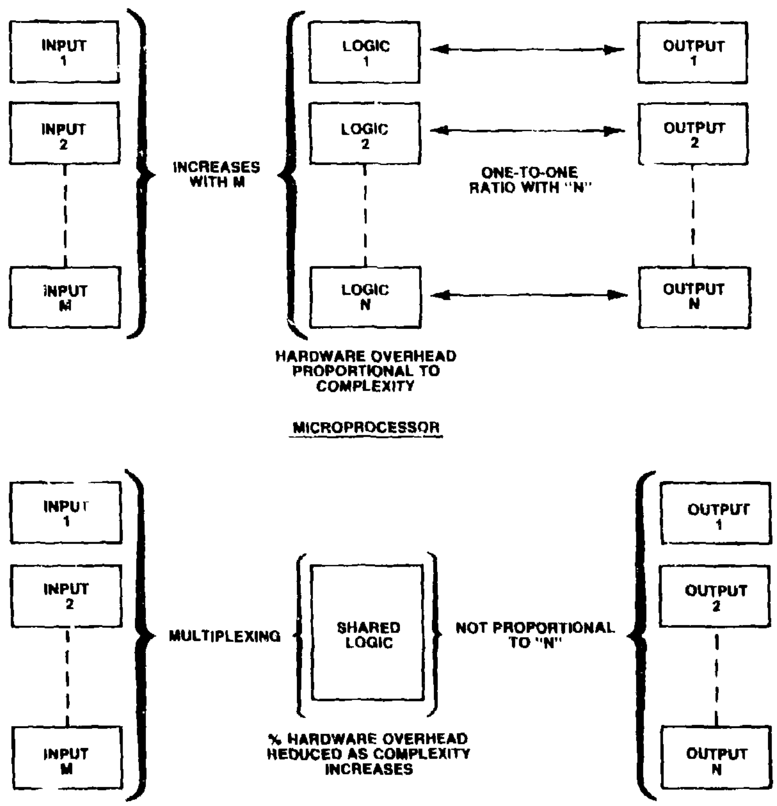

Figure 6 Discrete vs. Microprocessor Complexity Comparison

C. Flexibility

As discussed in Design Maintenance, changes can occur over the life of a system. Loosely interpreted, the impact of change on a design is proportional to the design's inherent flexibility. The implementation of microprocessors improves flexibility and thus reduces the impact of change on the design. The following example illustrates this advantage.

A Boolean expression representing a control function is implemented in a discrete design as a combination of logic gate devices interconnected by wires and printed wiring board traces. Any change to the expression results in a part and/or printed wiring board change. If the change involves the addition of functions, space limitations may prohibit its implementation. The change could only be incorporated as part of a total repackaging. If, on the other hand, the expression is implemented in software logic, a change in the expression could be accomplished by reprogramming. The addition of functions will only result in memory increases as long as no additional inputs or outputs are needed. The impact of changes is further reduced by providing adequate memory margins in the original design. Typically, the systems used here as examples were designed with margins of greater than ten percent. Another advantage is provided by technological advances. New memory devices with twice the capacity and pin compatibility are now available. In a recent product improvement of the 757/757 BPCU, the existing 2 kByte devices were replaced with pin compatible 4 kByte devices. The result was a doubling of memory (and logic) capacity with no increase in part count or printed wiring board redesign.

A major portion of the design effort for control, protection, and BIT functions is the establishment of analog signal set points and time delays. It has already been noted that the microprocessor-based control unit set points and time delays are not hardware dependent. This permits a wide degree of variability without parts changes. For instance, in an eight bit microprocessor system, an analog signal setpoint can assume one of 1024 proportional values (assuming a 10 bit A/D converter). Time delays can vary from 0.4 microseconds to infinity.

The BIT display is another control unit function where the flexibility advantage is evident. BIT messages can easily be altered in response to EPGS functional changes or to improve operator understanding. This is an obvious improvement over the inflexible pilot light fault isolation found in older systems.

IV. FUTURE DEVELOPMENTS

As the applications for microprocessor-based controllers expand, requirements are emerging for increased data processing rates, reduced costs, and minimal part counts. Data processing rate increases may be accomplished by enhancing the capabilities of individual processors or by utilizing multiple processors executing in parallel. With the principle cost of employing microprocessor-based control attributed to the generation, verification, and documentation of software, cost reductions are expected to result from the use of high level languages and the building of libraries of reusable software modules which can be used for several applications. As technological advances enable increased levels of functional integration on a single chip, the functions of many processor support chips are being incorporated into the processor chip, thereby eliminating the need for these devices and reducing system parts count.

A. Microprocessor Hardware

Since the time that microprocessors were first applied to aircraft electrical power systems, several new generations of microprocessors have come into the market place with significantly enhanced capabilities.

Preliminarily designated for incorporation in a system currently under development, the first of these new generations of microprocessors includes members such as the Intel 8086, the Zilog Z8000, the Motorola 68000, and others. Characteristics of these devices include 16 or 32 bit internal data processing structures and 16 bit data busses. These features increase accuracy and speed by reducing the number of bus transfers per instruction and reducing the time required for 16-bit arithmetic. Directly accessible address spaces range from 1 to 16 megabytes (8 bits per byte). They have clock frequencies approaching and exceeding 10 MHz and this speed advantage is enhanced by hardware multiply and divide features, data string handling instructions, and pipelining to fetch instructions asynchronously and in advance of their execution. An 8086-based central processing unit, including processor, clock generator, bus controller, address and data latches, EPROM, RAM, interrupt controller, chip select logic, and watchdog timer, which has been developed, tested, and proven functional, is illustrated in Figure 7 (5,6,7,8).

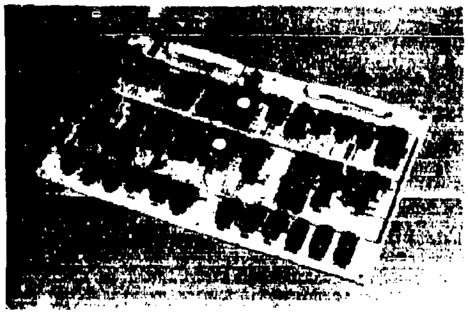

Figure 7 Microprocessor Board

The criteria discussed earlier (Section II-C) were applied when selecting the Intel 8086 16-bit microprocessor. The overriding concerns at the time of selection were the desire for 16-bit performance, the availability of hardware which meets military specifications, and the existence of support systems in-house.

Evolutionary extensions of this generation of 16-bit microprocessors are emerging. Intel's 80186 represents one such extension. Based on an enhanced version of the 16-bit 8086, the 80186 incorporates a clock generator, 2-channel direct memory access, interrupt controller, three 16-bit timers, memory, and peripheral chip-select logic and wait-state generator on a single chip. Depending on the system application, replacing the 8086 and its associated support chips with the 80186 could decrease the parts count by as many as 7 IC's and reduce the size of the board illustrated in Figure 7 by approximately one-fourth. Achieving system parts reduction as well as faster data manipulation capabilities, this device exhibits potential for inclusion in future aircraft generator control systems. (9)

As the capabilities of these new microprocessors expand, the demand for more program and data storage space increase accordingly. To accommodate these demands without yielding a significant parts count increase, memories are becoming more dense. Erasable programmable read-only memories, which generally store program code and constant data, are currently available with 128 kilobit/chip capacities. This represents eight times the density of the devices used in current systems. Scratch-pad memories, in the form of static random access memories, can provide 8 kilobits of storage area per chip. (10,11)

16-bit microcontrollers also have application in electric power system controls. Tentatively scheduled to become available in 1983, Intel's 8096 characterizes this generation of microcontrollers in terms of speed and functional integration on a single chip. On-board features include a 16-bit processor, RAM, ROM, analog-digital converter, universal asynchronous receiver/transmitter, priority interrupt controller, pulse-width-modulated output, software timers, input/output ports, a watchdog timer, and high-speed pulsed input/output. With this level of integration, the 8096 could effectively replace 12 chips of a typical 8086-based system, which would reduce the size of the board illustrated in Figure 7 by one third. (9)

In addition to developing faster processors with higher levels of functional integration, device manufacturers are beginning to address the issues of reducing power consumption and meeting military radiation hardening requirements. Power savings from 500% to 1000% are being achieved by applying CMOS technology to microprocessor development in place of the original NMOS and bipolar technologies. American Microsystems S99C91 CMOS 16-bit microprocessor and Harris' CMOS version of the 8086 are scheduled to be introduced in 1983 (12).

As the scope of military applications for microprocessor - based controllers expands, radiation resistance will be a requirement. The CMOS and HMOS technologies used in 16-bit processors are more resistant than the NMOS technology used in their 8-bit predecessors. This may extend their applicability to manned military aircraft. Other microprocessor families are also available based on technologies which can survive in environments much harsher than those present in manned systems. The microprocessors include the TI SBP9900, the Fairchild 9445, and so forth, using bi-polar, I^{2}L, and CMOS-SOS technologies. (13)

B. Software

The use of microprocessors to execute electric power generation control functions allows the implementation of more complex controls without increasing the hardware design effort. On the other hand, the development of control software has become a major undertaking. Several improvements in software design methods have been used in existing systems. These improvements include the implementation of hierarchical design techniques and formalized verification, validation, and configuration control techniques. Two more techniques, effective in reducing software costs, are the use of existing software modules as "building blocks" in the construction of subsequent systems and the use of high-level languages. (8)

The building block technique reduces the amount of software which must be developed, verified, and documented when creating a new system. For instance, 60-70 percent of the software modules for the 767 generator control unit are applicable for use in a developmental 737 GCU.

Existing systems have been implemented using assembly language which has advantages in that it can be used to generate modules which have been manually optimized to be efficient in the use of processing time and computer memory. These advantages are becoming less critical now that processor speed and memory density have been increased.

The use of high-level languages also has several advantages. A typical high-level language instruction may replace 5 to 10 assembly language instructions. This, coupled with the fact that high-level language programs are more readable, results in increased programmer productivity as well as reduced debugging, maintenance, and documentation efforts. (8)

In the development of an experimental microprocessor based voltage regulator for a wound field synchronous generator, the high-level language, PLM-86, was utilized. It was found that the time required to generate the software was greatly reduced. The time required to code and debug the algorithm was estimated to be reduced by 4 to 1. However, it was also found that the output of the language's compiler was insufficiently optimized for high speed execution and some of the processor's (Intel 8086) advanced features such as hardware multiply and divide were not adequately utilized. As a result of this, portions of the regulation algorithm were eventually re-written in assembly language.

The present strategy being used in the development of experimental generator control unit software is that a high-level language will be used for all non time-critical functions and that critical functions requiring high-speed execution will be implemented in assembly language routines embedded in the main program.

It has also been found useful to initially code, compile, and debug even high speed modules in the high-level language and then re-write them in assembly language based on the high-level language compiler output. The re-writing process allows manual optimization where necessary. This approach speeds the initial stages of the coding and debugging process. The high-level code serves as a "pseudo-code" for documentation purposes after the transition to assembly-language has been made, and the structure of the high-level language tends to be forced on the final assembly language routine resulting in a higher quality product.

The high-level language selection for the experimental system was dictated by availability at the time of selection. At present other languages such as C and Pascal might also be considered. These might have advantage in that they may be more portable across various microprocessor families. Two considerations appear to be overriding in language selection. The first of these is the requirement for operating in a high-speed real time environment. The other is the requirement to operate with existing support and development systems.

In addition, the high-level language, ADA, is expected to be mandated for use on military projects. It is also expected to be a more desirable choice than existing languages for performance reasons. Compilers and support environments are as yet not readily available, but ADA's presence on the horizon tends to limit commitments to other languages at this time.

V. FUTURE APPLICATIONS

Increased microprocessor capabilities and increased language capabilities make possible the implementation of several new features in aircraft electric power controls such as closed-loop digital control of voltage regulation, generator speed control, paralleling, and expanded built-in-test and fault tolerance; digital data sensing and communications; and others. Implementing closed-loop control functions in software provides several advantages. Complex control algorithms which may include non-linear or adaptive functions may be added without corresponding increases in part counts. It is possible to pass information between interacting loops. Component tolerances and drift don't affect performance. Accuracy and reliability are high. Finally, it is possible to modify control functions and parameters during development, without changing hardware design. (8,14)

A. Implementation Examples

One application for microprocessor control in an aircraft electric power generating system is voltage regulation. Figure 8 is a block diagram which functionally describes a simple voltage regulation implementation. The voltage at the point of regulation (POR) is sensed and compared against a reference voltage to produce a voltage error. To improve transient response and minimize steady-state error, this error signal is subjected to constant gain compensation and an integrator with gain for lead compensation. The integration is accomplished in software by summing the current voltage error with the error from the previous sample and applying a constant gain to the result. If the resulting current command exceeds pre-determined limits, it is forced to either its maximum or minimum (zero) values as is appropriate. It is then passed to an amplifier with a current control loop implemented in hardware to control the exciter field current.

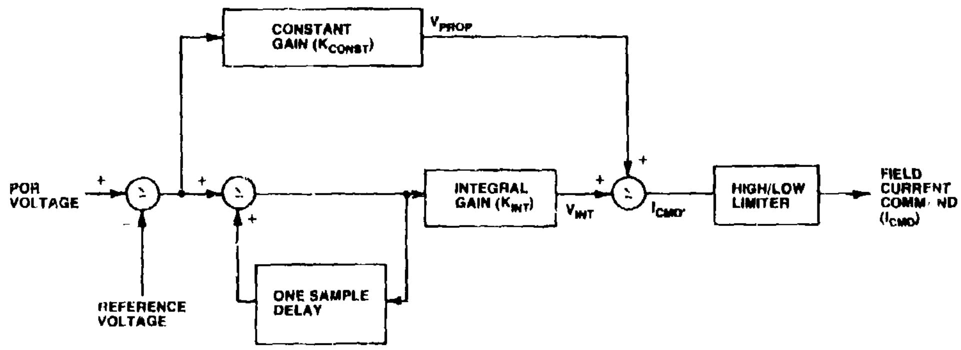

Figure 8 Voltage Regulation Implementation

Figure 9 is a flow chart showing how the control algorithm may be implemented in software. Based on preliminary simulation testing, this algorithm was found to execute in approximately 700 microseconds on a 5MHz Intel 8086. To achieve high quality voltage regulation, an update rate of 1600 microseconds is required. Thus, it is feasible to implement this control function in software with 60% of available processor time free for the execution of other functions. Should the system control requirements expand, employing a faster processor could significantly reduce execution time.

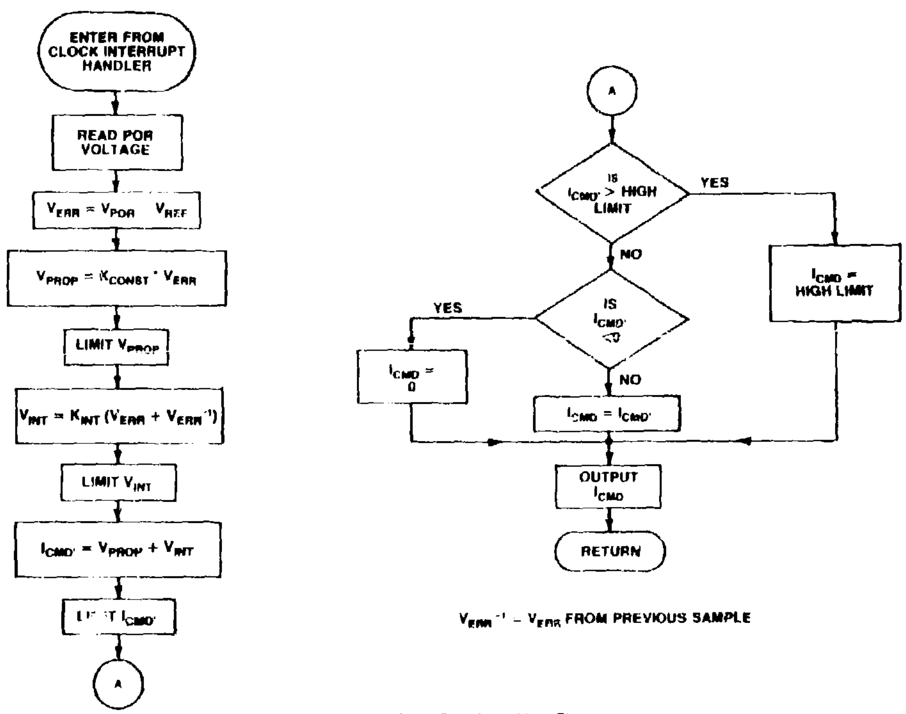

Figure 9 Voltage Regulator Flow Chart

Preliminary closed-loop voltage regulator tests have been run on a test system to verify the operation of this control algorithm. The results presently available do approach the desired steady state and transient responses. Further development is required. With a microprocessor-based controller, modifications are generally simple, however, involving software changes with little or no impact on hardware design.

In addition to digital control of the generator output voltage, a microprocessor can also control generator output frequency by trim head bias of the flyball governor or with a servo valve to control the constant speed drive between the engine and the generator. The advantages of this control means over the standard mechanical governor include fine frequency control, minimum frequency transients, and a reduction in moving parts.

With the availability of both voltage and speed control within the microprocessor, phase-locked paralleling of multiple generators can be implemented. Real and reactive load division error signals can be used as inputs to the control loops to balance loads between generators. This can all be accomplished with few additional parts.

While the potential for implementing a microprocessor-based controller is evident, a microprocessor system can also handle the built-in test feature of electric power generating systems. The advantages of this type of implementation have been proven in the field by reduced maintenance time and the generation of records of system performance. Enhancements in this area still have potential for continued reduction in mean-time-to-repair, unnecessary line replacement units (LRUs), and life-cycle costs.

Enhancements which may be considered, and which are made possible by the increasing capabilities of microprocessors and digital memories, include the following.

Additional sensing elements and more sophisticated logic can improve fault isolation capabilities.

Additional memory and more capable display devices can improve communications with maintenance crewmen. Messages can be more specific and readable and more information can be provided. It may be possible to provide "expert" systems which not only record and isolate faults, but which direct crewmen through further troubleshooting procedures when complete isolation is not immediately possible. These systems can also provide the operator with information such as LRU installation and removal procedures.

A means of implementing such an "expert" system could possibly be through a test box available at ground locations which would have a high-quality graphics display and which interfaces to BIT hardware and software within the flight equipment.

The availability of serial interface equipment such as the Arinc 429 MIL STD 1553 type communications bus allows built in test information to be passed to central aircraft health monitoring equipment. Other interface equipment could be present at ground stations to pass information to reliability monitoring computers at some central location. Here it could be processed to compile reliability data or to scan for repeated fault conditions throughout a fleet of aircraft and to flag these occurrences for special attention and corrective action.

In summary, the implementation of electrical power generating system control functions in software with microprocessors will provide for sophisticated and precise control and built in test capabilities. The hardware architecture of such a controller will depend on the quantity and complexity of system requirements.

B. Sample System Architecture

Figure 10 illustrates the architecture of a microprocessor-based generator control unit (GCU) currently under development, which can be incorporated in a parallel, integrated drive generating system. Control and protection functions are split between two independent processing systems. The control processor assumes responsibility for voltage regulation, generator speed control, and the coordination of generator paralleling, while the protection processor executes the protective and built-in test functions. Each processing system is comprised of independent input and output signal conditioning, local program and scratch-pad memory, and a watchdog timer to verify sequential and uninterrupted program code execution. Hardware is included to accommodate parallel interprocessor communications.

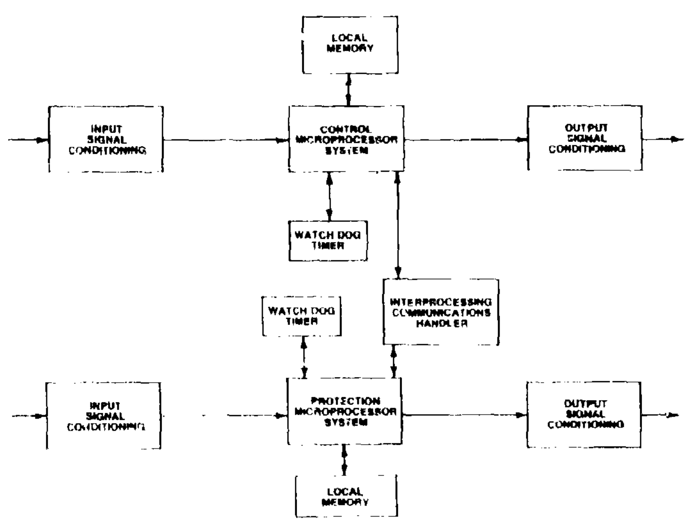

Figure 10 GCU Architecture With Two Independent Microprocessor Systems

This approach has several advantages. Dedicating each processor to a limited number of specific tasks with related update rates simplifies hardware and software requirements, thereby tending to minimize parts count and software development time. Furthermore, with each independent processing system monitoring the status of the other, no single failure in either system should exist undetected.

Though seemingly inappropriate for some current generation aircraft electric power generating systems, several alternate microprocessor-based GCU architectures appear viable for systems with varying requirements.

A minimum parts count can obviously be achieved by employing a single microprocessor system to execute both control and protection functions. The trade-off for the parts reduction is a reduction in throughput capacity, however. As a result, the feasibility of this approach depends on the complexity of system requirements and the speed of available processors.

The expansion of the throughput capacity of a GCU beyond the level attainable with a dual processor split protection and control scheme can be accomplished with a distributed processing scheme. One implementation of distributed processing, featuring a front end processor which performs all I/O interface processing and functions as a task master to divide tasks between the general data processors, is illustrated in Figure 11.

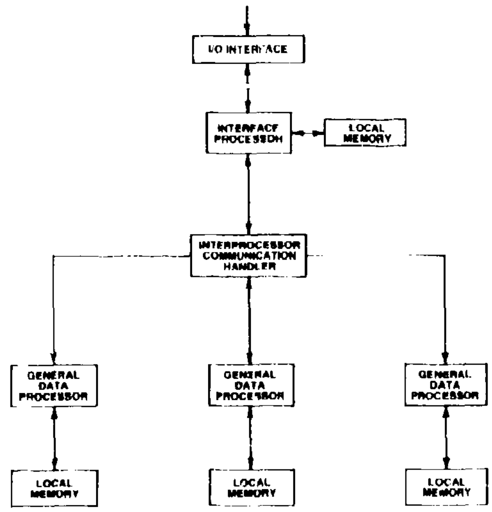

Figure 11 GCU Architecture Implementing Distributed Processing With Front End Processor

With multiple slave processors capable of performing any data manipulation task, this architecture has several advantages. The throughput of the system can be expanded by the simple addition of a general data processor and its associated support chips. Furthermore, in the event that a general data processor fails, the other slave processors can assume its responsibilities with a reduction in the update rate as required.

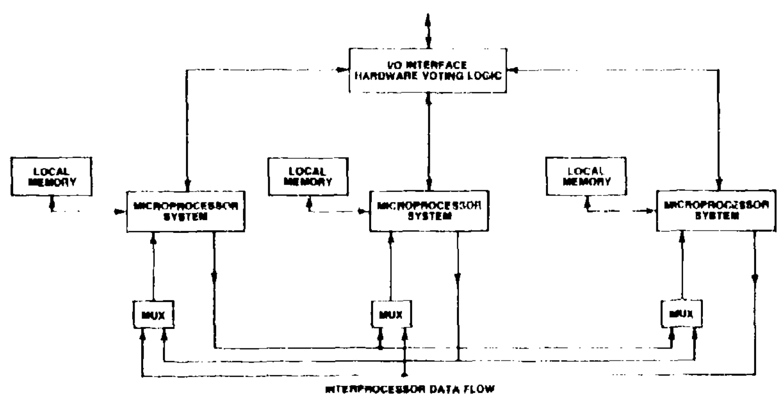

Figure 12 GCU Architecture Implementing Distributed Processing With Software Supervisor

Despite this improvement in fault tolerance, system functioning is ultimately dependent on the status of the front-end processor. Relying only on self-testing for failure detection within the front-end processing system, there is no guarantee of single failure detection. Moreover, in the event that a failure is detected, system architecture does not include a replacement system to assume responsibilities.

These weaknesses in fault tolerance and detection are overcome by the distributed processing scheme illustrated in Figure 12. In this architectural configuration there is no supervisory hardware module to fail. The role of supervisor exists in software resident in all processors. Redundancy is achieved by assigning all tasks to at least three proce sors and comparing the results with hardware voting logic. In the event of a processor failure, the supervisory software reassigns the responsibilities of the failed processor to the other remaining processors. This allows continued operation until the number of failed units precludes reconfiguration or until the remaining units are so overworked that throughput is degraded to an unacceptable level. (15)

The trade-off for the increase in throughput capacity, expandability, and fault tolerance achieved by distributed processing is a substantial increase in parts. Thus, the data processing demands and fault tolerance requirements must be substantially stringent to justify employing this scheme.

SUMMARY

Microprocessors are proving themselves in s veral advanced electrical power generating system control applications. Their success in handlin mplex EPGS requirements add credibility to the decision to select microprocessors over discret gic.

To implement microprocessors in a control unit design, a structured design approach is used. This approach divides the effort into the following major steps: requirements analysis, specification, architectural design, detail design, implementation, and maintenance. Hardware and software functions are broken down into single action, highly independent functional blocks. Standardization and consistency in the design of these blocks is stressed. The net result is the creation of reusable functional blocks that can be used in multiple applications. Duplication, in turn, results in improved reliability, reduced testing, and an overall increase in design efficiency.

Inherent to the microprocessor are the advantages of complex logic and timing capabilities and flexibility. In the control units, the extensive BIT coverage, control and protection schemes, and ease of function modification illustrate these advantages. As the complexity increases, the nonrecurring costs of software development are overshadowed by the recurring costs of producing an equivalent discrete logic alternative.

Projected advances in microprocessor hardware and high-level software languages will yield improved data handling capabilities as well as reduced parts counts and software development costs. As a result, future generations of EPGS's will use microprocessors to perform control functions of increasing complexity such as voltage regulation, speed control, and paralleling. While the complexity of the control functions increases, the parts count will be reduced by the introduction of single chip microcomputers and memory devices with expanded storage capacity.

As the quantity and complexity of the functions performed by microprocessor-based EPGS control units increases, new hardware control architectures may be pursued. Multi-processing, distributed processing, and advanced communication schemes offer increased throughput capacity, ease of expansion, and improved fault tolerance. This, in turn, improves the overall performance, reliability, and maintainability of the EPGS.

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DISCUSSION

G.B.Toyne, UK

You stated that the incorporation of modifications by software changes (rather than hardware) would reduce testing time. However, to obtain flight clearance requires proper testing and demonstration whether changes are in hardware or software. Does this mean that you have some other acceptable method of validating the software?

Author's Reply

By incorporating a change in software, rather than in hardware, testing at the control unit subassembly level can be minimized. This reduces the response time to a proposed change. It is agreed that functional system tests for the purpose of qualification would be required.

E.H.Warne, UK

How do you envisage the problem of control of software modification especially in high level languages? Furthermore it is essential to maintain full visibility in such systems.

Author's Reply

The concern for control of system function modification is real. As pointed out in the paper, a 2-thrust approach is necessary. Primarily, a structured design approach greatly reduces the possibility of error by imposing strict rules on the programmer. Likewise, written code is subjected to a predetermined test structure. Second, the building block approach enables the designer to re-use previously verified software. The structured design approach and the portability provide the full visibility required.

Ph. Ramette, Fr

Parmi les développements futurs de l'application des microprocesseurs au contrôle des systèmes de fourniture d'énergie électrique, vous avez mentionné l'utilisation de language de haut niveau par lesquels le nombre d'instructions est réduit par rapport aux languages actuels. Pourriez vous donner des précisions sur ces languages de haut niveau?

Author's Reply

The question is: Could I provide more information about high level languages?

We are currently using PLM, a high level language for the INTEL microprocessors. It was chosen because of its availability at the start of our development as well as its compatibility with our existing support systems. In the future we may consider ADA which is promoted by the US. It should be available in a year or so.

400 Hz GENERATORS EVOLUTION, EFFECTS ON INSTALLATION ARRANGEMENTS

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SUMMARY

The need for 400 Hz electric power keeps on increasing on airplanes. Up to now the way how to obtain constant frequency from variable speed drive is either hydro or electro-mechanical. New conversion systems are arising : they take benefits from the most recent improvements and developments in high speed electrotechnics and power electronics.

Rotational speeds running at more than 24000 r/p/m. are now accessible to the on-board generator and consequently a significant improvement of power-to-weight ratio is obtained. The installation of such a generator on aircraft entails an high speed gear which can be combined with it or, better, installed in the accessory gearbox. The cooling of the whole system can be realized by air or, better, by oil electronic components included. Optimum design leads to share the oil circuit of the gearbox with the generator cooling system.

INTRODUCTION

400 HERTZ Constant Frequency spread nearly to all AC main electric systems on military and civil airplanes. The choices result from analysis made by design offices of aircraft manufacturers, for every new aeronautical program and many factors have to be considered mainly :

- power quality
- integral weight of the whole system i.e the generator and the complete electrical equipment concerned
- economical points of view.

The balance leads very usually to choose a main system with 400 Hz constant frequency even though some consumers do not strictly need constant frequency. This tendency is more especially frequent as the power demand on-board is important.

For a generator supplying main electric system, a mechanical transmission in relation with the speed of the engine is used : this transmission is generally available on an accessory gearbox. Therefore, the on-board generator must be able to convert mechanical energy at variable speed (directly due to the normal engine speed) into electric energy, stabilized in voltage and frequency (according to MIL G 21480 (1) and MIL STD 704 (2) regulations).

The range of speeds available shows variable ratios between maximum and minimum speeds, depending on the type of the engine. Generally speaking the ratio is near by 1.5/1 for turboprops, varies from 1.9 to 2.5/1 for turbofans and can reach 3/1 for some turbojets.

TODAY'S STATUS OF 400 Hz GENERATORS

The systems being usually available today contain an AC generator driven at constant speed through a mechanical device inserted between the variable speed drive and the generator, the principle of which being either hydromechanical or electromechanical :

- The well known hydromechanical conversion is mainly developed by SUNDSTRAND ; constant speed drives are supplied either separated from the generator (HD and AGD system) or combined with it as realized in the IDG and IDGS systems.
- Another system based on an electromechanical conversion is developed by AUXILEC in France ; the system known as "AUXIVAR" is entirely integrated, and may include a self contained gearbox if the speed ratio exceeds 1.6/1. The AUXIVAR vario alternators are well fitted for large speed ratios up to 3/1.

With the hydromechanical conversion, the drive is cooled through an oil circuit by an oil-fuel heat exchanger which is a part of the aircraft installation. The AC generator is either air-cooled or, and in any case with the IDG system, oil-cooled, particularly with oil-spray. When the generator is oil-cooled, there is preferably only one oil-circuit shared by the drive and the AC generator.

With the electromechanical conversion, the system is generally air-cooled, up to now. The ram-air is provided either from a scoop fitted on the aircraft "skin" or in the engine air-intake. The installation of such an equipment is really simple as no extra exchanger is needed.

(1) Military specification : Generator system 400 Hz alternating current, aircraft general specification for.

(2) Military standard : Aircraft electric power characteristics.

The AC generators convert the constant speed mechanical power into constant frequency electric power. Their rotational speed gradually increased in the past from 6000 r.p.m. to 8000 r.p.m, after then from 8000 r.p.m. to 12000 r.p.m. Thus, the power to weight ratio and size ratio of the generators largely improved.

NEED FOR HIGHER SPEEDS

As a typical example, the weight improvements of 60 kVA 400 Hz generators, which were achieved during these last 20 years, throw light upon the steps of this evolution :

6000 r.p.m.	49 kg	1,2 kVA/kg
8000 r.p.m.	36 kg	1,6 kVA/kg
12000 r.p.m.	20 kg	3,3 kVA/kg

The evolution is similar with constant speed drives. For instance, for a 60 kVA IDG drive rated for a 2/1 speed ratio, when the minimum input speed changes from 3700 to 9400 r.p.m, the drive weight decreases from 29.4 to 20.9 kg (8.5 kg improvement, nearly 30 %) . Same results are obtained with electromechanical vario-alternators : a speed change of the AC generator from 8000 r.p.m. to 12000 r.p.m. allows for an increase of 50 % of the drive speed. Then, for two same sized machines, a power increase of 33 % and a decrease of 12 % in weight can be obtained.

Obviously, the evolution in cooling technologies and the use of new materials also contributed to this increase in power to weight ratio. But the speed is and will remain the real significant factor of improvement, because size and weight of an electric machine (and consequently of the drive) is closely related , with the electromagnetic torque involved in the power conversion, the value of which varies inversely to the rotational speed, for a given power. The electromagn ic size of the generator is then roughly given by the relation :

$$\left[D^2 L\right] \sim \frac{\text{POWER}}{\text{SPEED x B.A}}$$

with :

B : flux density in the air-grap

A : peripheral ampere conductors per unit of length (armature lineic load).

BEYOND THE 24000 r.p.m. LIMIT

Therefore today's and tomorrow's evolution is highly related with the increase of rotational speeds. But, at first, a particular attention is to be paid to polarity and frequency. As a matter of fact, it is not possible to increase by short steps the speed of the 400 Hz AC generators, the reason being inherently related to the constancy of this 400 Hz. The possible speeds, in inverse ratio to the number of pairs of poles, must follow discrete series : at 400 Hz, the milestones are 6000, 8000, 12000, 24000 r.p.m. (respectively with 4, 3, 2 or 1 pair (s) of poles). 400 Hz AC generators fitted on the more recent aircrafts are driven at 12000 r.p.m.

The "jump" to 24000 r.p.m. is already made by some manufacturers such as LUCAS AEROSPACE which presents a new range of compact generators with a target of 2 kVA/kg. These CCFG generators have a very attractive power to weight ratio, nevertheless one question remains : what will be the future evolution beyond the 24000 r.p.m. limit, closely tightened to the 400 Hz frequency.

In fact, a response can be found in the tremendous research investment afforded for many years by manufacturers of electrotechnical devices and electronic components, investment altogether shared by research workers in the University laboratories. All these joint efforts have led to the electronic means of power conversion, particularly General Electric and Westinghouse have worked a lot in this field, AUXILEC in France is also much concerned with solid state power systems. Examples of this technology are already demonstrated on military aircraft (A4, F/A-18, AV-8B, F5G).

The proposed systems are established on the association of an alternator with an electronic frequency changer. The alternator is driven at variable speed and therefore provides a variable frequency AC power. This new concept allows to relieve the alternator of the 400 Hz burden and to increase significantly the power to weight ratio. This result is obtained not only because the alternator is driven at a high rotational speed, but also because it delivers AC power at high frequency (f) with a high peripheral velocity in the air gap (V) in order to give more valuable advantages to high speed technologies.

HIGH FREQUENCY ALTERNATORS

The point is that increasing the polarity of the alternator instead of decreasing it (as mentioned here above) leads to reduce drastically the dead electromagnetic parts such as stator and rotor yokes, windings overhangs and field coil ends as shown on fig. 1. It can be easily understood in comparing the morphology of 2, 4, 16 poles machines and their respective weights (see fig. 2) :

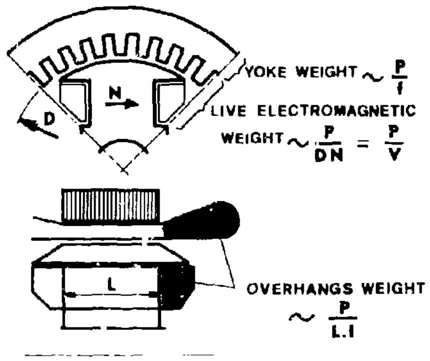


Figure 1 : Frequency and peripheral velocity influence on weight

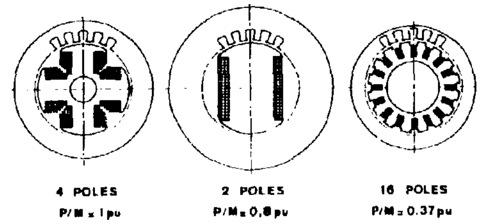


Figure 2 : Comparison of 4, 2, 16 poles machines

For different machines delivering the same power, the weight of active electromagnetic parts is approximatively with the same technology :

400 Hz	2p = 4	N = 12000 r.p.m.	Weight = M
400 Hz	2p = 2	N = 24000 r.p.m.	Weight = 0.8 M
800 Hz	2p = 4	N = 24000 r.p.m.	Weight = 0.55M
3200 Hz	2p = 16	N = 24000 r.p.m.	Weight = 0.37M

Frequencies as high as 3200 Hz are henceforth admissible with 50% cobalt laminations , .1 mm thick and quite compatible with modern cooling technologies such as oil-cooling.

HIGH PERIPHERAL VELOCITY

As shown on fig. 1 again, the live electromagnetic weight is in the inverse ratio to the peripheral speed and from another point of view it is also convenient to note that the alternator is tied with static commutation circuits and must be conceived with relatively low commutating inductances all the more as the system is due to operate at high frequency. This emphasizes interest for high peripheral speeds. It is well understood that the mechanical stresses of the rotor vary as the square of the peripheral speed ; in addition, problems of critical speed have to be taken into account. However, peripheral velocities near by 240 say 250 m/sec are now accessible within the power ratings in use on aircraft, either with wound rotors (W.R) or with permanent magnet (P.M) machines.

P.M. AND W.R. MACHINES

The use of permanent magnets in AC generators had been till now restricted to Auxiliary Permanent Magnet Generators of a few hundred watts, inserted within main generators. In this particular field, the recent advent of Samarium-Cobalt Magnets with their very high energy density up to 200 kj/m3 as shown in fig.3 (UGIMAG RECOMA 25) and their immunity to demagnetizing effect gave wings to another kind of AC power generation for aircraft application.

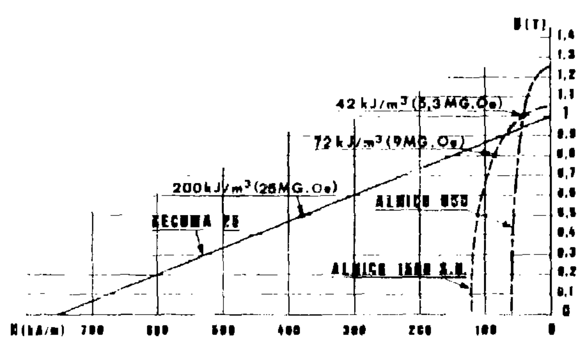


Figure 3 : Compared performances of Samarium-cobalt Recoma 25 with metallic Alnico magnets

It is now possible to conceive High speed Permanent Magnet Generators of some ten kW ; they are very compact in size and weight and in this respect highly competitive with wound rotor machines which remain fonctionnally more complex. As brushless machines, P.M. generators do not need a 3 - level construction as the latter with exciter and auxiliary P.M.G. do. In addition no cooling means is required to cool the rotor. This gives considerable simplification to the rotor/shaft architecture.

In order to illustrate the performances offered by the more recent high speed technics it is possible to mention two realizations achieved in this field by AUXILEC.

Potential performances at the highest speed allowable	Rated power	Overload	Weight	Rated power to weight ratio
P.M. alternator	90 kW	180 kW	12 kg	7,5
W.R. alternator	170 kVA	340 kVA	20 kg	8,5

FREQUENCY CHANGER

The electronic frequency changer associated with the high speed alternator may refer to different structures and commutation technologies :

Direct inverters such as cycloconverter (fig 4) deal directly with the waves delivered on the phases of the variable frequency alternator.

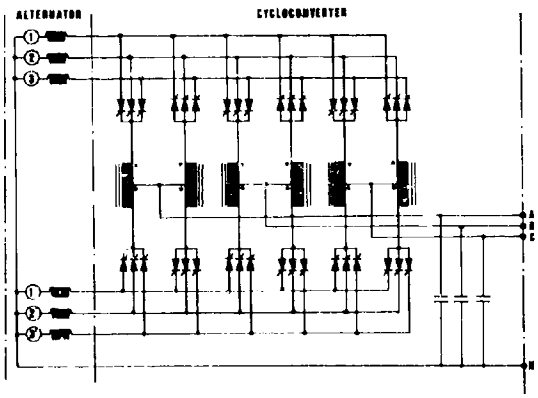


Figure 4 : Cycloconverter diagram

Indirect inverters such as rectifier inverter (fig. 5) call for an intermediate DC linkage between the alternator and the frequency changer.

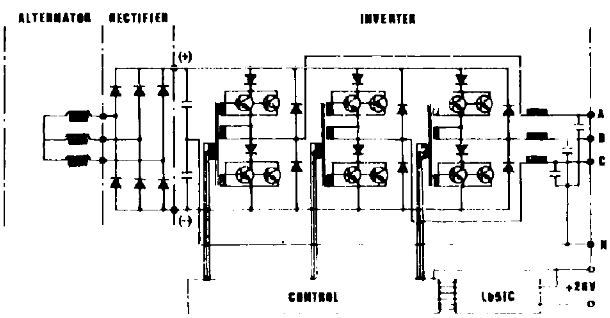
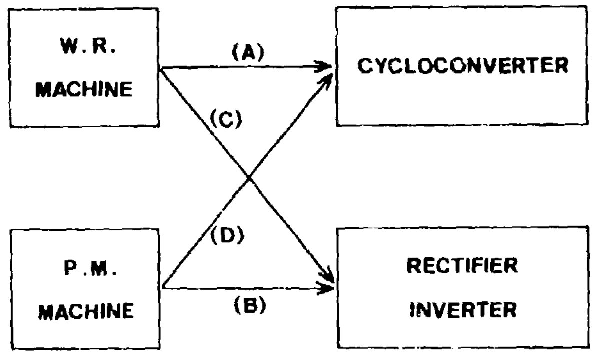


Figure 5 : Rectifier-inverter diagram



Consequently the association of the alternator with the electronic inverter can theoretically take on four main different forms, shown in fig. 6 : and referenced (A), (B), (C), (D)

Figure 6 : The four different forms of association alternator - electronic inverter.

The more advantageous of them are surely the interlinkages ref. (A) and (B) i.e wound rotor (W.R) machine with cycloconverter and Permanent Magnet (P.M) machine with rectifier- inverter.

A main reason for this derives from the fact that, due to field control, a wound rotor alternator, can provide the amount of reactive energy required through the cycloconverter without weight penalization, whereas a P.M. alternator shows its highest performances when supplying only active power as it is with rectifier circuits.

A competition is open between the two conversion systems : their respective chances mainly depend on power rating, speed ratio, cooling conditions and on the availability of the electronic components, either thyristors or power transistors, according to direct current, reverse voltage, turn-off time or commutating frequency characteristics.

Whatever may be the electronic way of conversion selected the system shows a very attractive flexibility as the electronic sub-assembly can be installed on the aircraft separately from the alternator as well as integrated with it in a same package.

The integrated design can take advantage of a unique oil-cooling circuit shared by the alternator and the inverter.

When these sub-assemblies are separated on the aircraft, their cooling systems can be chosen different for instance oil-cooling for the alternator, air-cooling for the inverter. In any case however the junction of the semi-conductors will not be exposed to a temperature exceeding 120°C.

To give an idea about the weight of different types of inverters conceived by AUXILEC, the following figures are proposed hereunder for integrated oil cooled items :

Direct conversion with thyristors :

40 kVA	(80 kVA overload)	: 20 kg
60 kVA	(120 kVA overload)	: 23 kg

Indirect conversion with transistors + diodes :

30 - 40 kVA	(60 kVA overload)	: 16 kg
75 - 90 kVA	(150 kVA overload)	: 27 kg

PERFORMANCES

Which advantages are expected from the new systems and which problems are encountered when installing them on airplanes ?

With respect to the intrinsic performances, it is obviously possible to compare the respective merits of these NEW high speed/solid state power systems with the "traditional" hydro- or electro-mechanical ones, which have been largely experienced for more than 25 years. However, both technologies are improving, in favor of an ever increasing power quality and of an enlarged power to weight ratio. But it has to be pointed out that it is in the area of high power electronics and high speed technics that the more recent progresses have been the more significant.

Comparison arguments are not emphasized in the limits of this document ; the reader will refer to the standards defining the power quality of electronic systems such as AIR 2300I B dealing with VSCF systems. However, here follow some outstanding points which have to be examined :

- High speed/electronic conversion systems can be used, with no restriction, in all the range of unit power ratings usually involved on aircraft from 20 to 90 kVA, further if necessary.

- Besides, from their principle, these systems are perfectly well fitted for running either on independant or paralleled channels with synchronized phases whatever may be the load or speed changes.

- The field of allowable speeds referring to generator power rating is based upon the technology to be worked up for the rotor and shaft constiuction from which derives the authorized peripheral velocity of the rotor considered as the mechanical fundamental parameter (Ref. to Appendix).

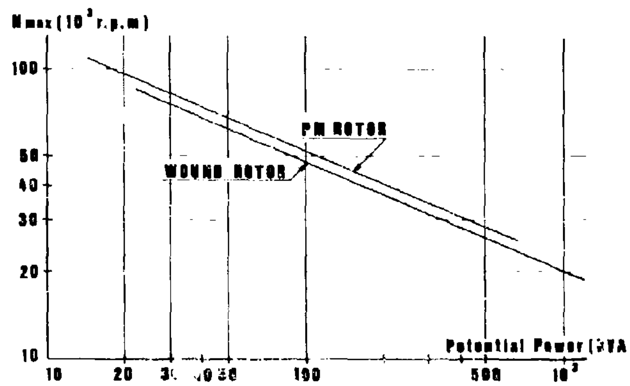


Figure 7 : Potential availability of high speed electric power for aircraft alternators

With a per[illegible] velocity of 240 m/sec or so the field spreads as it is shown on the diagram fig. 7 in which the m[illegible] rotation[illegible]l speed is given in terms of the "potential rated power" of the generator which is noth[illegible]se but the product of the rated power available at the lowest speed into the extreme speeds ratio [illegible]ther words it defines the power availability at the highest speed.

For ex[illegible] 60 kVA rated power generator running within a speed range of 2/1 ratio would result in a 120 kVA p[illegible]t[illegible]al rated power at the highest speed : then a rotational speed up to 45000 r.p.m. or 49000 r.p.m. an[illegible] d respectively with W.R and P.M machines. Inside this field, the driving speeds are always [illegible] under-critical, bearing in mind that variable speed operation needs to reject the first critical sp[illegible] and the operational maximum speed.

HIGH SPEED GEARBOX

As a matter of fact, such high speeds require an adaptation of the existing driving means with the interposition of a step-up gear which can be separated from or included in the generator or preferably in the accessory gearbox (A.G.B.). Operation with a conventional A.G.B fitted with a generator incorporating its own step-up gear would lead to a double mechanical power transformation with, as a result, a weight and efficiency penalty. On the other hand, combining directly the high speed gear into the A.G.B would lead presumably to a lighter design for the whole as the cinematic weight is approximately proportional to the transmitted mechanical torque wh[illegible]h decreases as the speed increases. If such a gear is not available in the A.G.B, a space and weight provision is needed in the generator for mounting a step-up gear inserted between the A.G.B pad and the so called generator.

Then the additional weight to forecast is about 2.4 kg for a 40 kVA generator and 3.2 kg for a 90 kVA generator with a set-up ratio of approximately 3/1.

ALTERNATOR COOLING

So far, the emphasis has been put upon oil cooling of the rotating machine and of the inverter.

Oil cooling is fully justified by the proofs given for many years on 8000 and 12000 r.p.m. alternators. The use of increasing rotational speeds confirms the choice of oil cooling for high speed rotating machinery as oil is needed for ball bearings lubrication. More, when hydrodynamic bearings are to be used at very high speeds, oil is thoroughly essential. But here oil cooling means oil circulation exclusively, the machine operating in a dry carcase. It must be pointed out that with high rotational speed it comes useless to proceed with oil spray techniques under pain of reaching unacceptable levels of drag loss as these losses vary according to a cubic law with the peripheral speed i - e :

drag loss $\sim V^3 d$ (HDL) f (Re)

with d , average density of air/oil

f (Re), in term of Reynolds number in the air gap.

So oil spray will not be used beyond a peripheral speed of about 120 m/sec in the air gap.
With oil circulating in appropriate internal ducts it is usually possible to accept an oil temperature up to 130°C for continuous operation without significant reliability alteration.

INVERTER COOLING

For the inverter sub-assembly a forced convection of air can be used as far as the air temperature allows it. On subsonic aircraft when the inverter is remotely located consideration can be given to a direct air cooling system : ram-air cooling is available in flight, and on ground either a fan or a suction effect from the engine air intake provides the necessary air flow.

It is also possible to take advantage of oil cooling for the power electronic components. Oil ducts machined in the near vicinity of the thyristors or transistors give considerable reduction to junction to oil thermal gradient. Thermal resistances down to .25°C/W can now be achieved and as a result, a better reliability is obtained for the inverter with an improved power to weight ratio of the whole power components. In addition, for oil to fuel heat exchanger lower difficulties are encountered. Nevertheless oil temperature must remain compatible with the maximum temperature allowable of the semi-conductors : then a permanent oil temperature of 90°C will be considered as acceptable and for short duration excursion to 110°C will be possible.

Taking into account the difference between the thermal limits allowed for alternator and inverter cooling oil, one could be tempted to design the cooling system with two separate circuits with two temperature levels. As an example a 60 kVA cooling system could be specified as follows :

	Permanent max oil temp.	Individual maximum heat rejection	Individual oil flow
Alternator	130°C	18 kW	18 l/min
Inverter	90°C	6 kW	6 l/min

Such a dual system although perfectly rational would duplicate pumps, exchangers and all the plumbing hardware.

On the other hand care for simplicity and for weight and cost alleviation would give preference to a unique oil system common to the two sub-assemblies mainly when they are integrated in the same frame. Then the 60 KVA cooling system taken above as an example will be specified as :

	Permanent max oil temp.	Total maximum heat rejection	Total oil flow
Generator	90°C	22 kW	22 l/min

In order to provide sufficient lubricating and cooling oil flow to the rotating machine and other built-in sub-assemblies such as the electronic inverter and the eventual step-up gear, basic hydraulic devices with external heat exchanger must be foreseen.

DESIGN FOR AUTONOMOUS OPERATION

For autonomy purposes, these devices may be considered as part of the general architecture of the generator. More benefits can be derived from an hydraulic system shared between the drive gearbox and the generator. With the self contained construction shown in fig. 8 in addition to the oil pump and pump gearing, all the usual hydraulic hardware must be included in the generator such as oil filter, relief valve, sump, oil-level indicator and so on.

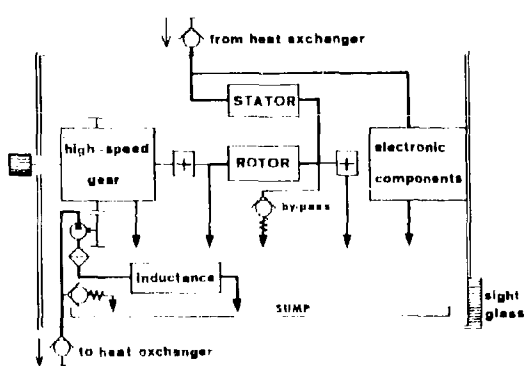


Figure 8 : Generator internal oil system for autonomous operation

DESIGN FOR AN OPTIMUM INSTALLATION

All these functions being already present in the drive gearbox, it is indisputably profitable to share the hydraulic cooling system between the gearbox and the generator as shown in fig. 9. Then only one heat exchanger is needed. When the generator is conceived with a dry drained case no argument invoking oil pollution can be sustained as during its transit through the generator, oil is not exposed to materials of a kind different from those included in the gearbox such as bearings, rotating seals, gears, or aluminium castings.

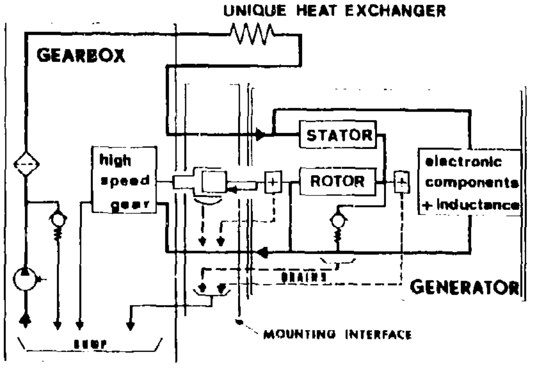


Figure 9 : Full integration of high speed drive and hydraulic hardware in the gearbox

In addition, the inclusion of the high speed gear in the gearbox affords obviously an elegant and logical solution, all the mechanical components being brought together in a sole framework.

This optimum design secures the whole system for significant weight saving, simplicity and improved reliability.

Taking benefit from these installation arrangements the high speed/solid state power system shows minimum weight and size particularly with the integrated package where the power electronic components surrounds the alternator.

Now, such a configuration bears comparison with the conventional systems in terms of power to weight ratio as 1.8 kVA/kg is today accessible with AUXILEC G FC system : this ratio will rapidly improve to 2kVA/kg and more within the next 5 years.

HEAT EXCHANGER DESIGN

The choice for air or fuel heat sink depends mainly on the flight profile of the aircraft. Generally speaking, for supersonic flight condition, fuel is the prime available heat sink : the oil to fuel exchanger is designed taking into consideration the conjunction of a maximum heat rejection with a minimum fuel flow bearing in mind the temperature limits of both fluids. For subsonic flight conditions an oil to air heat exchanger can be used for total heat transfer or as a complement to a basic oil to fuel system when low fuel flow conditions are encountered for instance with very low aircraft speed.

CONCLUSION AND RECOMMENDATION

Very important efforts have been devoted to new 400 Hz generating systems based upon high speed/solid state power conversion. These new systems show already attractive performances. Outstanding improvements will still arise in a near future. Paramount benefits can now be expected from this new technology if a careful attention is paid to an optimum design of the high speed drive, hydraulic and cooling arrangement. In this respect an adaptation of the accessory or engine gearbox is of the first importance.

APPENDIX

The curves $N_{max} = f(P)$, fig. 6, results from the dimensional electromagnetic law of the alternator and from the expression of the critical speed of the rotor shaft in terms of the essential dimensions of the machine, i.e :

$$\text{rotor size} \quad [D^2 . L] \sim \frac{P}{N \times [B.A]} \qquad (1)$$

$$\text{critical speed} \quad N_{crit} \sim \frac{1}{\sqrt{f}} \sim \left(\frac{E I_s}{M_r L_s^3}\right)^{1/2} \qquad (2)$$

with :
- D rotor diameter
- L iron core length
- P electrical power
- N rotational speed
- B air gap flux density
- A armature lineic load
- f static sag of the shaft
- E shaft material modulus of elasticity
- I_s polar inertia of the shaft $\sim D_s^4$ *
- M_r rotor weight $\sim D^2 L$
- L_s effective shaft length
- * D_s effective shaft diameter

B, flux density in the air gap can be considered as a semi constant : its variation with D diameter of the rotor can be approached for medium power machine according to the law :

$$B \sim D^{0.16}$$

On the other hand, A, lineic load varies more significantly with D. For a given copper loss density of the armature (losses to air gap area ratio) depending on cooling technology, the expression of which is $\sigma = \rho A \delta$ (ρ, resistivity, δ, current density) and with a slot depth assumed to be homothetic with D, the following law can be retained for A :

$$A \sim \sigma^{0.5} \times D^{0.5}$$

In other respects the three following coefficients are defined

$$K_N = \frac{\text{maximum allowed speed}}{\text{critical speed}} \frac{N_{max}}{N_{crit}}$$

$$K_D = \frac{\text{shaft diameter}}{\text{rotor diameter}} \frac{D_s}{D}$$

$$K_L = \frac{\text{alternator iron core length}}{\text{effective shaft length}} \frac{L}{L_s}$$

relations (1) and (2) then become

$$P \sim [D^2 L] \times D^{2/3} \sigma^{0.5} N \qquad (3)$$

$$N_{max} \sim K_N N_{crit} \sim \frac{D}{L^2} K_N K_D^2 D_L^{1.5} \qquad (4)$$

It is assumed that the maximum rotor periphered velocity V_{lim} according to the construction technology is reached for a rotational speed equal to the maximum allowed speed N_{max}

$$\text{i.e} \quad V_{lim} \sim D N_{max}$$

Under these assumptions relation (4) becomes

$$\frac{L}{D} \sim \frac{K_N^{0.5}}{V_{lim}^{0.5}} K_D K_L^{0.75}$$

and POWER TO SPEED relation becomes

$$P \sim \frac{V_{lim}^{3.16}}{N_{max}^{2.66}} \quad K_N^{0.5} \quad K_D \quad K_L^{0.75} \quad \sigma^{0.5}$$

The curve $N_{max} = f(P)$ is plotted according to this basic relation for W.R.and P.M.machines assuming V_{lim} = 240 m/sec, P being defined as rated power with provision for 200 % overload.

The weight of the active electromagnetic parts of the machine is

$$M \sim [D^2 L] \sim \frac{D^3}{V_{lim}^{0.5}} \quad K_N^{0.5} \quad K_D \quad K_L^{0.75}$$

and along the $N_{max} = f(P)$ curves the POWER TO WEIGHT RATIO follows the law :

$$\left[\frac{P}{M}\right] \sim V_{lim}^{2/3} \; N \; \sigma^{0.5}$$

The table hereunder gives example of the evolution of these parameters in terms of V_{lim} and N_{max} assuming the other coefficients to be constant.

N_{max}	V_{lim}	$P(N_{max})$	D/L	P/M
constant	+ 25 %	x 2	+ 12 %	+ 16 %
+ 30 %	constant	x 1/2	constant	+ 9 %

DISCUSSION

Ph. Ramette, Fr

Les transisto:.. ne se comportent-ils pas mieux que les thyristors vis à vis des problèmes d'I.E.M.?

Author's Reply

Les transistors de puissance sont des semi-conducteurs rapides (t.on, t.off $< 1\ \mu s$) reblocables par les circuits de commande de base; de ce fait, ils sont mieux adaptés à la modulation de largeur d'impulsion (P.W.M.) que les thyristors (meilleur compromis, masse rendement) et ils procurent une meilleure compatibilité électromagnétique (E.M.C.).

En contrepartie ces composants présentent actuellement des gains trop faibles, qui nécessitent une commande de base plus volumineuse que celle des SCR.

Pour l'instant les calibres en courant des transistors sont inférieurs à quelques 100 A et la limitation en tension est de l'ordre de 1000 V.

On peut dire que les thyristors sont arrivés à "maturité industrielle" alors que les trensistors de puissance sont encore en pleine évolution.

CHAIRMAN'S SUMMARY

by

G.Winterfeld
Chaiman of the Programme Committee

Trying to summarise at the end of this meeting one can make the following statements:

Auxiliary or Secondary Power Systems are very complex systems. Having to fulfil many and different tasks, the constraints under which these systems have to work demand the best possible compromise.

Experiences with existing systems have been carefully examined. They show the various options, from which progress towards better performance and efficiency as well as towards lower life cycle cost can be expected. The main avenues of progress are:

– Advances in small gas turbine technology, mainly turbomachinery design and manufacturing processes, thereby providing lower system weight and lower acquisition cost. An interesting feature seems to be the expendable gasifier by which costs for overhaul can be lowered.

– New concepts for the Secondary Power Systems of military aircraft including the requirements for redundancy are evolving, for example pneumatic instead of mechanical coupling between gear boxes and APU/M.E.

– New approaches for heat rejection from Secondary Power Systems are required. Some solutions have been indicated and must be further studied within a systems concept.

– Systems optimisation by closer cooperation between the designers of the airframe, the main engine and the secondary power system, as well as with the users, can provide more efficient solutions withour compromising engine and airframe performance.

– In the civil transport sector operating cost reductions can be achieved by a careful planning of the APU's usage together with ground power units. However, the general opinion seems to be that on-board-auxiliary power units are necessary in order to guarantee the needed airline operational flexibility.

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14. Abstract

The Conference Proceedings contain 18 papers presented at the AGARD Propulsion and Energetics Panel 61st B Specialists' Meeting on Auxiliary Power Systems, which was held in Copenhagen, Denmark, on 30 and 31 May 1983. Questions and answers of the discussions follow each paper.

The Specialists' Meeting was arranged in 3 sessions: Requirements for Auxiliary Power Systems (4); Provision of Auxiliary Power (10); and Electrical/Hydraulical Systems/Cooling of Equipment (4). The Specialists' Meeting was finalized with some concluding remarks by the Programme Committee Chairman.

For the provision of auxiliary power various technical systems have been developed and new solutions discusssed. It was the purpose of the meeting to review the current state-of-the-art, to exchange experiences and to discuss future problems of auxiliary power generation.

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