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FINAL REPORT—AN ADVANCED CONCEPT SECONDARY POWER SYSTEMS STUDY FOR AN ADVANCED TRANSPORT TECHNOLOGY AIRCRAFT

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ABSTRACT

The application of advanced technology to the design of an integrated secondary power system for future near-sonic long-range transports was investigated. The study showed that the highest payoff is achieved by utilizing secondary power equipment that contributes to minimum cruise drag. This is best accomplished by the use of the dedicated auxiliary power unit concept (inflight APU) as the prime power source for an airplane with a body-mounted engine or by the use of the internal engine generator concept (electrical power extraction from the propulsion engine) for an airplane with a wing-pod-mounted engine.

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SUMMARY

A study program to evaluate the application of advanced technology to the design of integrated secondary power systems (SPS) for future long-range transports has been completed. Goals established by NASA with emphasis on superior performance and improved economic characteristics provided background for the study objectives. These objectives included: (1) establishment of the payoff potential offered by application of advanced technology to future transports; (2) evaluation of the state of readiness of attractive technology areas; and (3) recommended action, including cost and schedule, required to bring the state of readiness to airplane commitment status.

Final results of this study are set forth in the three parts of this document. Key results of the study are presented in the following paragraphs.

The study described in part I has shown that the highest payoff is achieved for near-sonic airplanes (model 767-611) by utilizing secondary power equipment that contributes to minimum cruise drag. Evaluation of all reasonable combinations of power sources, distribution systems, loads, and control methods indicates that a dedicated auxiliary power unit (APU) concept is favored. This concept consists of dual auxiliary power units driving the accessories (air compressor/cooling units, hydraulic pumps, and electric generator). The only accessory on the propulsion engine is an electric starter that is also capable of operating as a generator (generator starter drive).

The internal engine generator (IEG) concept extracts only electric power from the propulsion engine. The electric generator is incorporated internally on the engine rotor shaft. Electric motors drive air compressors, vapor cycle cooling units, and hydraulic pumps. The large penalty of the IEG concept is attributed to the location of three engines in the tail on the baseline airplane configuration. Engine frontal area (drag) for this configuration could be reduced by use of remote gearboxes and by locating the SPS accessories in the fuselage. The comparative results of the SPS configurations selected for critical analysis are included in table 1. Integrated advanced SPS concepts were subjectively examined, and promising SPS configurations for the 1975 to 1985 time period were selected for detailed performance-oriented evaluation.

TABLE 1.--TECHNICAL AND ECONOMIC END RESULT STUDY SUMMARY—MODEL 767-611

OEW = 195 510 lb (88 680 kg) TOGW = 356 000 lb (161 480 kg)

Item	Unit	Secondary power system concept			
		Conventional technology		Dedicated APU III	Internal engine generator IV
		Bleed/shaft I	Shaft II		
Uncycled parameters (delta changes)					
Equipment weight	% of OEW	Base	-0.10	-0.35	-1.61
Cruise SFC	%	Base	+0.42	+2.72	+1.82
Cruise thrust[a]	%	Base	+1.75	+7.89	+2.05
Cruise Drag	%	Base	0	-0.88	-0.62
Cycled TOGW for equivalent payload-range-performance objectives	%	Base	+0.41	+0.52	-1.96
	lb	Base	+1470	+1860	-7000
	(kg)	Base	(+669)	(+845)	(-3180)
Net total value of technology[b]	S/airplane	Base	+42 100	+68 591	-604 423

\+ Denotes payoff
\- Denotes penalty
[a]Reduction as a result of not extracting power from main engines
[b]Assumes equal maintenance costs

The study plan is shown in figure 1. The task involved integration of energy types, user elements, power sources, distribution subsystems, and control functions together with airplane sensitivities to establish an optimum SPS on a total-airplane basis.

The study described in part II has shown that a significant payoff is achieved with any one of the advanced technology SPS configurations evaluated for the model 767-620 airplane (two wing-mounted engines and one tail-mounted engine). The results are provided in table 2.

This effort was initiated late in the program and directed to a better understanding of the IEG payoff for a 1975 engine design go-ahead. The SPS configurations used for comparison exhibit payoff nearly equal to that for the IEG, with moderate technical risk. The generator starter drive must develop the start function and establish confidence through qualification testing. The concept of installing model 747/DC-10 technology accessories in the fan duct bifurcation must be proven by mockup studies. Engine size will be a significant factor in the success of this concept.

The IEG exhibits good potential for application on wing-mounted engines. The concept will require interest and support by the government because of the relatively high development costs.

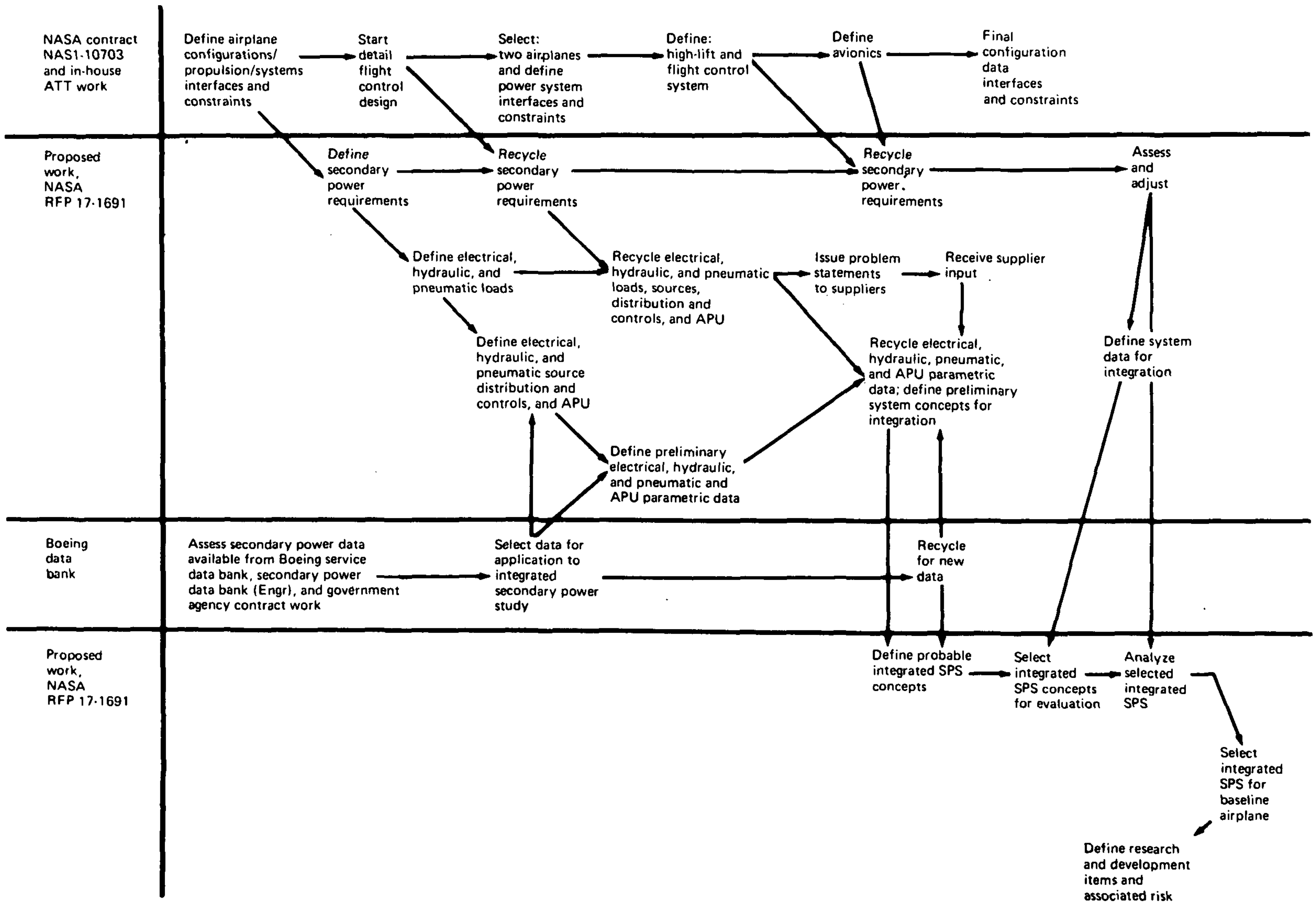

FIGURE 1.—SECONDARY POWER STUDY FLOW PLAN

TABLE 2.—TECHNICAL AND ECONOMIC END RESULT STUDY SUMMARY—MODEL 767-620

OEW = 184 840 lb (83 840 kg) TOGW = 340 600 lb (154 490 kg)

Item	Unit	Secondary power system concept				
		Conventional technology		Engine pad-mounted generator/ starter	Internal engine generator	
		Chin mount installation	Bifurcation mount installation		Cycloconverter	DC link
Uncycled parameters (delta changes)						
Equipment weight	% of OEW	Base	−0.065	−0.144	−0.037	−0.063
Cruise SFC	%	Base	0	−0.02	−0.03	−0.03
Cruise thrust[a]	%	Base	0	−0.04	−0.07	−0.07
Cruise drag	%	Base	+1.7	+1.8	+2.3	+2.3
Cycled TOGW for equivalent payload-range-performance objectives	%	Base	+2	+1.95	+2.77	+2.73
	lb (kg)	Base Base	+6800 (+3100)	+6640 (+3020)	+9450 (+4300)	+9300 (+4230)
Net total value of technology[b]	$/airplane	Base	+236 100	+192 625	+249 210	+281 810
Maintaninability assessement relative to base		Base (100%)	May not be acceptable to airlines at this time	98% as good	90% as good	
Reliability assessment		All concepts are considered adequate for airline use.				

\+ Denotes payoff
\- Denotes penalty

[a]Reduction as a result of not extracting power from main engines

[b]Assumes equal maintenance costs

The cycloconverter IEG is considered a small favorite over the dc-link IEG because of industry development to date. However, either will require substantial development funding to allow commitment to a 1978 engine design go-ahead.

Information is included to identify the relative effect of Mach number on payoffs associated with the SPS. Equivalent SPS gains result in less total payoff as Mach number decreases from 0.98 to 0.90.

The recommended research and development items are presented in part III. All items are worthy of accomplishment; however, anticipated funding limitations indicate a need to prioritize items.

The IEG is a high-priority development item indicating good payoff potential. Risk is high and industry will probably defer development without government support. Vapor cycle cooling, electric motor/hydraulic pump (30 gpm), integrated actuator packages, hydraulic control valve erosion reduction, cabin air recirculation, electric wiring, and others are recommended for funding in that order.

The development of these items will provide the building blocks to an optimized integrated SPS. Inasmuch as each new airplane must be evaluated on an individual basis, it is imperative that substantial building block data be available before preliminary design of the airplane.

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INTRODUCTION

The overall secondary power system (SPS) encompasses those systems generating power for uses other than for primary propulsion, and associated users of this power. These systems are used to manage and control the operational activity of the airplane and to provide a safe and comfortable environment for the crew and passengers. The nomenclature used by the Air Transport Association (ATA) to identify these systems is as follows:

ATA No.	Title
2100	Air Conditioning and Pressurization
2300	Communications
2400	Electrical Power
2700	Flight Controls
2900	Hydraulic Power
3000	Ice and Rain Protection
3200	Landing Gear
3600	Pneumatics
4900	Airborne Auxiliary Power
7500	Engine and Nose Cowl Anti-Icing System
8000	Starting

BACKGROUND

Secondary power systems in past airplane programs were independently developed by individually responsible design groups with heavy dependence on generally available equipment that required a minimum of development. Technological and ecological considerations for tomorrow's airplane will be more complex, will necessitate the use of other than state-of-the-art equipment, and will require more efficient integration to produce a satisfactory solution to the SPS problems. In addition, military airplane programs which had been a prime source of new hardware developments are now more limited, and lead time between commercial concept and hardware has become longer.

United States industry must therefore take firm action now if America is to produce a competitive commercial airplane in the next 10 to 15 years.

OBJECTIVE

The objective of this investigation was to provide a definition, analysis, and identification of significant technical and economic sensitivities and interfaces for an integrated SPS applicable to advanced subsonic commercial transports.

A secondary, but also important, objective was to identify technology trends and potential research and development items that should be conducted to produce a competitive commercial airplane in the next 10 to 15 years.

CONDUCT AND SCOPE OF THE STUDY

This study was totally analytical and was conducted primarily by technology specialists, with the assistance of designers and economic analysts where required.

The study was conducted in three parts. Part I reviewed the 36-airplane matrix in the Advanced Transport Technology Program (ATTP), NASA contract NAS1-10703, to determine the impact of payload-range differences and FAR noise requirements on the SPS. It was concluded that the study results would not be compromised by selecting the model 767-611 airplane as the baseline for this study. SPS functions, loads, distribution and control methods, prime power sources, and configuration limitations were identified for this airplane. Safety regulations, airline constraints, and airline reports on maintenance and reliability of SPS components were reviewed. Airplane sensitivity data generated by the ATTP study were examined. The above inputs were used to conceive general SPS concepts and variations that would satisfy the design and safety requirements, operational constraints, and maintainability objectives. Limited funding available for the study required that heavy reliance be placed on engineering judgment to select four SPS configurations for a further performance-oriented trade study. The results of this study on a total-airplane basis are shown as a takeoff gross weight change and total dollar value of technology.

The four SPS configurations are concepts most likely to mature in the 1975 to 1985 time period. The use of the term "conventional technology" in this study refers to use of 747/DC-10/L1011 technology hardware.

Part II was initiated late in the program as an outgrowth of NASA studies investigating the merits of the internal engine generator (IEG). A limited add-on study was conducted to determine the overall technical and economic payoff for a 1975 configuration; i.e., a 1975 engine program go-ahead resulting in a 1980 airplane certification. Three integrated SPS concepts and two variations that meet the requirements and objectives established in part I and that could reasonably be expected to be available in 1975 were selected for detailed analysis. The baseline airplane configuration was updated from the model 767-611 to the model 767-620, the airplane selected as having the most potential in the ATTP study.

Part III identifies the research and development programs that should be conducted to advance technology in the area of secondary power systems.

SUPPORTING WORK

The work accomplished by The Boeing Company on the Advanced Transport Technology Program was used extensively to support and define payoff on a total-airplane basis. Engine sensitivity data used in this study were similar to those given in the advanced technology engine study work by General Electric and Pratt & Whitney on contracts NAS3-15544 and NAS3-15550, respectively. Other contractor research work used in arriving at the concepts presented in this report was the following: (1) work accomplished by AiResearch Manufacturing Company, "Investigation and Development of New Concepts for Improvement of Aircraft Electrical Power Systems," contract NAS12-659, and (2) work by Pratt & Whitney Aircraft, "Lightweight Small-Frontal-Area Accessory and Drive System Program," contract N00019-68-C-0428.

SIGNIFICANCE OF THE STUDY

This study may be used as a reference to understand secondary power system functions and the processes involved in arriving at the optimum system for a given airplane configuration. The results are significant in evaluating the potential payoff of new concepts and forming the planning base for government support of technology advances needed in the area of secondary power systems.

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SYMBOLS

A	ampere
ac	alternating current
AEEMS	automatic electric energy management system
AFCS	automatic flight control system
AIT	autoignition temperature
APU	auxiliary power unit
ASMC	automatic systems management and control
ATA	Air Transport Association
ATTP	Advanced Transport Technology Program
AWG	American wire gage
BAC	British Aircraft Corporation
BMS	Boeing materials specification
CSD	constant speed drive
CT	current transformer
dB	decibel
dc	direct current
DOC	direct operating cost
DPCT	differential protection current transformer
emf	electromagnetic force
EPU	emergency power unit
Ess	essential
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulations
FCC	flat conductor cable
g	gram
GCU	generator control unit
gpm	gallons per minute
GSD	generator/starter drive

HMG	hydraulic motor generator
hp	horsepower
Hz	Hertz
IDG	integrated drive generator
IEG	internal engine generator
instl	installation
inv	inverter
k	kilo or thousand
kg	kilogram
kgf	kilograms force
km	kilometer
kVA	kilovolt amperes
kW	kilowatt
lb_m	pound mass
m	meter
M	Mach number
mm	millimeter
mmf	magnetomotive force
MTBF	mean time between failures
N	Newton
N_2	high pressure compressor rotor speed, rpm
NASA	National Aeronautics and Space Administration
nmi	nautical mile
OAT	outside air temperature
OEW	operating empty weight
Ovbd	overboard
psia	pounds per square inch absolute
PTU	power transfer unit
QAD	quick attack-detach adapter

rad	radians
R&D	research and development
RCCB	remote circuit breaker with electromagnetic control
rms	root mean square
RPC	remote power controllers
rpm	revolutions per minute
s or sec	second
SAS	stability augmentation system
SCR	silicon controlled rectifier
SFC	specific fuel consumption
SGW	small-gage wire
SPS	secondary power system
SST	supersonic transport
TAS	tentative airworthiness standards
T.O.	takeoff
TOGW	takeoff gross weight
TRU	transformer rectifier unit
VFR	visual flight rules
VSCF	variable speed, constant frequency

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PART I

SECONDARY POWER SYSTEM CONSTRAINTS

The SPS configuration is directly dependent upon the airplane physical configuration, requirements imposed by government and industry, engine design, and the SPS functions and associated loads.

Airplane Configuration

The airplane configuration shown in figure 2 was used to establish the locations of the primary SPS power sources, SPS components, system loads, and length of distribution lines required. The flight profile shown in figure 3 was used in conjunction with the airplane configuration and performance requirements to establish the SPS loads.

The overall airplane design sensitivities of takeoff gross weight (TOGW) to various input parameter changes are shown in figure 4. The input parameters related to variations in the SPS are operating empty weight (OEW), drag, thrust, and specific fuel consumption (SFC).

Requirements

Safety criteria are established by the Federal Government through Federal Aviation Regulations (FAR) and the imposition of special conditions for each new airplane model. For an airplane requiring major changes, the FAR are completely rewritten as Tentative Airworthiness Standards (TAS) (refs. 1 and 2) and incorporated into the FAR at the appropriate time. The regulations applicable to the SPS are FAR Parts 25, 33, and 121. A typical recurring special condition for subsonic transports is to show safe operation of the airplane under visual flight rules (VFR) conditions for a period not less than 5 minutes with the normal electrical power inoperative. For the wide-body transports that are dependent on electrical power for safe flight, the regulation was changed to permit portions of electrical power to remain on with proper isolation and safety precautions. Compliance with these regulations for certification must be shown by documentation of analysis and tests. All of the SPS configurations described in this study meet the requirements of the regulations.

Several paragraphs of the FAR require updating, either for clarification or to make them more representative of jet-engine-powered long-range transports. These are identified below:

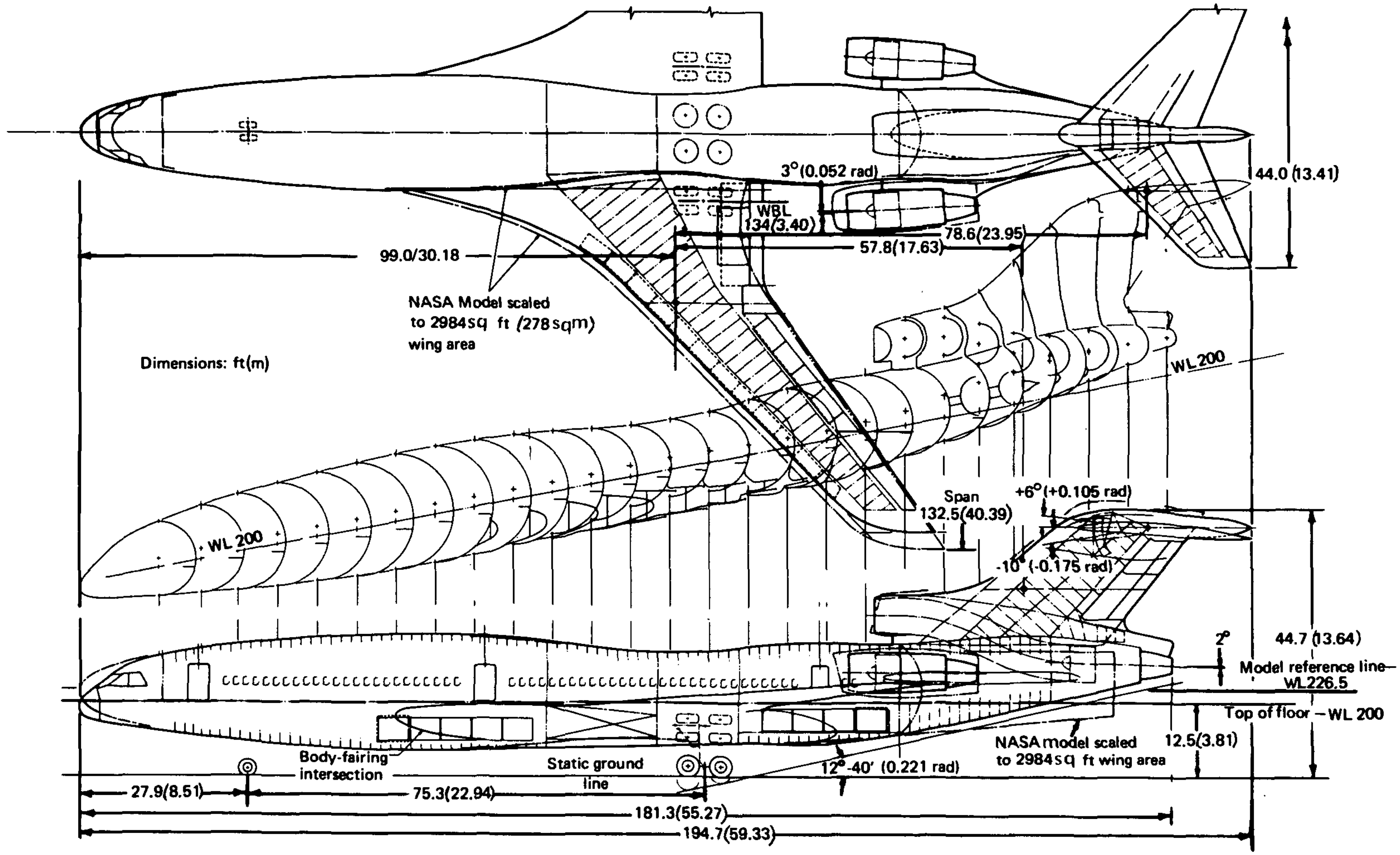


FIGURE 2.—MODEL 767-611

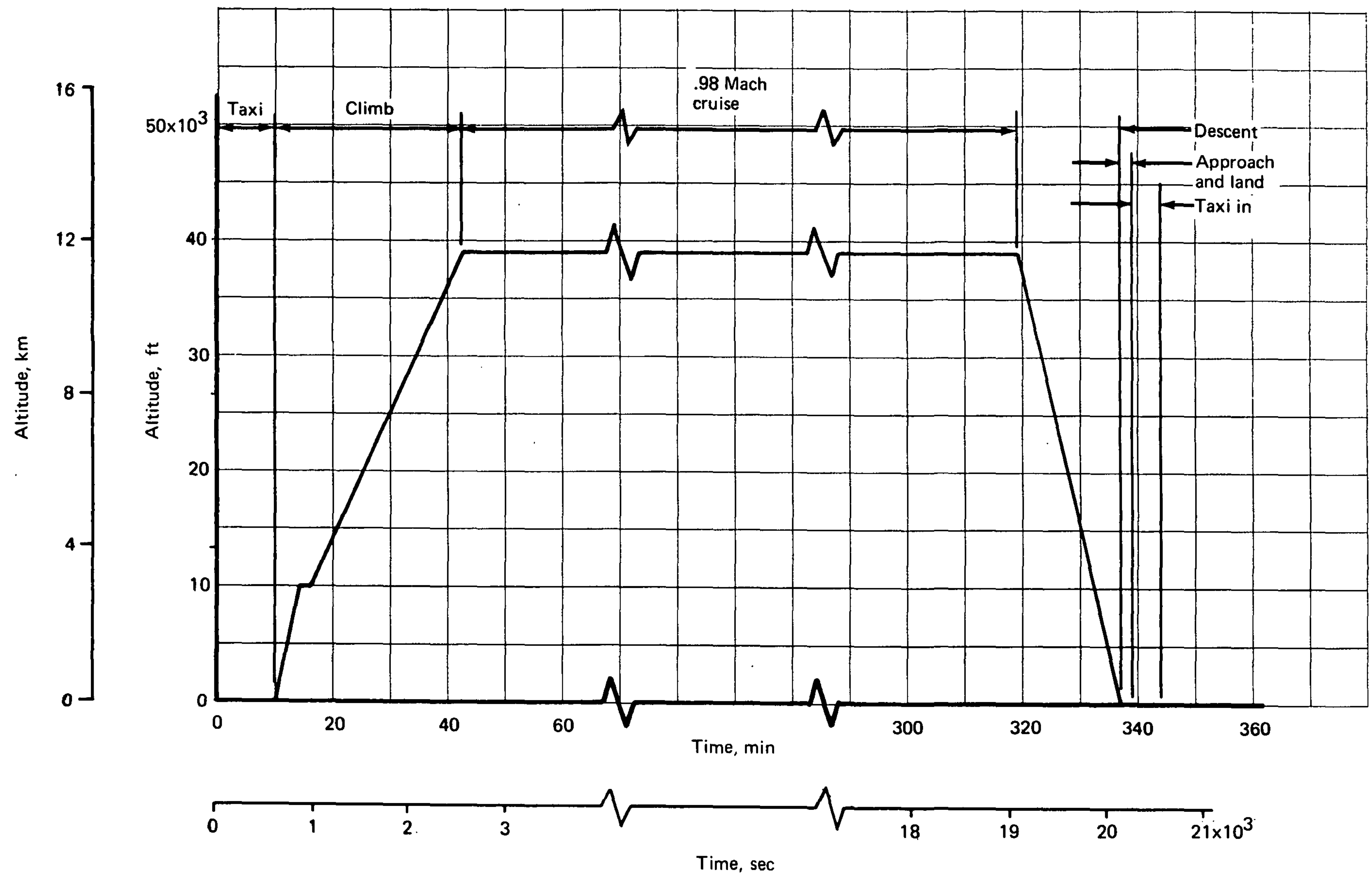


FIGURE 3.—TYPICAL FLIGHT PROFILE

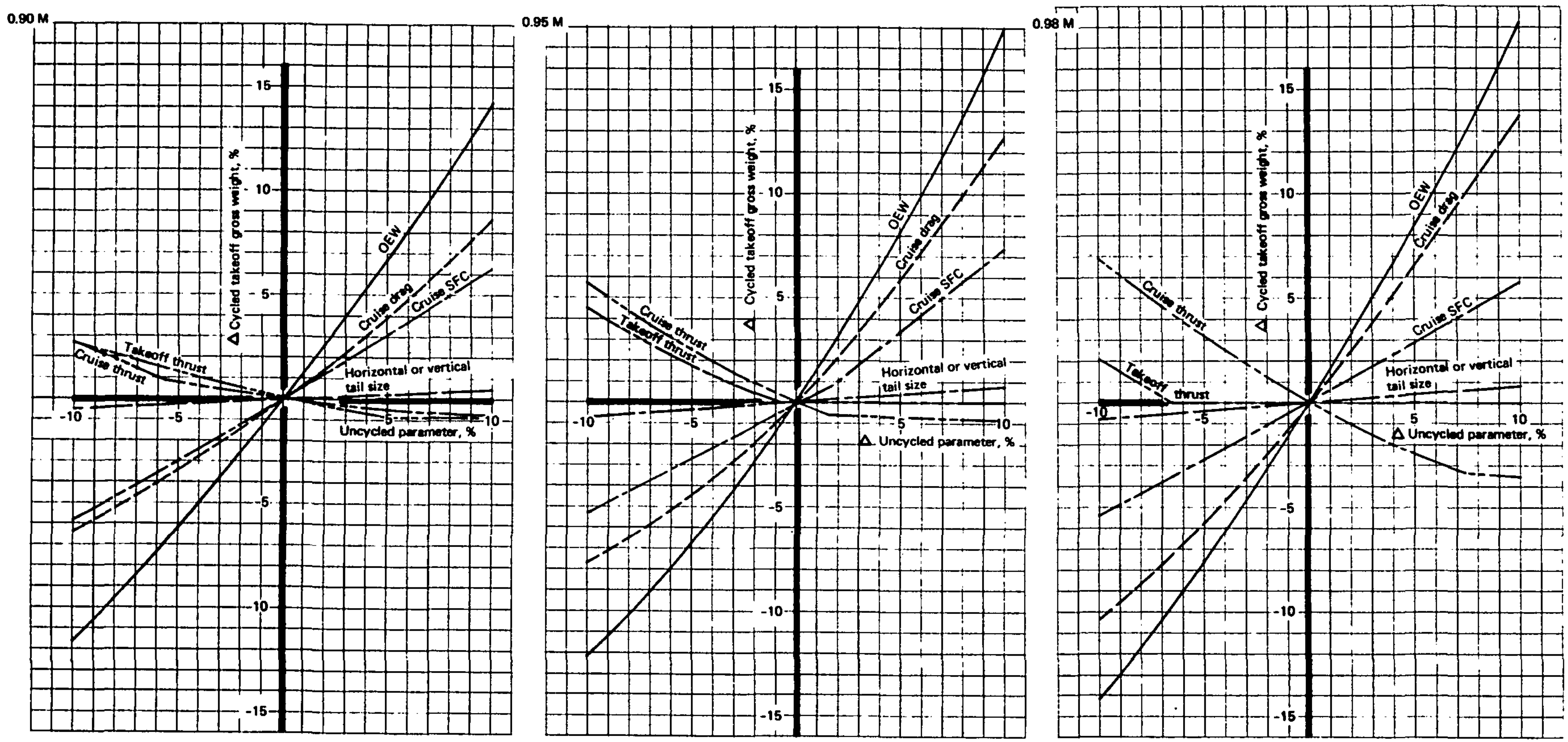


FIGURE 4.—DESIGN SENSITIVITY TO INPUT PARAMETER CHANGES

FAR No.	Title	Remarks
25.831	Ventilation	The wording in TAS (ref. 1) is more descriptive and should be used in FAR.
25.1419	Ice Protection	Update meteorological conditions to reflect additional (available) information.
25.671(d)	Control Systems—General	All engines out should be related to probability number.
25.1435(a)4	Hydraulic Systems	Arbitrary pressure fluctuation requirement does not allow option through design to account for pressure fluctuations.

Airplane manufacturers establish the basic criteria for airplane design, which includes the SPS, to ensure superior technical, economical, and safe performance. The criteria include such things as level of redundancy, system performance, and environment. Airlines assist in establishing the criteria by providing service experience, a joint committee-recommended system operational performance, and specific requirements in the negotiated detail specification for the airplane.

Airplane Engine Constraints

The engine design can place severe constraints on an SPS that relies on the engine for the primary power source. Pertinent factors are compressor stages available for engine bleed air, shaft and bleed power extraction capability, location of power takeoffs, and space for the driven accessories. When engine design go-ahead is authorized, the degree of SPS integration to be incorporated in the engine must already have been established. This generally precedes airplane program go-ahead by 1 year.

Another constraint is the reluctance of engine manufacturers to power critical engine accessories by means other than direct engine shaft power. A typical example is the engine oil pump. Failure of its power source could result in severe damage to the engine. Therefore, the propulsion engine in this study has, as a minimum, an austere gearbox and associated shaft drive provisions to drive the oil pump and any other potentially critical engine accessories.

SPS Functions and Loads Analysis

SPS Functions

The SPS manages and controls the operational activity of the airplane and provides a safe and comfortable environment for the crew and passengers through the functions shown in table 3. Each function normally requires a specific energy form to accomplish the assigned task. The required energy form and its usually related functions are:

- Electric

  Communication, navigation, automatic flight control, instrumentation, lighting.

- Force or torque

  Fuel boost pumps, engine starting, thrust reverser operation, ground coolant air movers, equipment coolant air movers, flight control surfaces, airplane utility systems, cabin pressurization.*

- Heat (supply and/or sink)

  Galley, cabin conditioning, engine and airframe ice protection.

The primary SPS power sources available are shaft and pneumatic power from the main engines, onboard auxiliary power units (APUs), and stored energy such as battery power, accumulator pressure, and monopropellant drives. Extensive studies conducted for both existing and advanced technology airplanes have established that the most advantageous arrangement results from powering some specific functions directly with engine shaft or pneumatic power extraction, other specific functions from an engine-driven electric power generation system, and the remainder from an engine-driven hydraulic power system. The stored-energy system has been found satisfactory for short duration loads only. The preferred energy forms shown in table 3 were selected by the appropriate technical specialists on the basis of these past studies. Energy forms marked "not practicable" have been found to offer no advantages in system trades. Those marked "not applicable" are generally incompatible with the specific function.

*Cabin pressurization must be provided by fresh air in sufficient quantities to provide for passenger comfort and compensate for airplane leakage.

TABLE 3.—SECONDARY POWER SYSTEM FUNCTIONS

Functions	User end preferred energy form					Remarks
	Electrical	Hydraulic	Pneumatic	Shaft[a]	Stored[b]	
Communication electronics	1	NA	NA	NA	3	
Navigation equipment	1	NA	NA	NA	3	
Automatic flight control system						
Automatic guidance and control system	1	2	2	NA	3	
Stability augmentation system	1	2	2	NA	3	
TR units, battery charger	1	NA	NA	NA	NP	
Instrumentation	1	NA	NA	NA	3	
Lighting	1	NA	NA	NA	3	
Galley						
Heating	1	NA	NA	NA	2	
Cooling	1	NA	NA	NA	1	
Fuel boost pumps	1	1	NP	NP	NP	
Main engine starting	2	2	1	2	1	May be electrical, hydraulic, or pneumatic
APU engine starting	NA	NA	NA	NA	1	
Primary thrust reverser	2	1	1	1	2	May be engine fuel pressure
Fan thrust reverser	2	1	1	1	2	
Cabin air source	2	2	1	2	2	
Cabin air conditioning						
Refrigeration cycle	1	2	1	2	NP	
Heating cycle (direct)	1	NA	NA	NA	NP	
Ground air movers	1	NP	1	2	NA	
Equipment cooling air mover	1	NP	1	NA	NA	
Engine inlet ice protection	1	NP	1	NA	NA	
Airframe ice protection	1	NP	1	NA	NA	
Cargo handling	1	NP	NP	NA	NA	
Powered doors and air stairs	1	1	1	NP	3	
Aileron actuation	2	1	NP	NA	NA	Assumes full powered flight with no manual reversion
Spoiler actuation	2	1	NP	NA	NA	
Rudder actuation	2	1	NP	NA	NA	
Stabilizer actuation	2	1	NP	NA	NA	
Elevator actuation	2	1	NP	NA	NA	
Leading edge flap actuation	3	1	1	NA	2	
Trailing edge flap actuation	3	1	1	NA	2	
Landing gear actuation	NP	1	NP	NA	2	Free fall backup
Nose gear steering	2	1	NP	NA	NA	
Wheel brakes	NP	1	NP	NA	3	
Powered aircraft wheels	2	2	2	2	NA	

[a]Direct power from fuel/air combustion process

[b]Power from bottled gas or fluid, chemical process, or mechanical energy

NP Not practicable
NA Not applicable
1 Current state-of-the-art technology normal mode
2 Advanced technology normal mode requires research and development
3 Current state-of-the-art technology emergency and/or backup mode

Categories shown are concept oriented and trade studies of fixed airplane configurations are required to determine the preferred approach.

SPS Loads Analysis

The prime power supply requirements (propulsion engine and APU) are a composite of function loads, power source (generator, hydraulic pumps, etc.) and overall efficiencies. To establish the power supply required for each type of power distribution system, it was necessary to determine which functions would be supplied by each system, the function loads, and the duty cycles. The functions were assigned to the distribution systems on the basis of preferred energy form, as shown in table 3. Where alternate sources could be used, the function was assigned on the basis of judgment by the appropriate technology specialist. In general, it was found that the effect of the alternate on overall study results was minimal.

The systems load profile composite shown in figure 5 was generated for the model 767-611 and was based on duty cycles established for the flight profile. These loads were, in turn, used to establish power system configuration, component sizes, and operating penalties. The power levels during the various flight modes are relatively constant except for high short-duration loads which could represent system sizing design points. A typical critical load for the hydraulic system is landing gear retraction on a go-around. This results in a high power demand at low engine speed and possibly during engine spinup. Pertinent comments for each power distribution system follow.

Electrical.—The loads were categorized and totaled for each of the airplane operating conditions, as shown in figure 6. The categories represent the total load of equipment that must be powered electrically (cannot reasonably be supplied power from another source) and load of equipment that is normally powered electrically (table 3).

Hydraulic systems.—The functions to be powered hydraulically were identified (table 3), and the load profiles shown in figure 7 were generated for the likely load combinations of the various operating conditions for the system diagram shown in figure 8. A conventional hydraulic system configuration using three independent systems operating at 2.07×10^7 N/m^2 (3000 psi) were used for the loads analysis. The load profile represents the peak flow rates that occur during the specific operating condition.

Pneumatics and APU.—A pneumatic system loads analysis was generated. Air conditioning requirements are pneumatic (air supply is required), but the power source may be airplane engine compressor bleed air, APU bleed air, or an air compressor driven by shaft, electrical, or hydraulic power. The nacelle and wing anti-icing systems of present-day airplanes use hot air heating of surfaces to be anti-iced because of the readily available source of hot air and relatively short duration of the loads. The engine starting system on present-day airplanes is pneumatic because of the relatively light weight. The airplane engine starting airflow required is approximately 1.45 kg/sec at 3.1×10^5 N/m^2 (3 lb/sec at 45 psia). The airflow requirements for anti-icing and cabin conditioning are shown

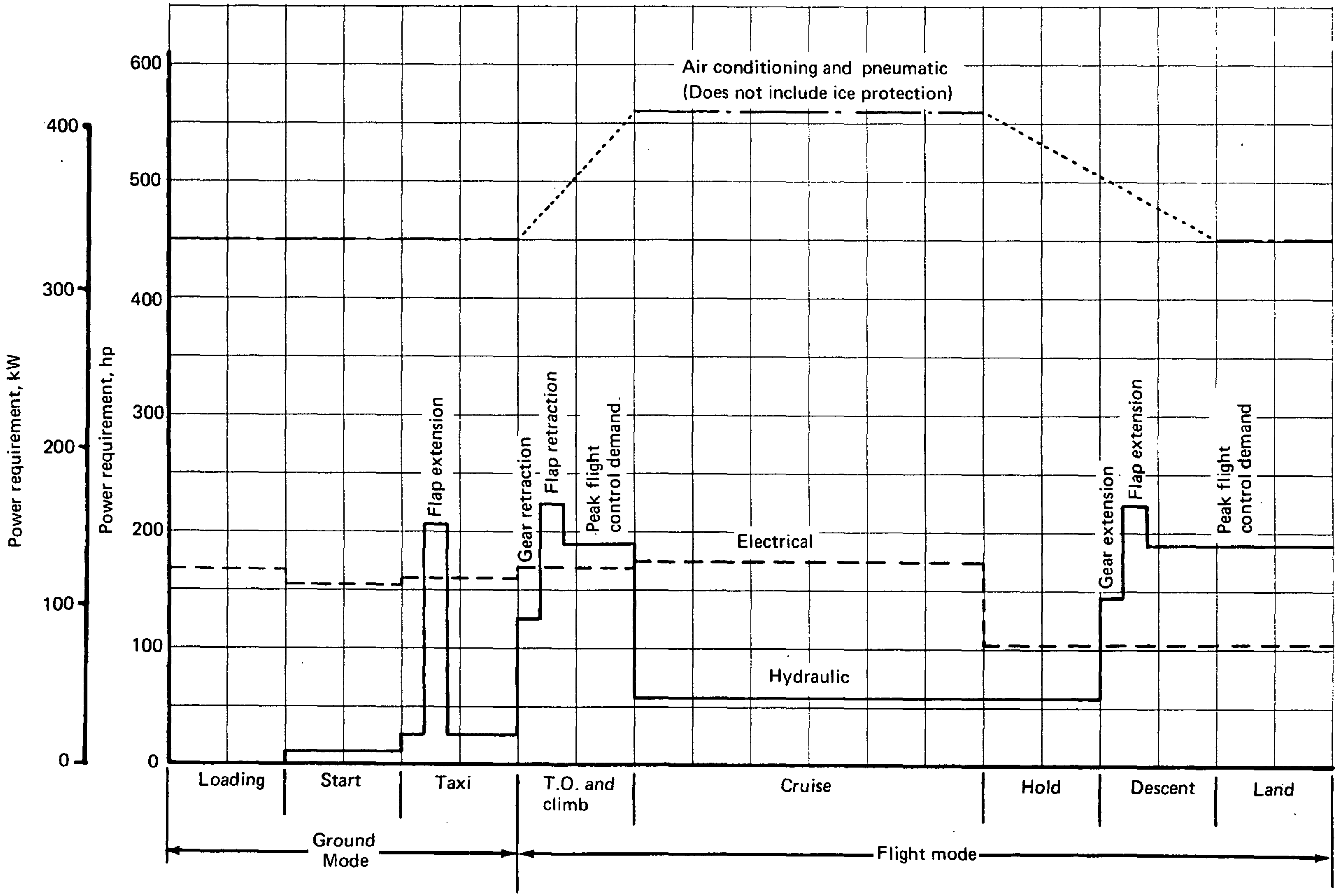


FIGURE 5.—SECONDARY POWER SYSTEM LOAD REQUIREMENTS

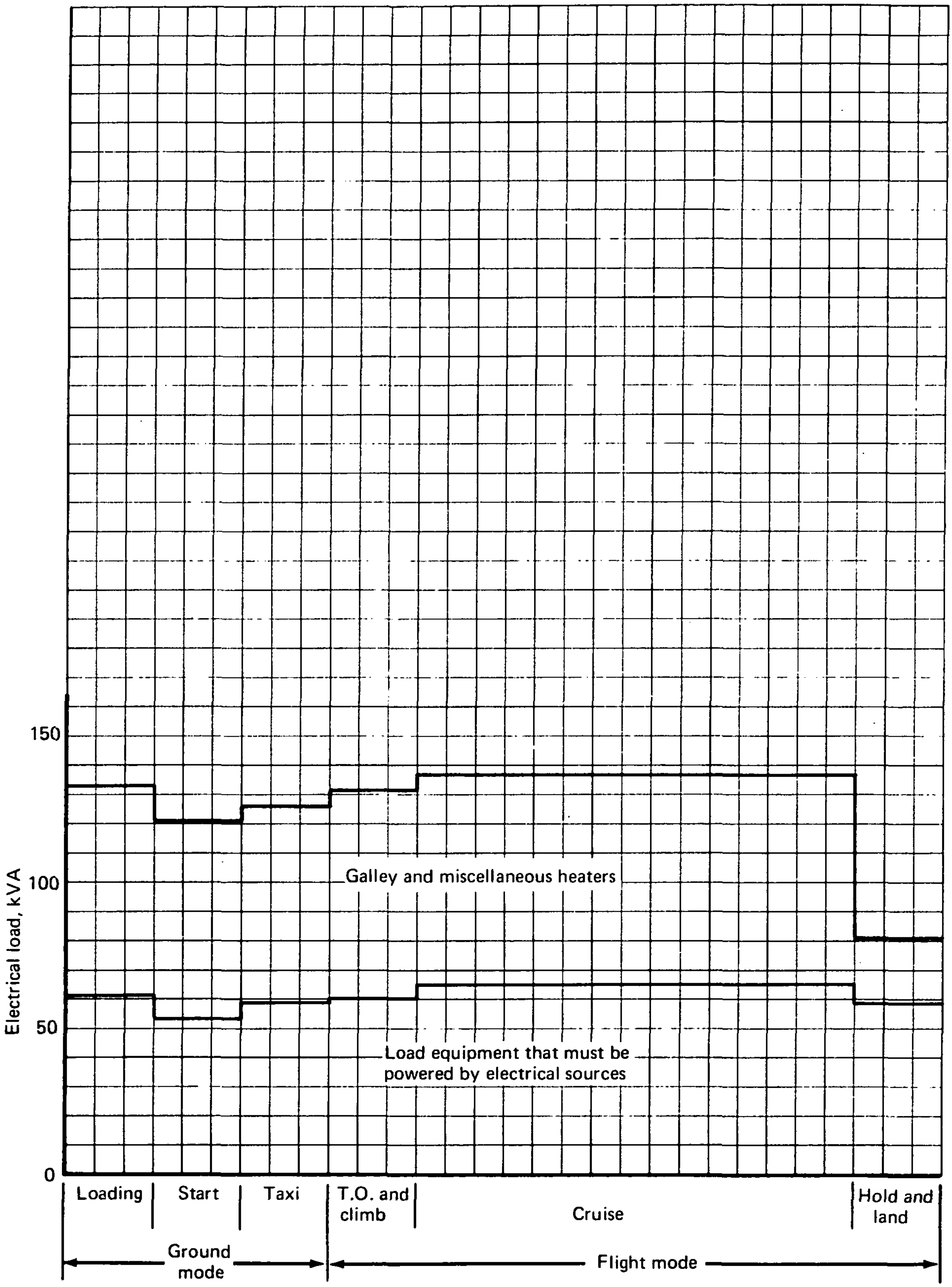


FIGURE 6.–ELECTRICAL LOAD REQUIREMENTS

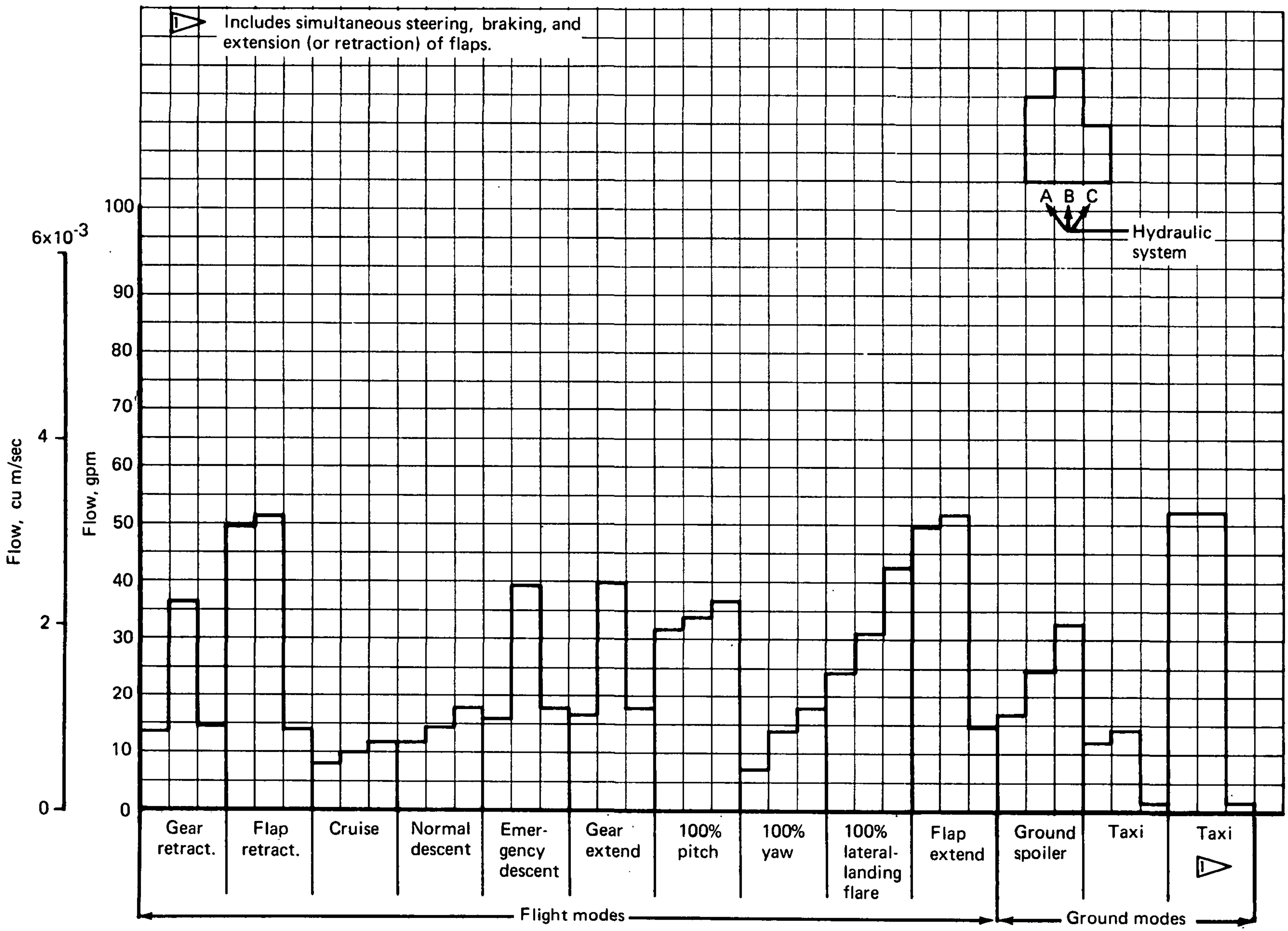


FIGURE 7.–HYDRAULIC SYSTEM FLOW DEMANDS

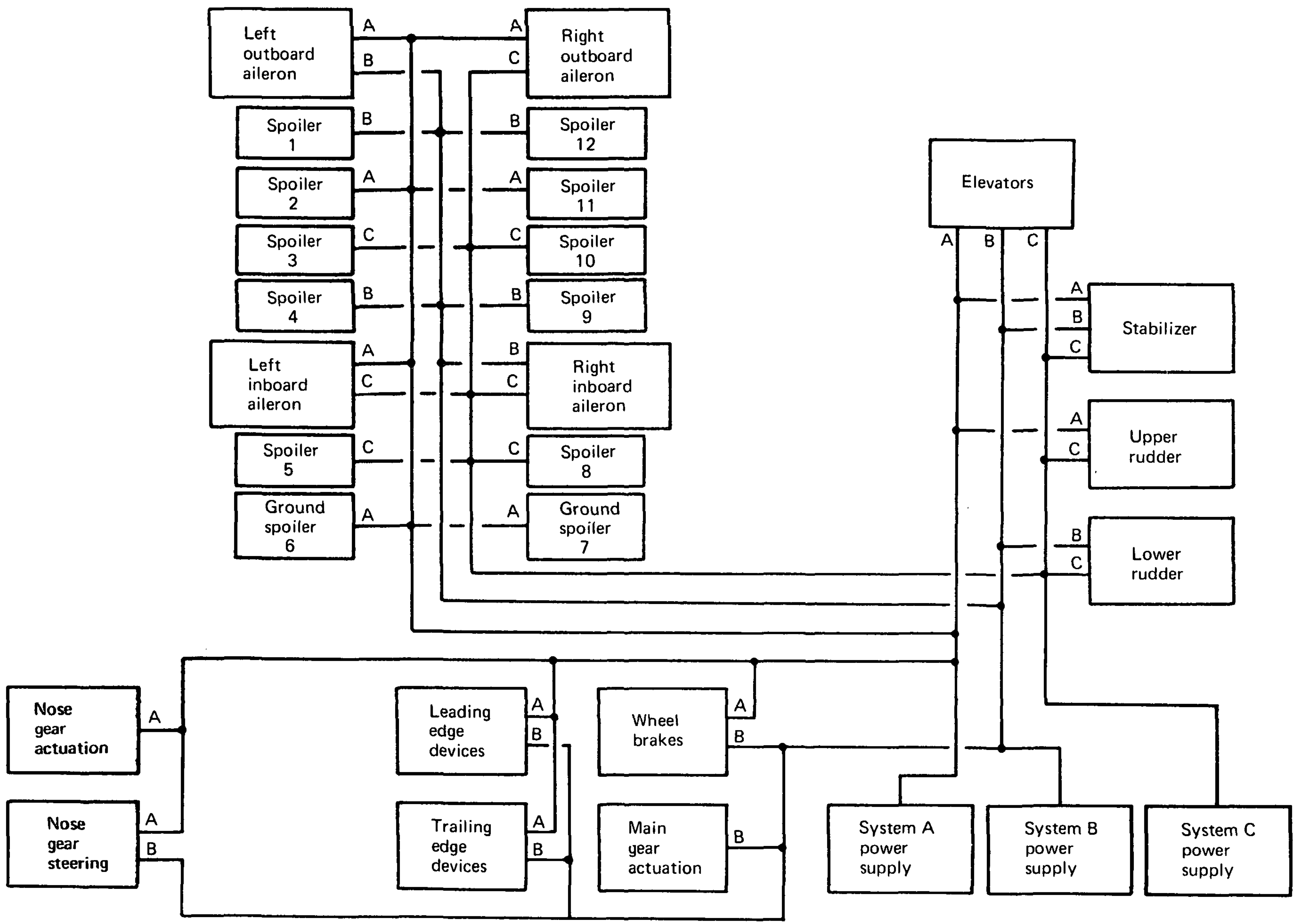


FIGURE 8.—HYDRAULIC SYSTEMS BLOCK DIAGRAM

in figure 9. The pneumatic pressure requirements associated with the airflow requirements are as shown in figure 10. The air conditioning requirement is usually dictated by the cabin pressure, duct losses, and cooling requirements. The range of pressures shown for air conditioning include both vapor cycle and air cycle cooling units. Vapor cycle cooling uses an independent separate cooling system, whereas air cycle cooling depends on the expansion of compressed air in a turbine to accomplish the cooling.

TECHNOLOGY TRENDS

General

The historical trend in design of commercial transport airplanes has been toward higher speeds with associated conveniences offered to the traveler, increased safety of flight, and operational advantages for the airlines. Each step increase in speed and safety has required technological advances to make the increase economically acceptable. This includes advances in airplane aerodynamic configuration, engine design, and airplane system design. This study is concerned with system design, which is heavily dependent on the airplane configuration, particularly on engine design and placement. This interrelationship is discussed in subsequent sections.

Governmental regulations related to operating noise, particulate materials, and noxious gases are becoming more restrictive. One means of reducing noise is to increase engine bypass ratio (ratio of fan airflow to the primary gas generator airflow). The optimum engine bypass ratio for the model 767-611 is approximately 3 to 4. The high bypass ratio results in relatively high engine bleed air extraction penalties (fig. 11), which in turn results in high SPS operating costs. Particulate material emissions must be reduced through primary gas generator redesign, consideration of which is beyond the scope of this study. Noxious gas concentrations may be reduced by both redesign of the engine primary gas generators and reduced main engine ground operation. This study considers means to reduce main engine ground operation for both systems checkout (disconnectable gearbox with continuous-duty starter) and airplane ground movement (powered wheels).

Along with the higher speed, airplanes are being designed with fully powered flight controls, full-time stability augmentation systems, and all-weather landing systems to enhance operational performance. The effects on SPS are to require multiple channels and an adequate degree of isolation between channels for both the hydraulic and electrical systems.

Maintenance techniques are being continually upgraded and have a resulting feedback into the design of new airplanes. These techniques include such items as "on the wing" maintenance of

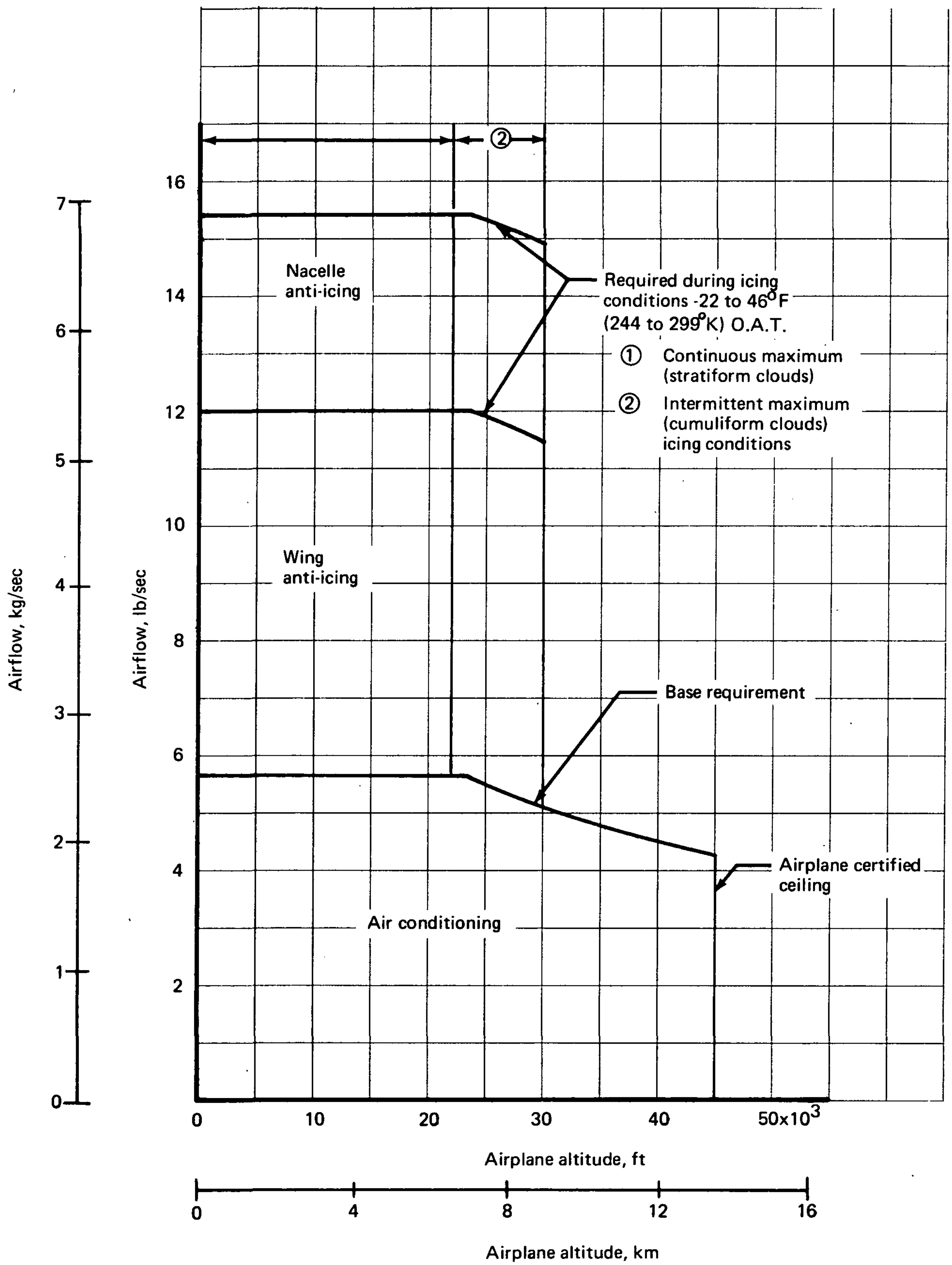


FIGURE 9.—PNEUMATIC AIRFLOW REQUIREMENT

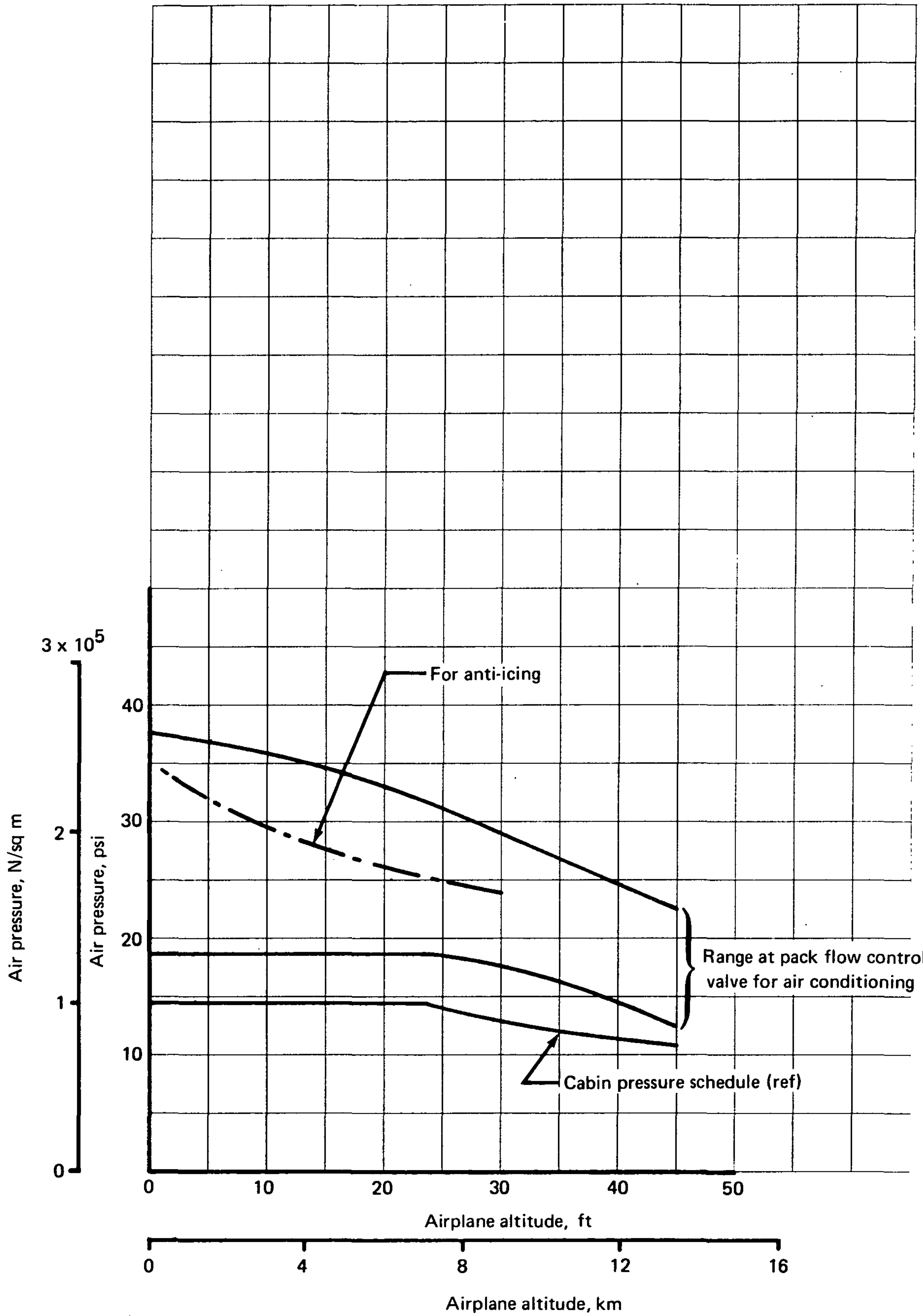


FIGURE 10.–PNEUMATIC PRESSURE REQUIREMENT

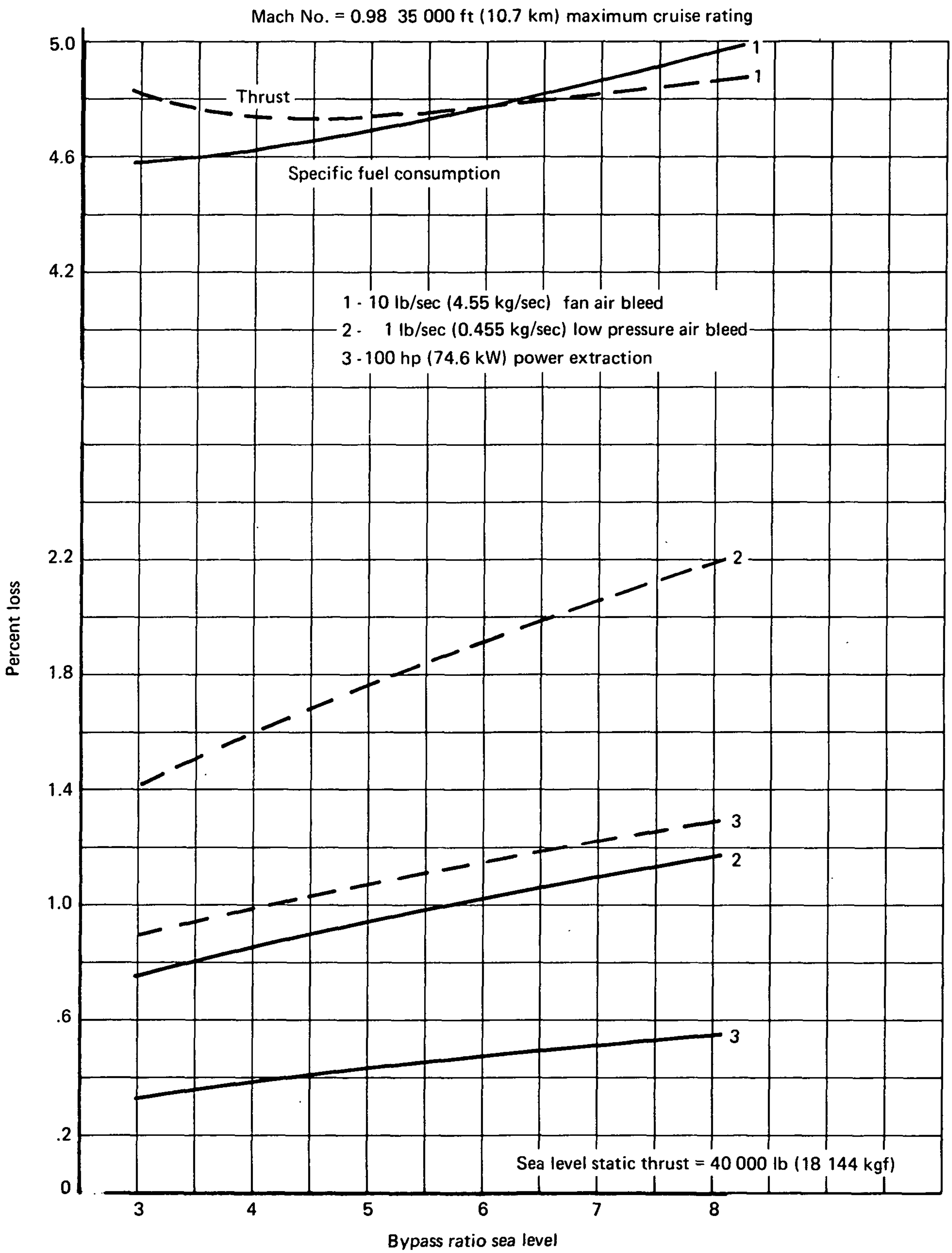


FIGURE 11.–SENSITIVITY CURVES FOR ATSA BPR-2800-24 ENGINE

engines, fault detection systems located on the flight deck, and improved systems designs to allow more complete ground checkout without engine operation. In addition, reliability requirements for components and systems are being continually increased, not only to reduce maintenance costs, but also to meet more stringent safety requirements.

Maintenance considerations include delay or interrupt costs, maintenance frequency, maintenance man-hours, maintenance removals, and indirect maintenance costs. Although each airline approaches the maintenance problem in a different manner, their cost summaries generally fall into the above categories. About 250 items account for 70% of direct maintenance cost. The relative rating of the various SPS systems from typical airline maintenance samples are shown in table 4.

Means of reducing overall SPS costs without degrading performance or safety are continually reviewed. The costs include purchased equipment, installation, and cash direct operating costs (DOC). Purchased equipment and installation costs are reflected in the airplane purchase price. The DOC costs are those absorbed by the airline for in-service operation and include the cost of carrying and maintaining the SPS. Table 5 shows that the SPS weight for the noted family of airplanes is a significant portion of payload (25% to 40%). Any reduction in SPS weight means that either payload can be increased or a smaller airplane designed for the same operational requirements. The estimated weight of the model 767-611 SPS is 5620 kg (12 392 lb), which is 6.3% of OEW or 31% of payload. The predicted cash DOC and the effects of the SPS on DOC for the model 767-611 for the average revenue flight (1853 km (1000 nmi)) are shown in table 6.

TABLE 4.—SECONDARY POWER SYSTEM MAINTENANCE COST RANK

ATA no.	System	Rank[a]	
		System	Subsystem
21	Air conditioning	10	
21-13	Bleed air valves (engine)		45
21-51	Air cycle system		21
21-52	Ram air system		59
36	Bleed air precooler	23	
24	Electric power generator	2	
24-13	CSD and controls		2
29	Hydraulic power plumbing	8	
30	Wing anti-ice and control cabin anti-ice	24	
49	APU—general	5	
49-0	APU		3
78	Thrust reverser system	7	
80	Engine starter	16	

[a]Relative rank in respect to total maintenance costs for systems and subsystems.
Subsystem and system ranks shown are independent of each other.
Lower number represents higher maintenance cost.

TABLE 5.—SECONDARY POWER SYSTEM WEIGHTS

Model	Weight		Percentage of	
	lb	(kg)	SPS/OEW	SPS/payload[a]
737-100	5 626	(2 560)	10.1	32.2
737-200	5 594	(2 540)	9.7	28.7
727-100	8 556	(3 880)	9.8	40.5
727-200	8 695	(3 940)	9.1	32.4
707-121	7 809	(3 550)	6.6	26.4
707-321	7 787	(3 540)	5.9	24.1
747-21P	19 927	(9 040)	5.6	26

[a]Payload based on 15/85 first class/tourist passenger configuration, domestic rules

TABLE 6.—MODEL 767-611 CASH DIRECT OPERATING COST (1971 BOEING FORMULA)

Item	Percent of total cash direct operating cost per trip
Crew	24.6
Fuel	[a]31.5
Insurance	5.9
Airframe maintenance	[a]26.4
Engine maintenance	11.6
	100.0

[a]6.2% of fuel costs and 23.2% of airframe maintenance costs are attributable to secondary power systems, for a total of 8% of trip cash direct operating costs.

Yearly total cash DOC = $2.84 million

The SPS accounts for 8% of the trip cost. This represents an expenditure of $228,000 per year or $3.19 million for the life (14 years) of the airplane. The present value of this amount, based on 15% rate of return, is $1.3 million. If a particular advancement in technology reduces the DOC of the SPS by 25%, an expenditure of up to $0.325 million *per airplane* ($97.5 million for a 300-airplane program) would be justified to develop that advancement.

Airplane drag must be minimized to provide the most economical cruise operating characteristics. Therefore, means are continually being explored to reduce weight, size, power consumption, maintainability, surface wetted area, external aerodynamic interference, ram air usage, and penalty for overboard discharge air outlets. Significant items related to various systems are noted below.

Systems

Electrical

Isolated operation.—Systems designed for parallel generator operation are normally lighter than isolated systems, but when parallel, a single fault may cause protective trips of more than one power source or an overload can temporarily degrade power to all airplane loads. Therefore, the need for redundancy of power sources is not met by the conventional parallel electrical system, and the trend is toward isolation of electrical sources.

Load center location.—The trend is to have centrally located electric load centers. Because FAR safety regulations require that circuit breakers to many of the electrical loads be resettable in flight, the primary electrical distribution center (main power shield) and subbuses have in the past been located in or near the flight deck, incurring a significant weight penalty. The use of remotely controlled circuit breakers permits locating the main power shields (load centers) closer to the generators and to the larger electrical loads, making possible very large reductions of generator feeder weight and power distribution wiring. Wiring congestion, weight, complexity, and installation costs would also be reduced. However, application of remote control protective devices to airplanes is just beginning. A limited number of thermal circuit breakers which can be electromagnetically tripped or closed by remote control are used in recently certified wide-body commercial airplanes. Continued development of remote circuit breakers to make them smaller, lighter, and more reliable is required to realize the total benefits of strategically located electric load centers.

High-voltage systems.—The integrated SPS configurations utilized 115/200-V electrical systems. Use of higher voltages could have a significant effect on distribution system weight and bulk. Doubling system voltage theoretically permits a reduction in conductor weight of 75%. A study conducted for the 747 airplane showed that a 230/400-V system would save 319 kg (704 lb) over the 115/200-V system. To realize the maximum benefits from increased system voltage, compatible load equipment needs to be developed. For example, the above-mentioned weight savings would be greatly lowered if transformers had to be used to accommodate current technology equipment. Advancements in wiring technology and the increasing use of aluminum wire also reduce the total benefits to be realized by some percentage points. Also, significant reduction in wire length and weight resulting from the use of load centers away from the crew area will further decrease the potential weight reduction. A study evaluating the use of double voltage on the model 767-611 with high-voltage-compatible load equipment showed maximum potential weight savings of 69 kg (152 lb). The disparity between the 767-611 and 747 airplanes shows that the total benefits to be realized are heavily dependent on airplane configuration and size and on electrical system configuration.

High frequency.—Practical use of high frequencies is seriously questioned, due to the large quantity of available user equipment, including associated ground support equipment and test equipment based on 400-Hz power. The reduction of motor, transformer, and generator weight calculated for a 1600-Hz system on the baseline airplane was 140 kg (310 lb). Without an increase in system voltage, this weight savings would be more than offset by large increases in generator and load feeder weight. Significant added costs would be associated with a change to high frequencies. The major costs are attributable to lack of commonality with other airplanes in a given fleet and lack of compatibility with existing ground support equipment.

Advanced wiring technology.—The trend in wiring technology is toward expanded use of small gages and flat conductor configurations. High-power feeders will extend the use of aluminum conductors and adapt flat wire configurations for improved thermal characteristics and reduced overall impedance. Matrix interconnection centers will be more widely used to provide flexibility for circuit changes. Use of the integrated wiring concepts will result in reduced installation and maintenance costs as well as lower weight.

Power generation.—The trends are to use oil-cooled generators, integrate the constant speed drive (CSD) and generator into one component (IDG), and provide the propulsion engine starting function. These are described in the following section.

Hydraulics

Powered flight controls.—Most of the flight control actuation functions on modern commercial transports are handled hydraulically to reduce pilot workload by means of fully powered flight control surfaces. Until the latest generation of commercial airplanes, the pilot could still manually move the control surfaces (manual reversion) in the event of a hydraulic power failure. For the latest generation airplanes and for the advanced technology transports, however, the flight controls are fully powered. The power level required precludes use of manual control. Elimination of manual reversion requires added redundancy built into the control surface actuation systems, increasing the demand for hydraulic power. In some cases, sufficient power for flight control can be extracted from the engine-driven hydraulic pumps with engines inoperative but windmilling from ram air in the unlikely event of all engines failing. This may not be the case for very high bypass ratio engines; if so, other sources such as ram air turbine or emergency power unit (EPU) may be required.

High-speed pumps.—Pumps with operating speeds up to 596 rad/sec (5700 rpm) are currently used in commercial transport airplanes. However, these pumps are low-capacity units and not typical of the main engine-driven pumps in the hydraulic systems. Each pump size has its typical rated design speed, which decreases as pump size (displacement) increases. Pump weight and envelope also increase as displacement increases. A typical pump displacement/speed relationship for axial piston pumps is shown in figure 12. Pump delivery can be increased by increasing pump operating speed rather than by using larger displacement pumps, thereby realizing both a weight and envelope reduction. The smaller envelope offers potential for a reduction in engine frontal area, thus reducing airplane drag for an engine accessory gearbox installation. Experience to date has shown that higher speed, large-displacement pumps are within existing technology capability. However, increasing the pump operating speed for a given pump beyond the rated speed results in reduced pump life unless the pump design is changed and improved materials are used in its construction. Typical pump life as a function of operating speed is shown in figure 13. It is noted that increased speed for a given pump dictates higher pump inlet pressurization to prevent cavitation. Potential savings in pump weight may, therefore, easily be cancelled by required changes in the reservoir pressurization system.

Decreased pump life is generally not acceptable to airplane operators. Therefore, further extensive development is required before benefits of higher pump speeds can be realized.

Distribution system.—Means to increase reliability and reduce weight are continually being investigated because of the extensive plumbing required for the hydraulic distribution system of large airplanes. Reliability advancements include increased use of permanent tubing joints, improved

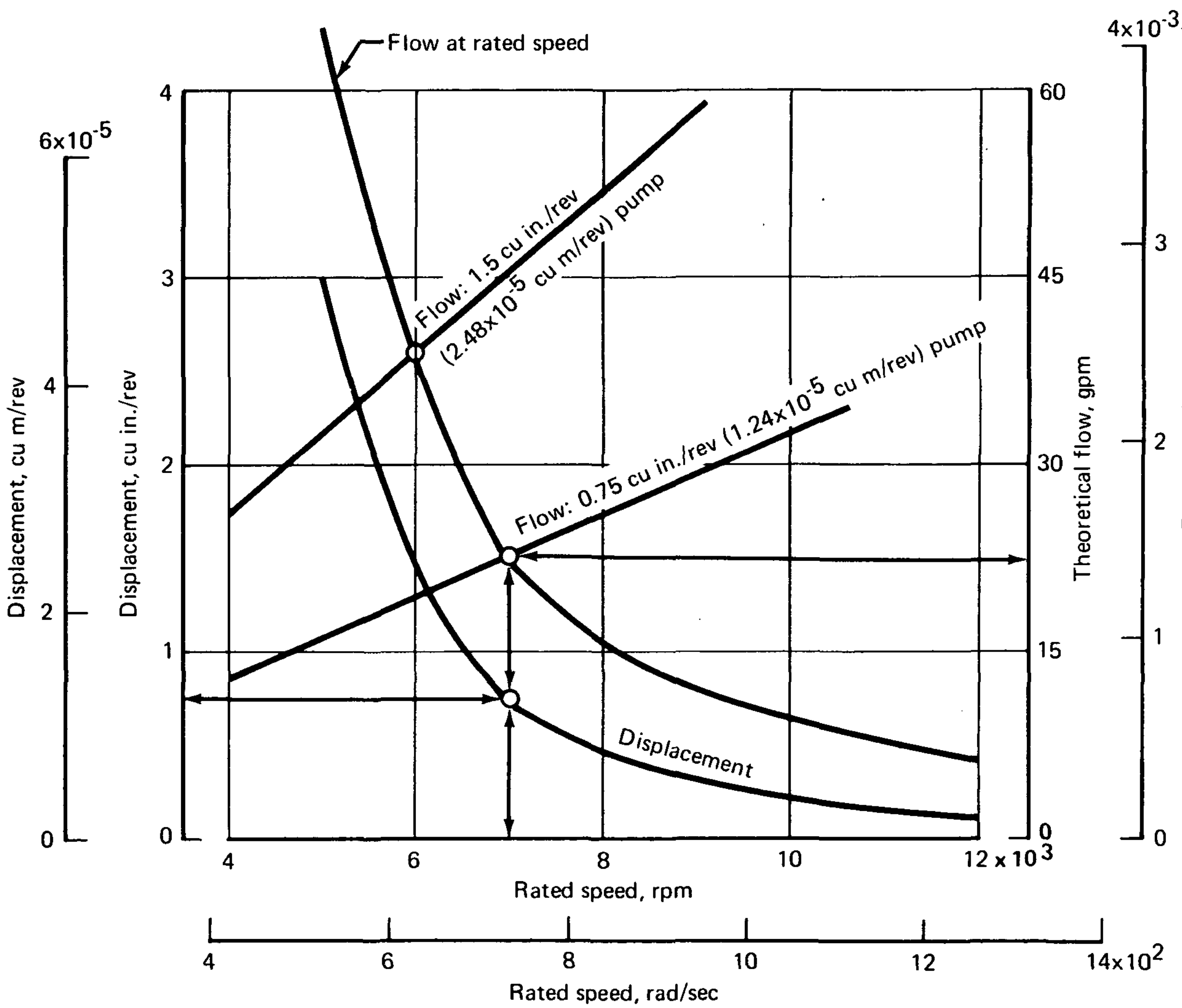


FIGURE 12.—TYPICAL AXIAL PISTON PUMP SPEED, DISPLACEMENT, AND FLOW COMPARISON

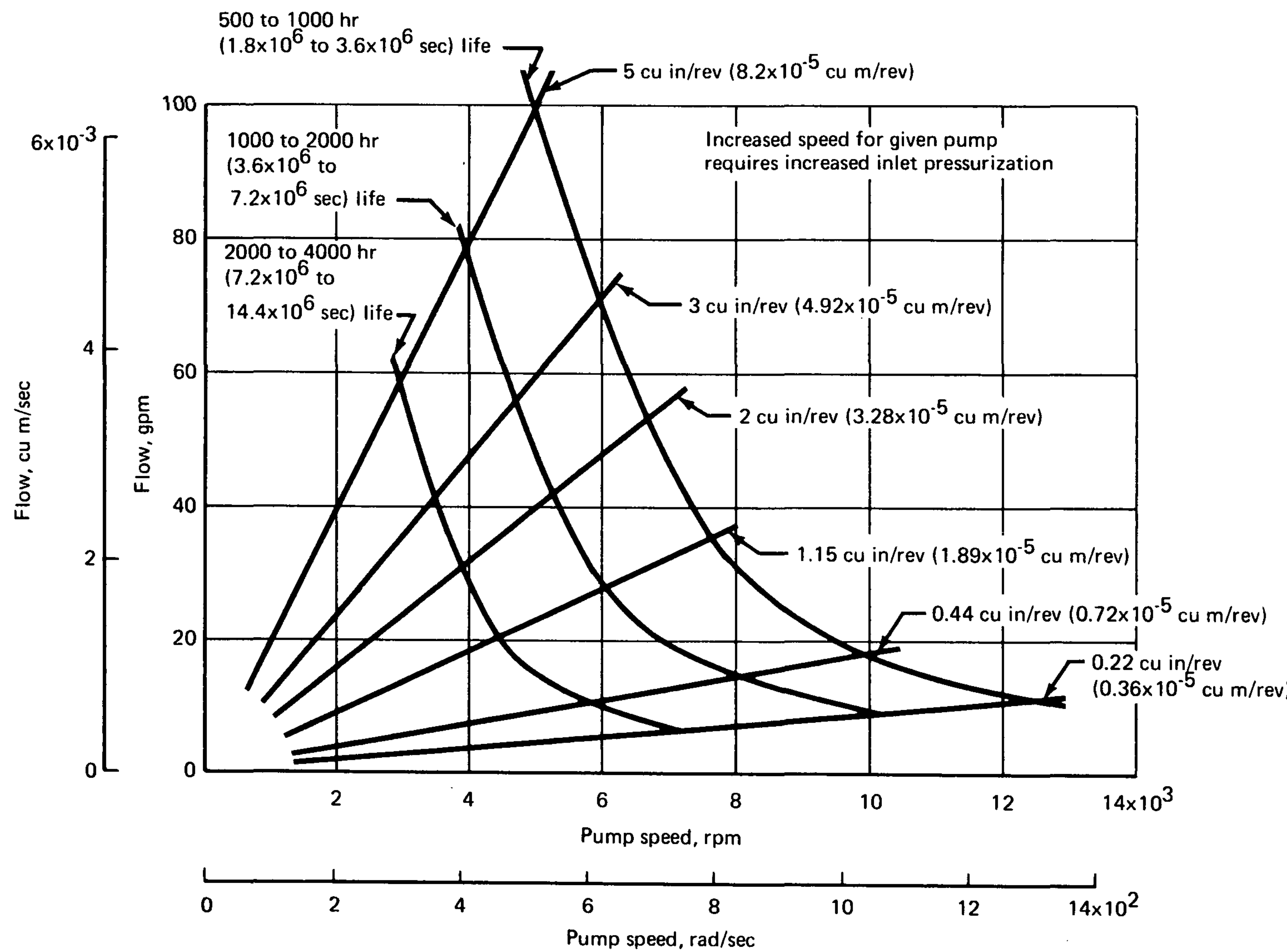


FIGURE 13.—PUMP SPEED EFFECT ON PUMP LIFE

reconnectable joints, improved clamping, and the increased use of coiled tubing in place of hoses (ref. 3). Weight reduction is being achieved through the use of high-strength, lightweight titanium tubing. A typical weight comparison between titanium and stainless steel pressure line tubing is shown in figure 14.

System pressures.—The system pressure level selected for the model 767-611 is 2.07×10^7 N/m^2 (3000 psi). This pressure level was selected on the basis of developed component reliability. Studies indicate that increasing the system pressure from 2.07×10^7 to 2.76×10^7 N/m^2 (3000 to 4000 psi) could result in a weight reduction of up to 68 kg (150 lb) for the advanced technology airplanes. Increased system pressure allows the use of reduced actuator piston area, resulting in a reduction of actuator dynamic stiffness due to the reduced oil spring rate. This must be carefully analyzed to establish the effect on flight control performance. The optimum system pressure for minimum system weight is configuration dependent. Higher pressures have been used to some extent in military and experimental airplanes and on some commercial transports for utility functions. However, until seals, equipment, and distribution systems components can be developed to a level of high reliability, higher pressure levels will not be acceptable to the airlines.

Flight control and landing gear actuation.—The operating requirements and resulting high loads generally result in the selection of hydraulic cylinder actuators for gear retraction and flight control surface movement. The hydraulically actuated systems invariably show the lightest weight and highest reliability for the same degree of stiffness and actuation time when compared with other systems.

Integrated actuator packages.—Integrated actuator packages (self-contained electric/hydraulic units) have been used on the British Aircraft Corporation model VC-10 and for some military applications. The primary advantages of currently available integrated actuator packages are decreased airplane vulnerability and increased airplane survivability (refs. 4 through 7). Subsequent development may show a weight advantage.

Powered wheels.—Self-contained wheel drives, powered from an onboard power source (APU) other than main propulsion engines, have the potential of improving airplane operational flexibility, reducing noise and air pollution, and reducing the exposure to jet blast from main engines during taxi and parking operations. There is also a potential for operating cost reductions resulting from reduced fuel consumption and the elimination of the need for towing equipment and related ground personnel, as well as a potential reduction in delay time. These savings will be somewhat reduced by capital investment costs, increased maintenance cost, and increased airplane weight. Powered wheels are currently in the investigative and concept stage (ref. 8). The power supplied to the drives could be hydraulic, electric, or pneumatic. On the basis of studies to date, hydraulic power is preferred.

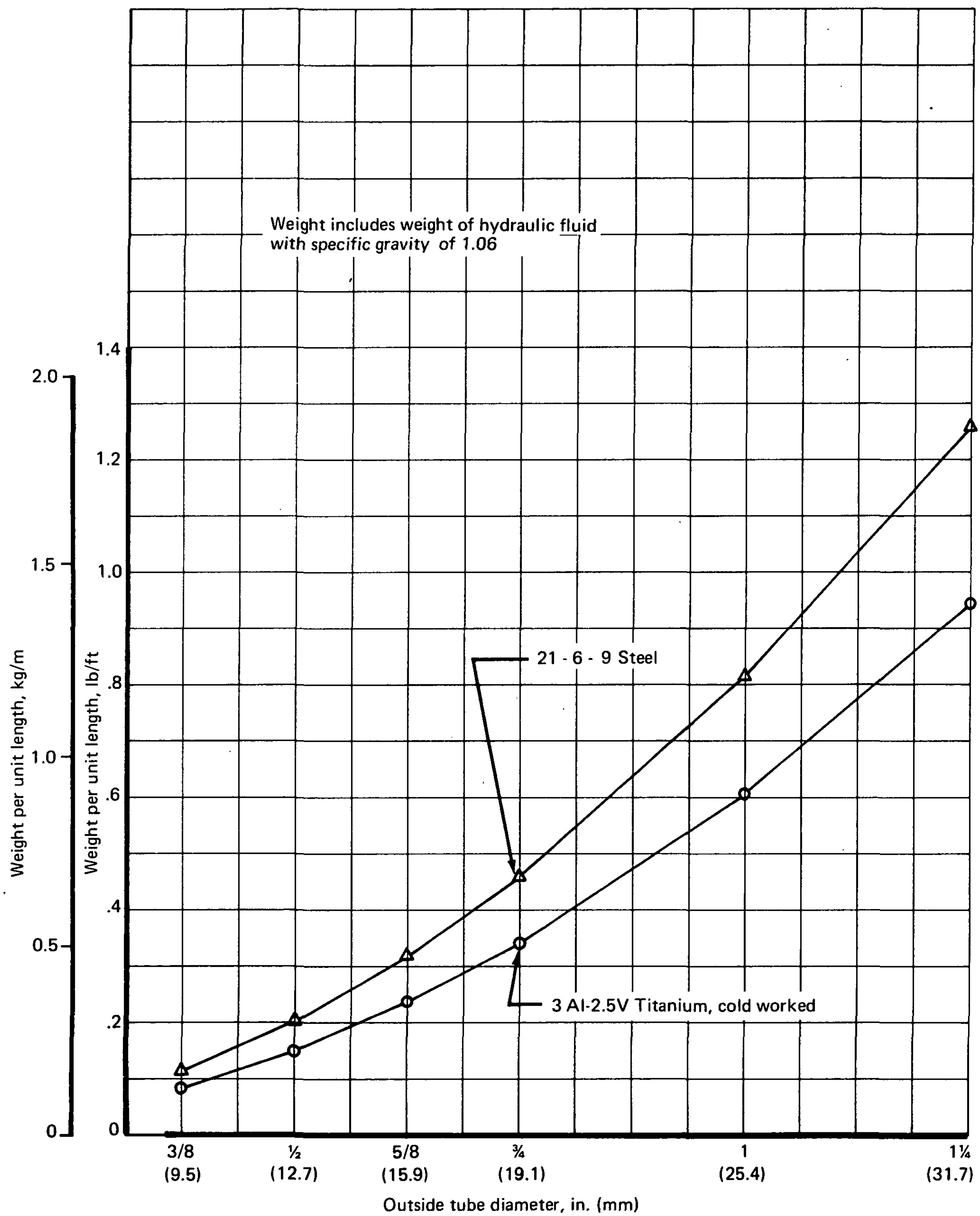


FIGURE 14.—TITANIUM AND STEEL PRESSURE LINE TUBING WEIGHT COMPARISON

Pneumatics

The general trend in pneumatic systems has been to provide increased airflow capacity to meet increasing demands. These demands result from increased requirements in the areas of passenger comfort, dispatch capability, pneumatic motor usage, and increased engine inlet anti-icing.

Source.—The primary source of pneumatic power on the commercial jet transports is two stages of propulsion engine compressor bleed. The lower stages have less penalty; however, the pressure is insufficient for high-altitude and low-power descent, so a higher stage with greater penalty is required for some operations. Switching or mixing provisions allow use of the appropriate compressor stage. The use of engine bleed air as the pneumatic power source is expensive from an overall operating penalty consideration because the airflow normally is not extracted at the most economical pressure and temperature level. Also, as engine bypass ratio is increased to meet noise requirements, the relative cost of bleed air for a given engine size increases (fig. 11). This results from a higher percentage of the primary gas generator air being extracted. The minimum pressure level required from the air source must be sufficient to meet cabin pressurization requirements, duct system losses, and cooling unit operation. As engine bleed extraction levels are normally not optimum for all operating conditions, consideration has been given to the use of engine fan or ram air compressed by a separate compressor driven by the engine shaft. Studies have shown that there can be a significant operating penalty reduction on long-range airplanes, provided that the compressor is designed for maximum efficiency by the use of variable speed or geometry. This saving is partially offset by increased maintenance and equipment costs. Another alternative is to extract engine bleed air at the optimum pressure level available in the engine. This would require the ability to select any one of three or more stages of engine bleed air over the total flight profile. Maximum efficiency could be realized through the use of mixing ejectors properly scheduled for stage switching. This alternative requires definition before initiating engine design to ensure compatibility with the engine configuration and performance.

Wing anti-icing.—Industry has continued to explore problems related to wing anti-icing, to increase the understanding of how ice shapes build up and their effect on airplane handling characteristics. As a result of the better understanding of icing problems, the degree of ice protection (on turbine-powered, swept-wing airplanes) needed for safe flight has continually been decreasing. In addition, the meteorological conditions specified in FAR were established in the late 1950s based on data available at that time. Since then, a significant quantity of meteorological data has become available. The airplane manufacturers have been reviewing the data to determine whether the FAR criteria should be updated.

Engine starting.—The normal method of providing engine start torque is through an air turbine motor that is used strictly for starting engines. The recent trend has been to attempt to integrate the starting function capability with other functions such as the generator/starter, jet fuel starter/APU, and the hydraulic pump/starter motor. The objective is to delete one rotating component with a resultant savings in weight and reduced frontal area drag.

Cooling units.—The increased requirements in passenger comfort and the airline requirement to be able to dispatch with one cooling unit inoperative have led to the use of oversized cooling units. These units are generally oversized to the extent that passenger comfort can be maintained with one unit inoperative.

Pneumatic actuators or motors.—Pneumatic motors are being used to a greater extent for some short-duration, high-power loads in locations where ducts exist for other uses. This generally results in potentially lighter system weight and higher operational reliability. However, the effect of the use of bleed air for pneumatic actuators or motors on overall pneumatic system sizing is to increase the engine bleed air precooler and ducting sizes. This can be overlooked, with the result that no real weight savings may exist. Noise generation during ground operation must be minimized.

Engine inlet anti-icing.—Noise regulations are tending to cause the addition of noise attenuating devices in the engine inlets. These devices, frequently in the form of concentric rings, require anti-icing and will increase the engine bleed air extraction requirements unless other means are employed to limit the ice buildup.

TRADE STUDIES

General

The SPS has a significant effect on the overall cost of airplane ownership and must be integrated on a total-airplane basis. This was accomplished by screening the SPS constraints, functions, loads analysis, technology trends, and interfaces into a workable SPS concept. The constraints, functions, loads analysis, and technology trends have been identified in the sections "SPS Constraints" and "Technology Trends."

Typical prime power source/system interfaces for the propulsion engine and the APU are shown in figures 15 and 16. Maintainability and reliability experience with existing hardware and objectives for the future were also reviewed.

Potential benefits to be gained by considering advanced technology items for individual systems are identified in part I, "Technology Trends," and in part III. These items (e.g., advanced wiring, high-pressure hydraulic system, cabin air recirculation) should be considered in any trade

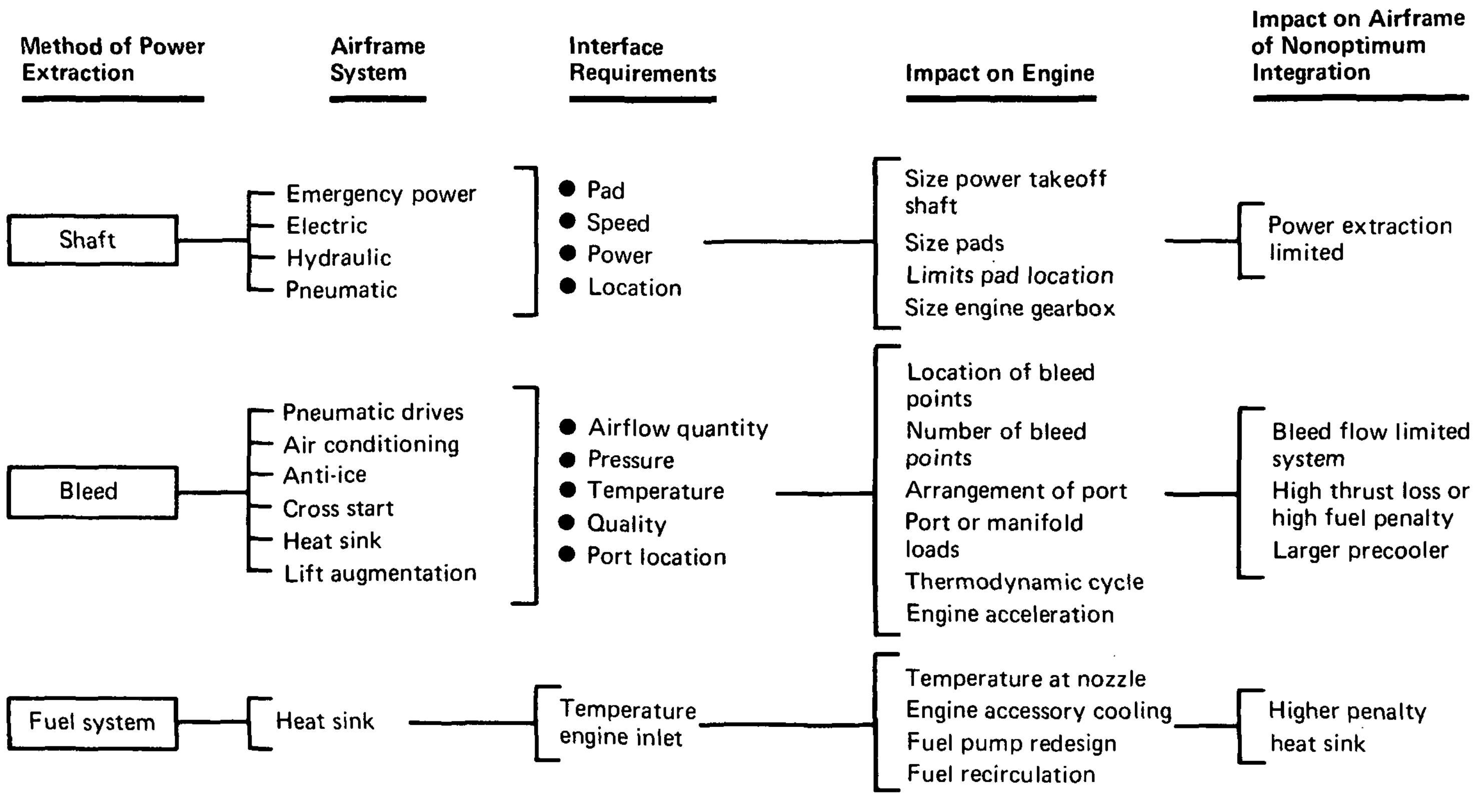


FIGURE 15.—ENGINE/SYSTEM INTERFACE

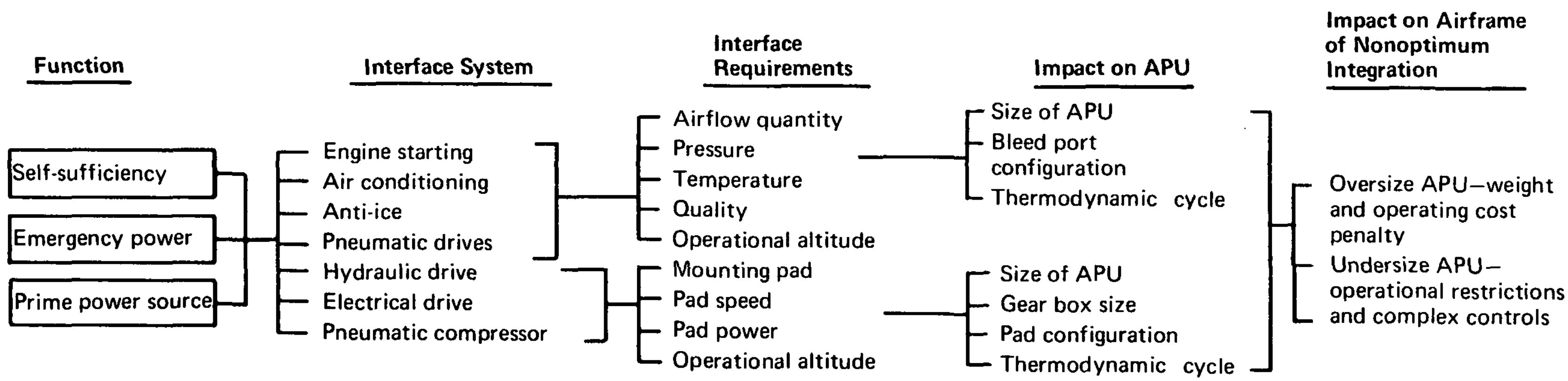


FIGURE 16.—APU/SYSTEM INTERFACE

study but require considerable research and development to formulate into workable integrated SPS concepts. This unfortunately has not been accomplished to date. An evaluation of airplane sensitivity data, however, indicated that the best payoff on a total-airplane basis for a near-sonic airplane would be achieved by reducing cruise SFC and cruise drag. It was therefore decided to focus the attention on prime power sources for the SPS which have the greatest impact on SFC and drag, and to utilize advanced system concepts as required to support this goal.

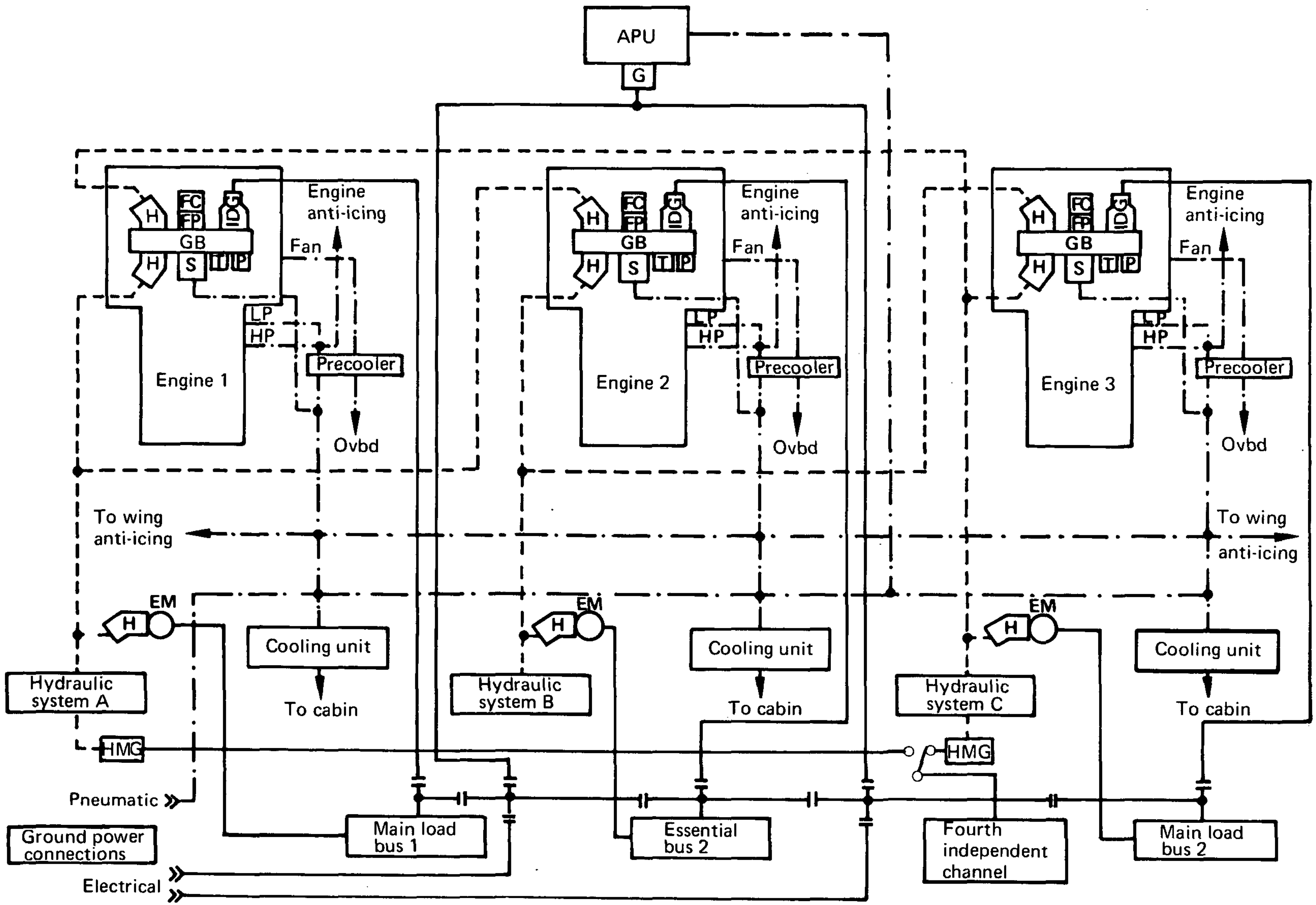
A substantial number of general SPS concepts and variations were considered to meet the requirements and objectives. These concepts were inspected and judged as to the worth of a more in-depth analysis. Budget limitations made it necessary to judge the concepts on a somewhat intuitive basis. However, the judgment was made by engineering specialists qualified by many years of experience in design and analysis of airplane secondary power systems. The four SPS concepts selected for detail analysis are described below.

System Description

Conventional SPS–Configuration I (Fig. 17)

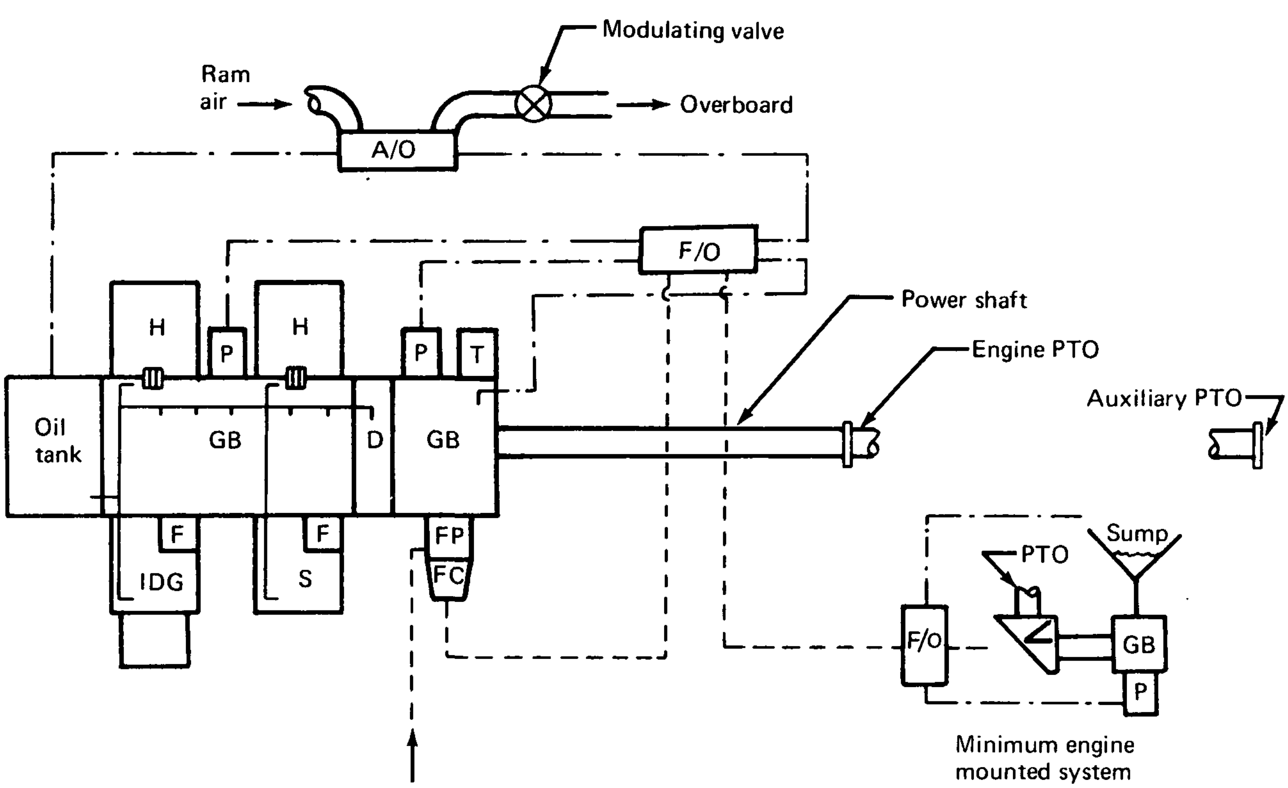
This configuration utilizes conventional-technology components. Primary power for pneumatics is two-stage bleed air from the propulsion engines. The APU is used for ground operation only. Primary power for the electrical and hydraulic systems is engine shaft power, with generators and hydraulic pumps mounted on the remote engine-driven accessory gearbox (fig. 18). A detailed description of each of the secondary power distribution systems is given below to identify components shown in the configuration diagrams and in the system trades.

Electrical system description.—The electrical system uses pad-mounted integrated drive generators (IDG), conventional transformer rectifier units, battery chargers, nickel-cadmium batteries, and static inverters. The three main generating channels and a fourth small channel powered by hydraulic motor generators are isolated in normal operation. Figure 19 is a simplified diagram showing bus arrangements and equipment sizes. Principal electrical loads are divided equally between main buses 1 and 2, normally powered by generators 1 and 3, respectively. Generator 2 is normally connected only to essential bus 2, which is lightly loaded. Main bus 1 can be connected to generator 1 or 2, and main bus 2 can be connected to generator 2 or 3. This arrangement provides for dispatch with one generator out, and no single generator is required to power more than one-half the total electrical load plus the loads of essential bus 2. An isolated "three-main-bus" arrangement would require that each generator would have to be rated to handle two-thirds of the total electrical load. Dual battery-inverter standby systems are provided to furnish that electrical power required during an all-generator- and/or all-engine-out situation.



APU	Auxiliary power unit	EM	Electric motor	HMG	Hydraulic motor generator	Ovbd	Overboard
Fan	Fan air bleed	G	Generator	HP	High-pressure bleed	P	Oil pump
FC	Fuel control	GB	Gear box	IDG	Integrated drive generator	S	Air turbine starter
FP	Fuel pump	H	Hydraulic pump	LP	Low-pressure bleed	T	Tach generator

FIGURE 17.—CONVENTIONAL SECONDARY POWER SYSTEM—CONFIGURATION 1



Nomenclature

Angle drive
Spline
---- Fuel lines
—·— Lubricating oil lines

A/O Air-to-oil heat exchanger
D Disconnect
F Filter
FC Fuel control
FP Fuel pump
F/O Fuel-to-oil heat exchanger

GB Gearbox
H Hydraulic pump
IDG Integrated drive generator
P Oil pump
PTO Power takeoff
S Starter
T Tachometer

FIGURE 18.—TYPICAL REMOTE ENGINE DRIVEN GEARBOX

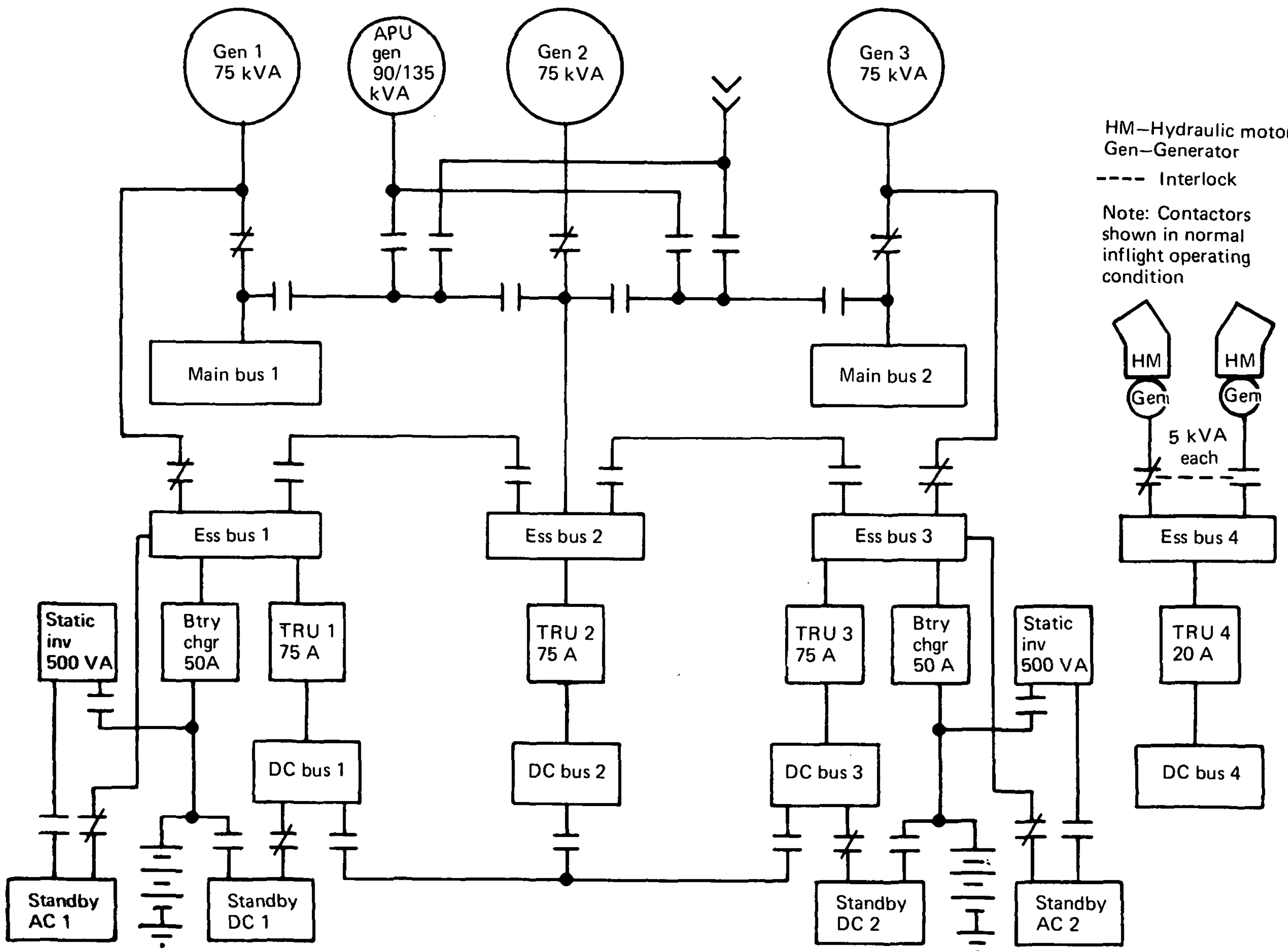


FIGURE 19.—ELECTRICAL SYSTEM SCHEMATIC—CONFIGURATIONS I AND II

Hydraulic system description.—There are two engine-driven hydraulic pumps for each of the three hydraulic systems. The two pumps in each system are driven by different engines to preclude the loss of hydraulic power to any system in the event of an engine failure. The cross connection of pumps of one hydraulic system between two different engines generally results in a weight penalty but is the best overall solution when engines are mounted in close proximity. An electric-motor-driven hydraulic pump is provided in each system to permit system checkout (pressurization) and brake accumulator charging without starting main engines.

The major components in a typical hydraulic system (excluding user equipment) are shown schematically in figure 20. Included are the hydraulic reservoir, pumps, heat exchanger, modular equipment packages, and distribution system. The case drain module contains filters, check valves, differential pressure indicator, and thermal switch. Flight control modules include the check valve, motor-operated shutoff valve, and pressure switch. The pressure module includes filters, check valves, relief valve, differential pressure indicator, and pressure switches. The return module contains the filters, check valves, relief valves, shutoff valve, and differential pressure indicators. The motor-operated shutoff valves in the flight control modules allow the isolation of power control units in the event of a malfunction. Hydraulic systems A and C each have one hydraulic motor generator set (HMG) to provide the fourth independent channel of electric power. Only one HMG is used at any one time.

The hydraulic system flow demand (including HMG required for this configuration) and total pump flow available for the various operating modes are given in figure 21. Flow demands are satisfied for all operating modes with the exception of simultaneous demands for steering, brakes, and flap extension (or retraction) during the taxi mode. Priority valves are included to ensure that more critical functions (brakes and steering) are given priority over flap loads. This will result in somewhat slower flap operating rates, but these are acceptable for ground operation. Two 2.33×10^{-3} m^3/s (37 gpm) engine-driven and one 3.78×10^{-4} m^3/s (6 gpm) electric-motor-driven pump are installed per system. One of the two engine-driven pumps in system B will satisfy gear retraction requirements in the event of an engine failure during takeoff.

Pneumatic system description.—The pneumatic system uses two stages of engine bleed as the air source. The engine bleed provides engine inlet anti-icing, wing anti-icing, cabin conditioning, and pressurization. The air supplied to the airplane ducting is routed through precoolers near the engines wherein the discharge temperature is controlled to hold 505° K (450° F) normal maximum operating temperature downstream of the precooler. Air is extracted from the manifold for wing anti-icing and cabin air conditioning systems. Ground connections are provided so that conditioned air can be supplied directly to the cabin, or ground cart air can be supplied through the cooling unit to the cabin. In addition, APU or ground cart air can be used to start the engines, and cross-start capability is provided.

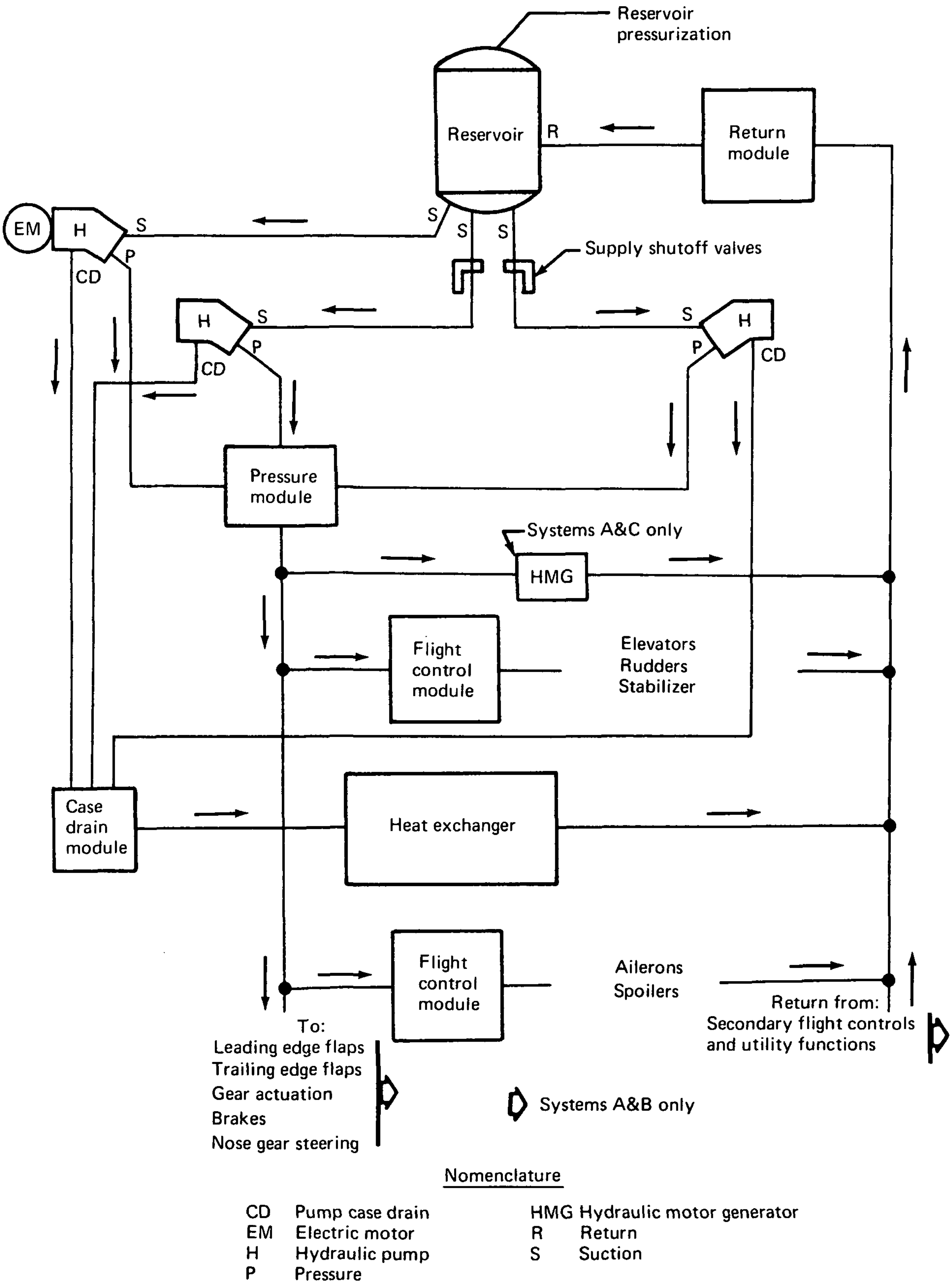


FIGURE 20.–TYPICAL HYDRAULIC SYSTEM COMPONENTS FOR CONFIGURATIONS I AND II

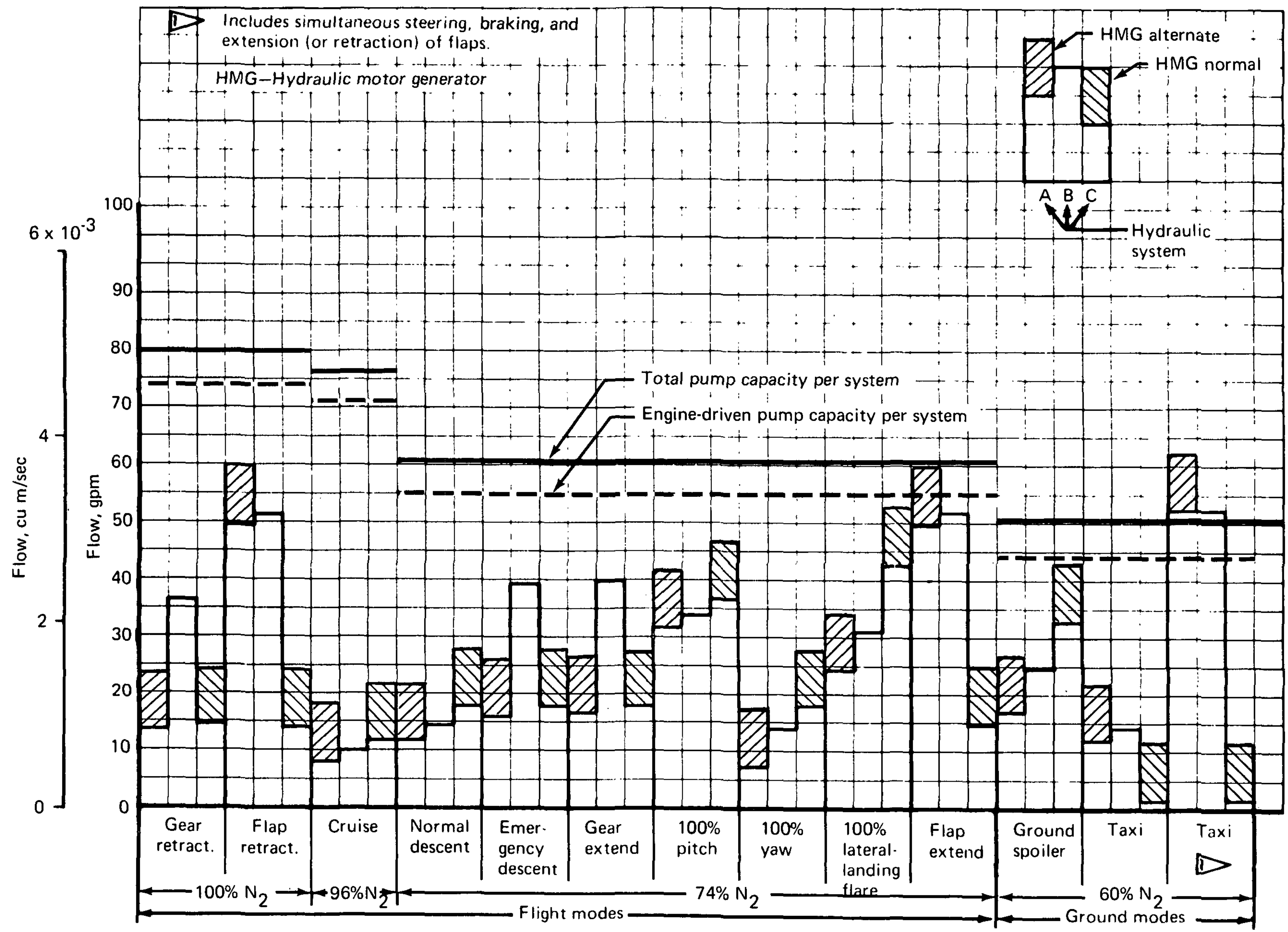


FIGURE 21.–HYDRAULIC FLOW DEMAND AND PUMP CAPACITY–CONFIGURATIONS I AND II

The cooling units and the engine air sources are manifolded to permit isolated operation between any engine and cooling unit. The cooling unit is a simple/bootstrap air cycle machine. A typical system schematic is shown in figure 22.

Three cooling units are used. They are sized such that cabin comfort can be maintained with one unit out and pressurization maintained with two units out. This allows the airline to dispatch with one cooling unit inoperative.

Conventional SPS With Shaft-Driven Compressor–Configuration II (Fig. 23)

Configurations I and II are the same except that a remote gearbox-mounted, shaft-driven compressor using fan air is the airplane pneumatic power source. This configuration was selected on the basis that the engine would be essentially free of bleed air manifolding, which would improve engine maintenance, and that shaft power results in a lower bleed air operating penalty.

The engine inlet is anti-iced by high-stage engine bleed air, while the wing is anti-iced by air from the shaft-driven compressor.

Auxiliary Power Unit SPS–Configuration III (Fig. 24)

This SPS arrangement is representative of the "dedicated" auxiliary power unit (APU) concept; i.e., 100% use on the ground and in flight. The APU provides power for the cabin air conditioning and hydraulic systems. In addition, it supplies part of the electrical power for normal operation and furnishes sufficient electrical power for electric ground starting of engines. The use of the generator/starter drive (GSD), in an effort to reduce engine frontal area, was subsequently proven to be the wrong choice. The SPS configuration was established on equipment supplier information which indicated that a smaller 90-kVA GSD was sufficient to start the propulsion engine. This rating was then changed to 135 kVA, but was received too late to allow reconfiguration of the SPS for better utilization of the electrical power. A more prudent choice would have been to retain the air turbine starter for the engine and resize the electrical power generation system.

Electrical system description.—The electrical system is characterized by a large generating capacity. Sizing the GSD to start the propulsion engines resulted in a per-channel capacity of 135 kVA. The APU-mounted generators (if complete self-sufficiency is required) must be sized to provide 135 kVA plus starting power to the GSDs. With all systems operating normally, a total of five times 135 kVA, or 675 kVA, of electrical power is available.

The maximum electrical power demand, on the other hand, including three electric-motor-driven hydraulic pumps, is 240 kVA.

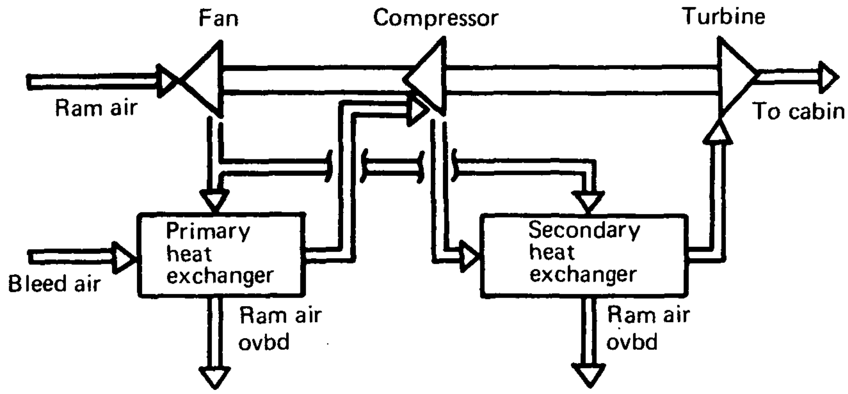
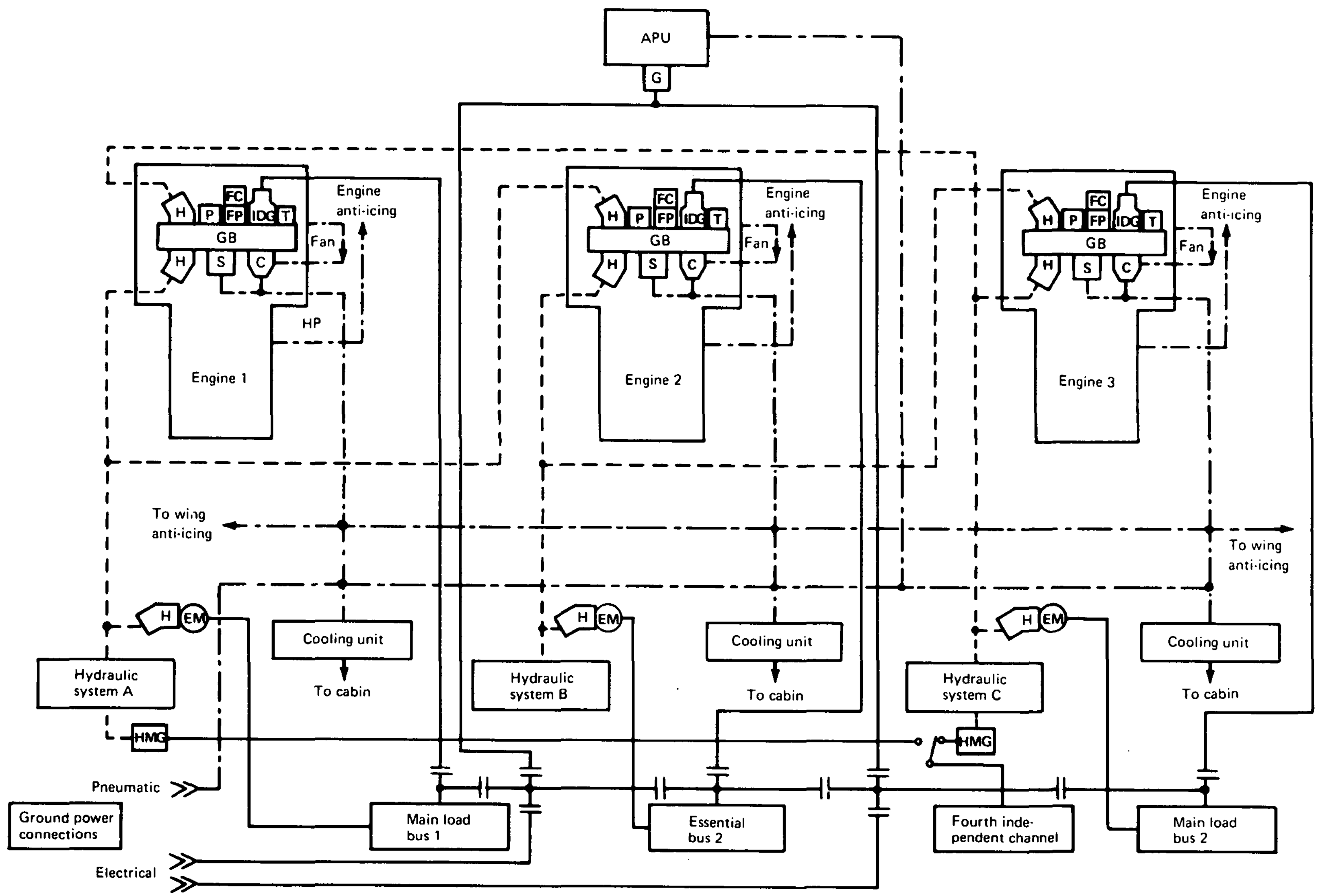
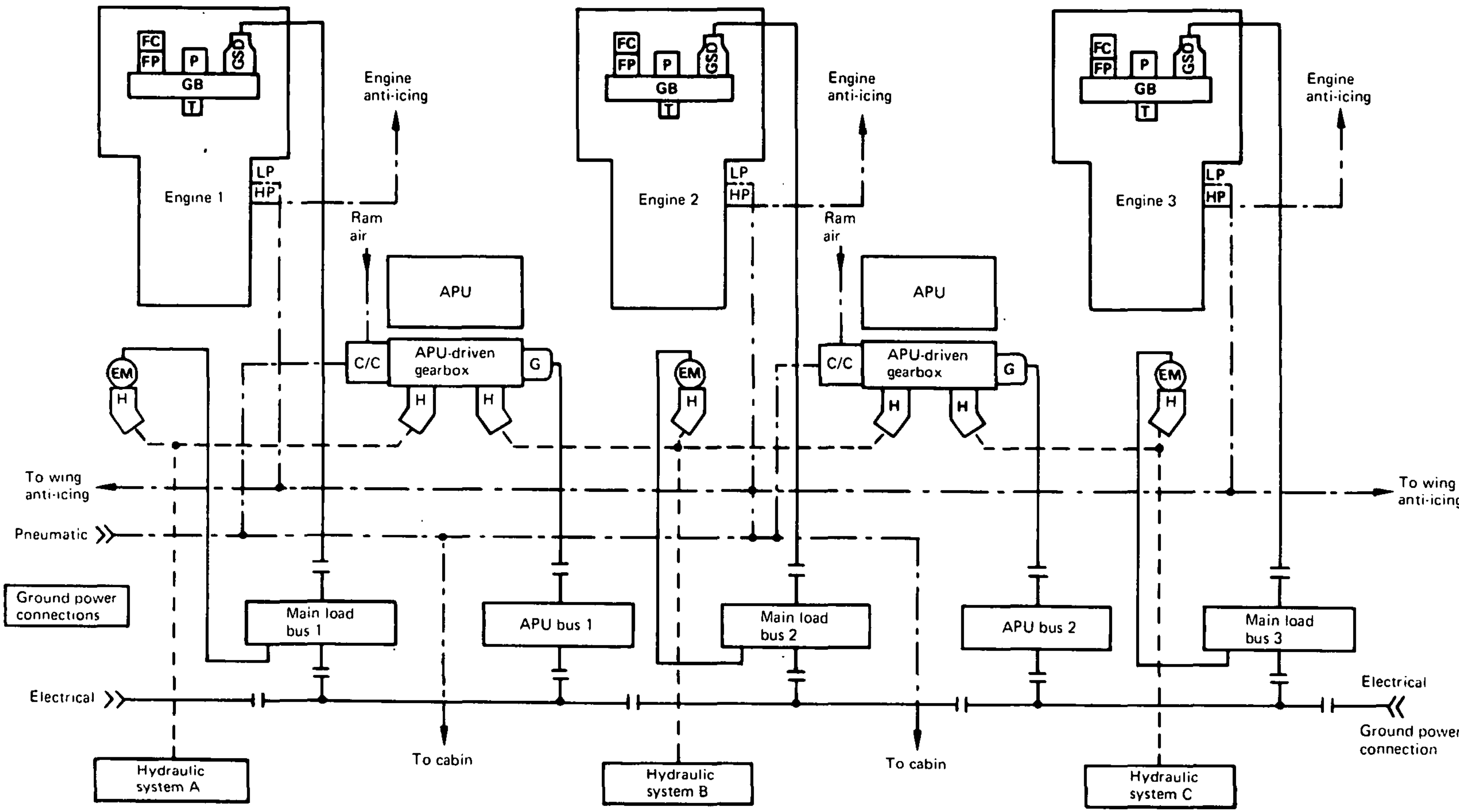


FIGURE 22.—SIMPLE/BOOTSTRAP CYCLE



APU	Auxiliary power unit	FP	Fuel pump	H	Hydraulic pump	P	Oil pump
C	Engine driven compressor	EM	Electric motor	HMG	Hydraulic motor generator	S	Air turbine starter
Fan	Fan air bleed	G	Generator	HP	High-pressure bleed	T	Tach generator
FC	Fuel control	GB	Gearbox	IDG	Integrated drive generator		

FIGURE 23.—CONVENTIONAL SECONDARY POWER SYSTEM WITH SHAFT-DRIVEN COMPRESSOR—CONFIGURATION II



APU	Auxiliary power unit	FC	Fuel control	GSD	Generator starter drive	LP	Low-pressure bleed
C/C	Compressor/cooling unit	G	Electric generator	H	Hydraulic pump	P	Oil pump
EM	Electric motor	GB	Gearbox	HP	High-pressure bleed	T	Tach generator
FP	Fuel pump						

FIGURE 24.—AUXILIARY POWER UNIT SECONDARY POWER SYSTEM—CONFIGURATION III

The bus arrangement and equipment size for this configuration is shown in figure 25.

Hydraulic system description.—The hydraulic system for configuration III is identical to that for configurations I and II except for changes in power-generating equipment and the elimination of HMGs. Minor changes were made in the distribution system to accommodate changes in pump sizes and the location of the APUs. Systems A and C include two 1.64×10^{-3} m^3/s (26 gpm) pumps, one APU driven and one electric motor driven. System B includes two 1.64×10^{-3} m^3/s (26 gpm) APU-driven and one 3.78×10^{-4} m^3/s (6 gpm) electric-motor-driven pump. The pump size in system B is dictated by gear retraction requirements in the event of a pump failure or an APU failure on takeoff. Some degradation in gear retraction time after failure was allowed, consistent with current practice. The flow required was fixed at 2.02×10^{-3} m^3/s (32 gpm) (100% of flight control plus 80% of gear retraction demands satisfied). This flow requirement will be maintained in system B under the conditions stated. System flow demands and pump capacities are shown in figure 26.

Configuration III requires both APUs to be operable for dispatch. The sizing of the pumps and the hydraulic system power supply arrangement are based on that requirement.

Consideration was given to what changes would be required in the hydraulic power supply to allow dispatch with any one of the two APUs inoperative. The hydraulic pump arrangement shown in figure 27 will provide flow capacity and redundancy with one APU inoperative. This variation will result in a weight penalty of 149 kg (328 lb) and was not used.

Pneumatic system.—The pneumatic system differs considerably from that of configurations I and II in that only engine and wing anti-icing are provided by engine bleed air. Cabin air is provided by two APU-driven powered bootstrap cooling units. These units have sufficient compressor capability to provide the cabin pressurization. Fresh air is compressed by the compressor wheel element. The air is then routed through a heat exchanger and back through the turbine wheel element to be cooled. Controls and gearing are required to match the compressor and turbine to the cabin requirements. A typical cooling unit schematic is shown in figure 28. This unit would require extensive development but offers potential to eliminate one heat exchanger per cooling unit, eliminate most of the engine bleed air manifolding, and improve maintenance through the reduction of other components required in the simple bootstrap cooling units.

A bleed air backup system is provided to pressurize the cabin in the unlikely event that both cooling units should fail. The proximity of the wing anti-ice supply line and the conditioned air manifold allows this cross tie to be made for a nominal weight penalty of 4.5 kg (10 lb).

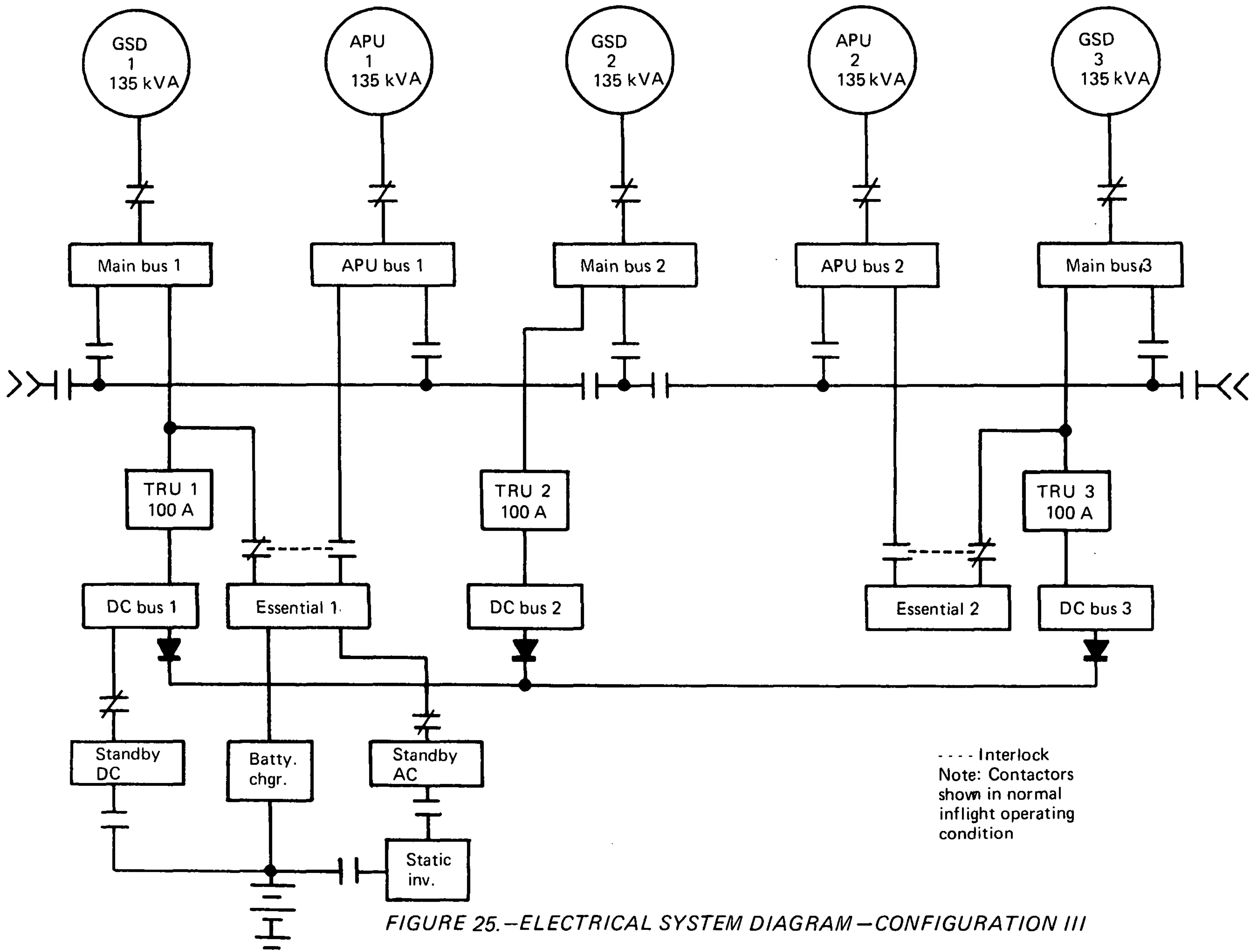


FIGURE 25.—ELECTRICAL SYSTEM DIAGRAM—CONFIGURATION III

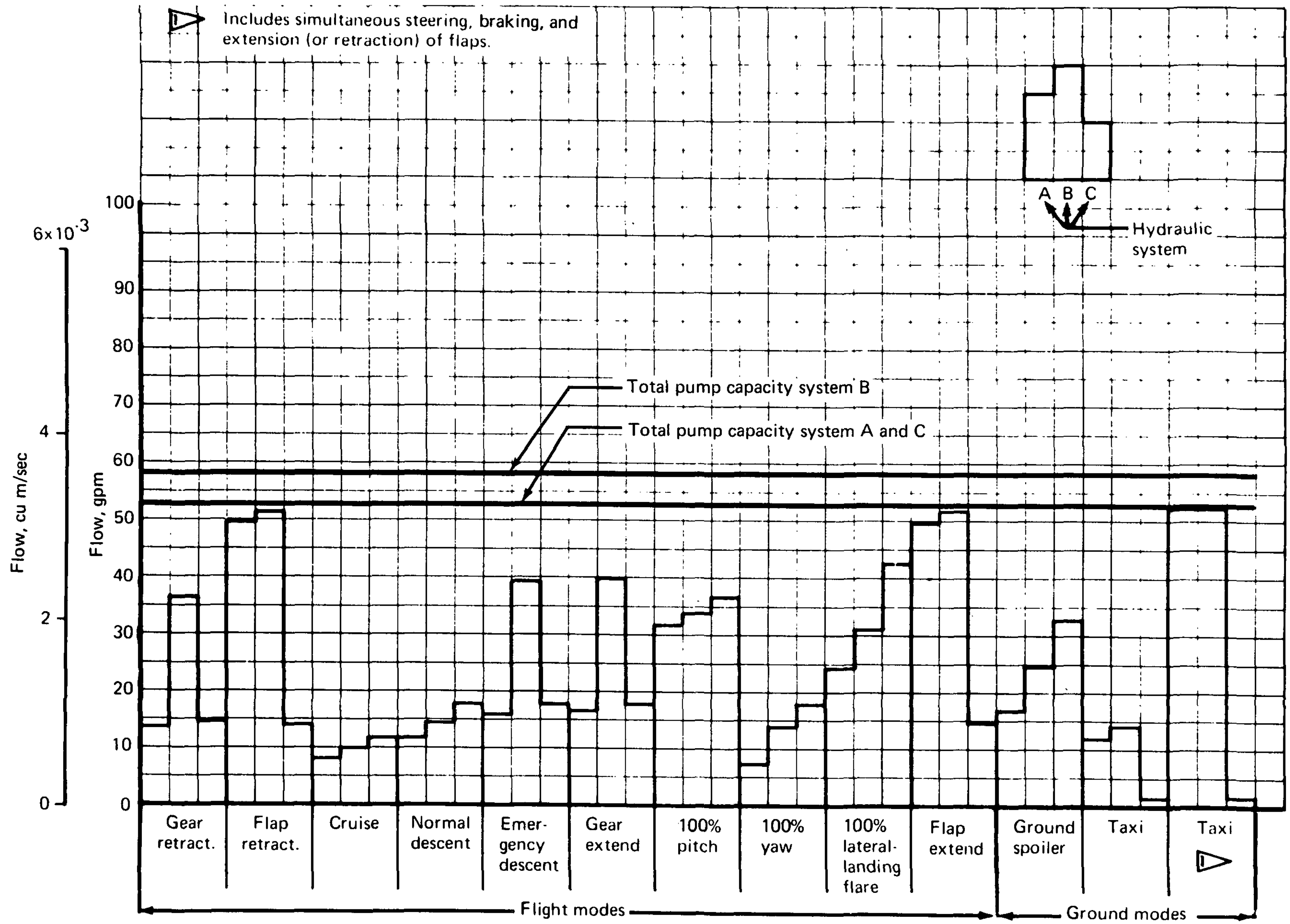
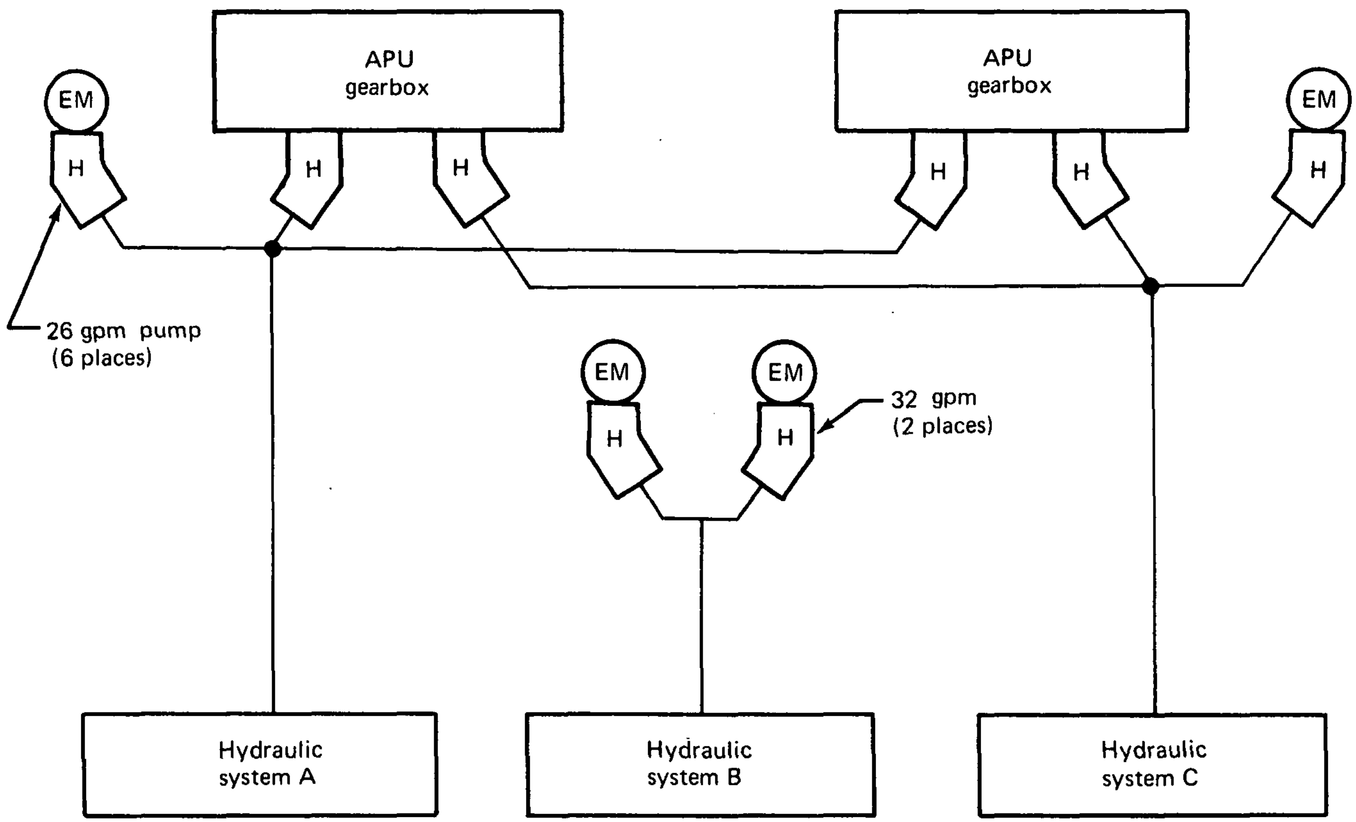


FIGURE 26.—HYDRAULIC FLOW DEMAND AND PUMP CAPACITY—CONFIGURATION III



APU – Auxiliary power unit
EM – Electric motor
H – Hydraulic pump

FIGURE 27.–ALTERNATE HYDRAULIC POWER SUPPLY–CONFIGURATION III

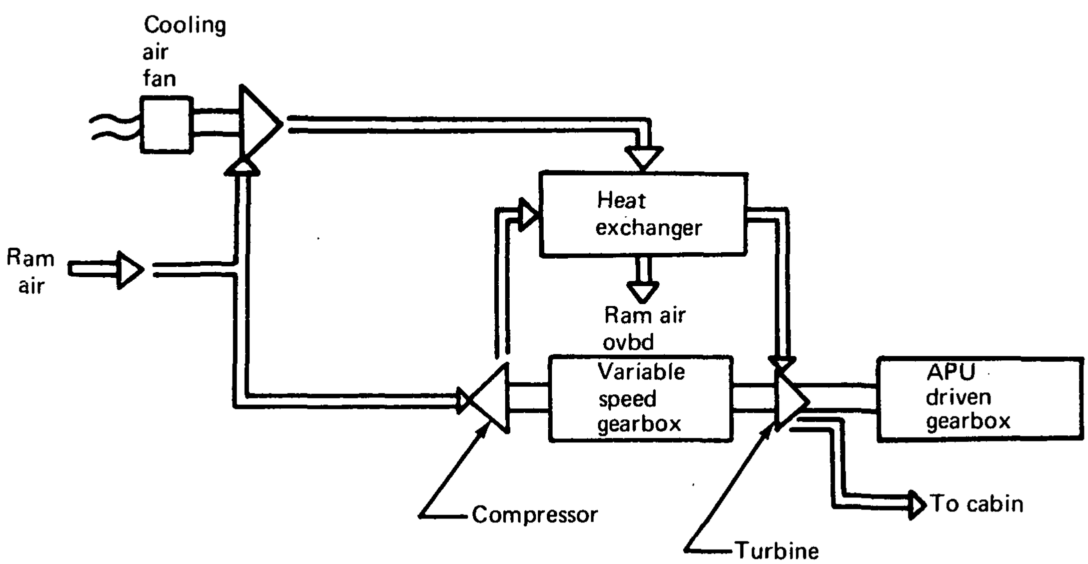


FIGURE 28.–POWERED BOOTSTRAP AIRCYCLE COOLING UNIT

Internal Engine Generator SPS—Configuration IV (Fig. 29)

The internal engine generator (IEG) configuration integrates the generator installation within the engine frame (fig. 30). The generator performs the engine starting and electric power generating functions and minimizes the number of SPS accessory components installed on the engine. An APU which is ground operable only provides the air source for ground air conditioning and electric power for ground operation and engine starting.

Electrical system.—The IEG supplies power for all SPS use. Engine shaft power is converted directly to electrical energy, and large portions are converted to pneumatic or hydraulic power by large electric-motor-driven compressors or hydraulic pumps. The IEG, as a synchronous motor, performs the engine starting function. Other electrical energy sources are the APU generator and the standby battery-inverter system. The peak power to perform required SPS functions (fig. 5) exceeds 596 kW (800 hp). The maximum average load used to determine generator size is approximately 522 kW (700 hp). Each of the three IEG systems is rated at 340 kVA so that no load reduction is required after a failure of one generating channel.

A bus arrangement and equipment size for this configuration is shown in figure 31. Each generator/converter is connected to two main buses during normal operation. Load bus transfers in the event of failure may be accomplished as indicated in the table of figure 48. This arrangement is assumed so that generator sizing can be based on the formula (generator rating equals one-half of the average peak airplane load plus system losses).

The dc-link type of VSCF system was used for this study because the only available data to confirm sizing parameters for this size unit was from the NASA/Lewis computer program.

The dc-link variable-speed, constant-frequency (VSCF) system is an adaptation of the well-known square wave bridge. The resulting dc is switched in the proper sequence to various sections of a multitapped three-phase transformer. Thus, the rectifier forms a "link" between the generator and the "inverter." Switching of the dc to the output transformer is accomplished by silicon controlled rectifiers (SCRs) which are programmed by control circuitry to switch in the proper sequence to produce a quasi-ac output. The output transformer is designed to provide a "stepped" approximation of the sine wave, and filters are employed to provide the quality of power required.

In the generating mode, output power is a constant 400 Hz, and frequency is regulated by the "firing" control circuitry. Voltage is regulated by generator excitation control. In the starting mode, the dc-link input is disconnected from the generator and connected to a power source. The output is disconnected from the load bus and connected to the generator.

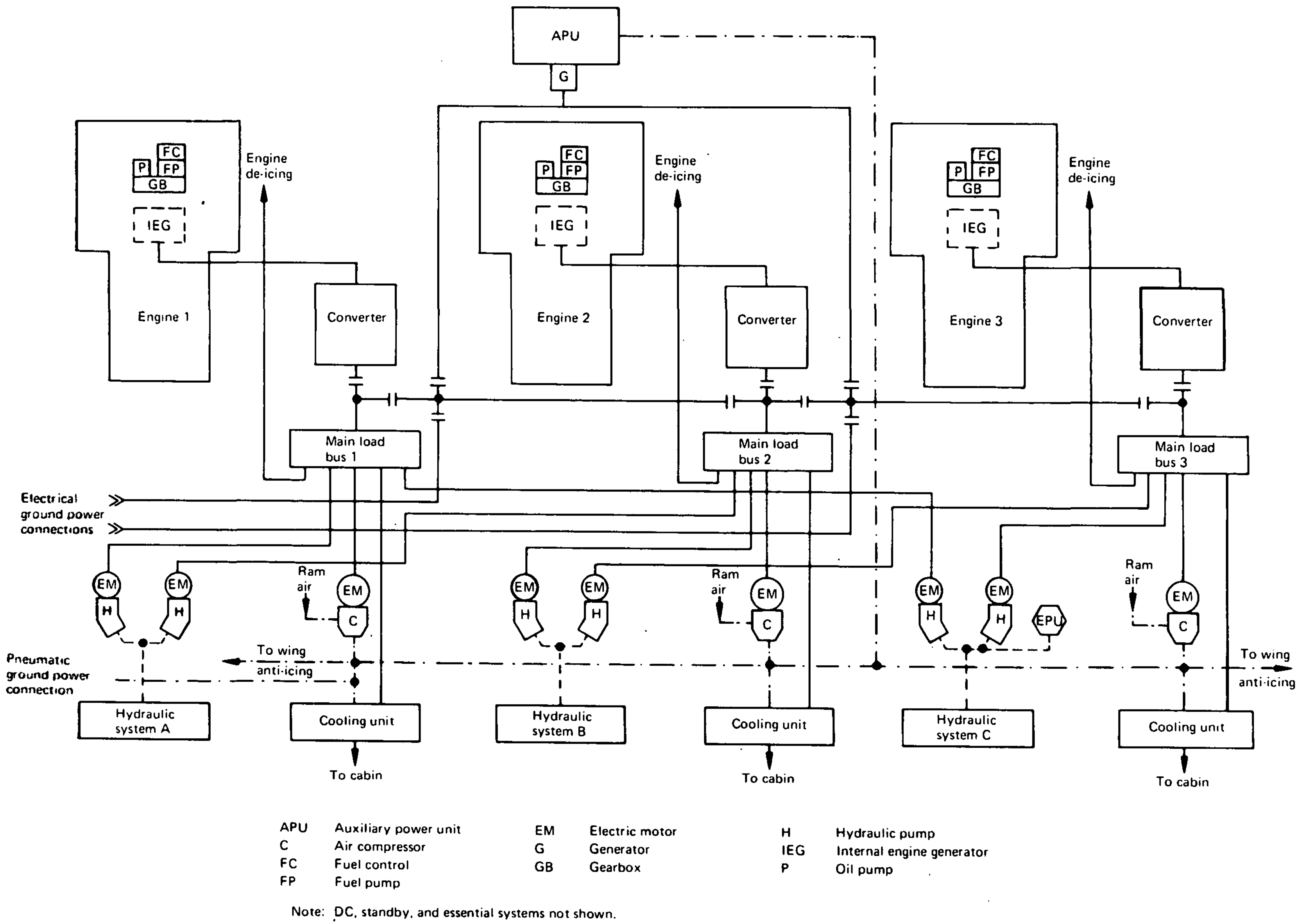


FIGURE 29.—INTERNAL ENGINE GENERATOR SECONDARY POWER SYSTEM—CONFIGURATION IV

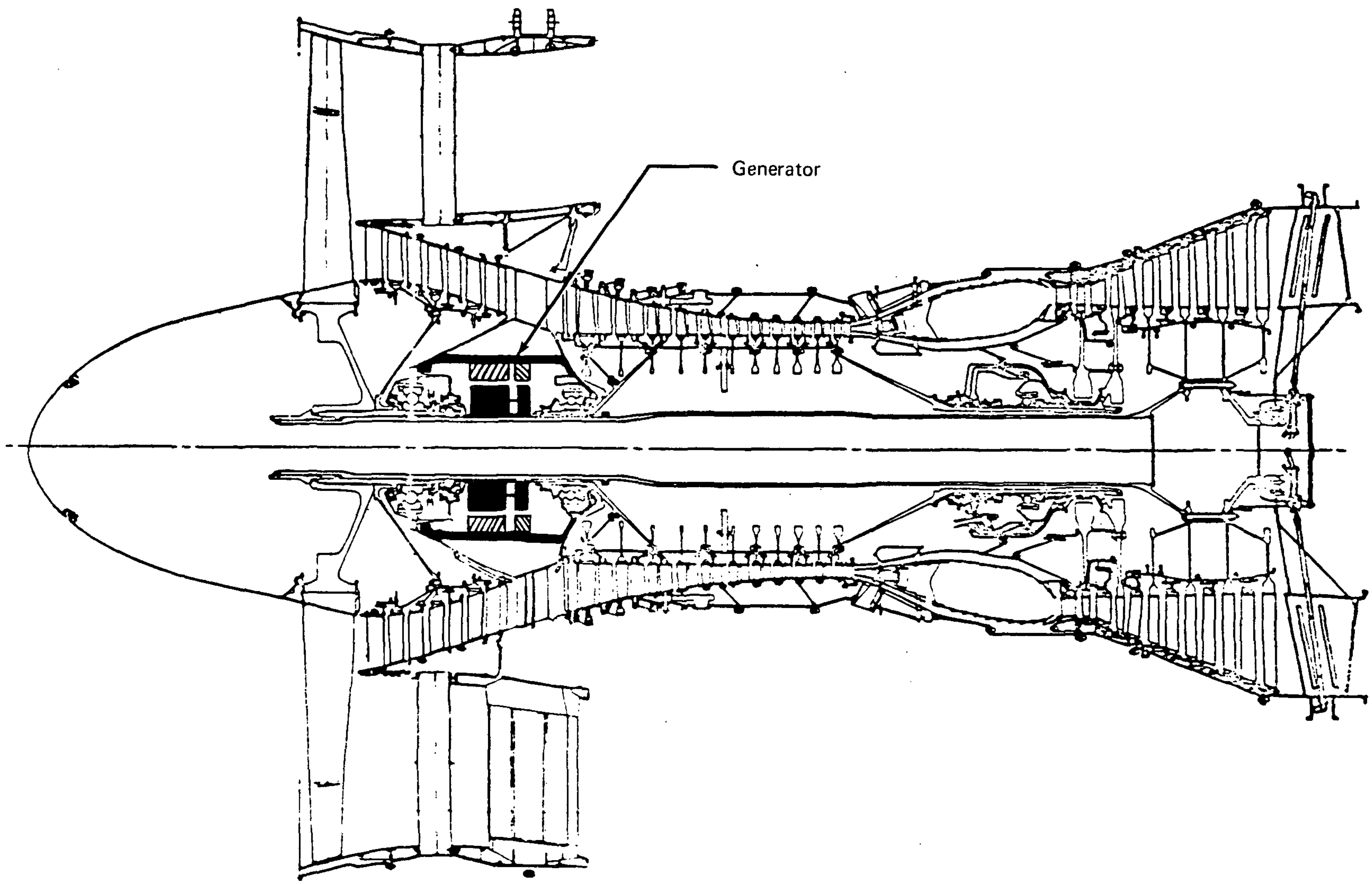


FIGURE 30.—INTERNAL ENGINE GENERATOR

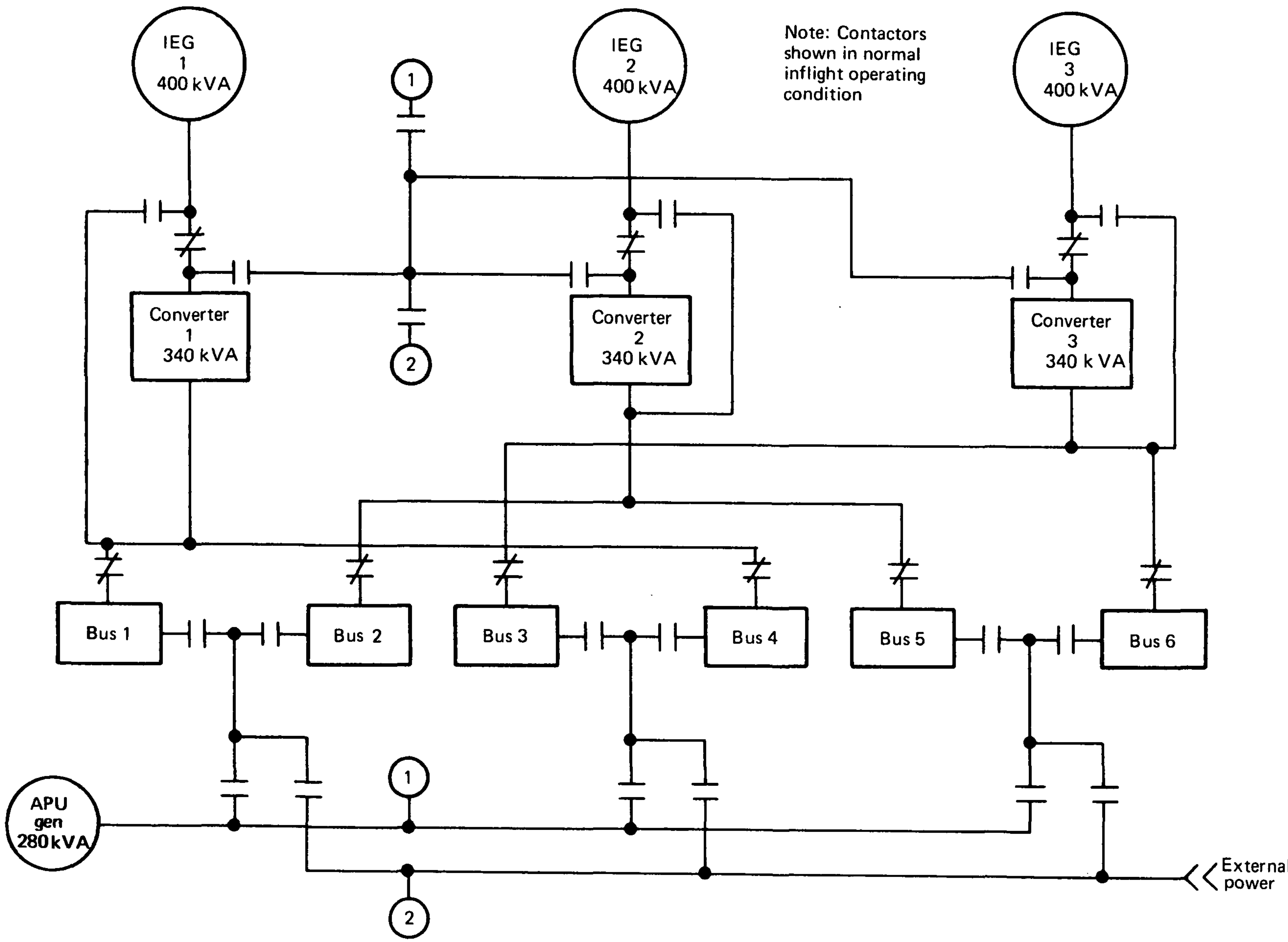


FIGURE 31.–ELECTRICAL SYSTEM DIAGRAM–CONFIGURATION IV

Operation of the IEG as a synchronous motor requires that it be supplied closely controlled frequency and voltage. As motor speed increases, higher frequency and voltage is automatically programmed to maintain the required torque.

The cycloconverter VSCF approach can be considered sufficiently comparable to the dc-link VSCF, for study purposes, that the estimates for dc-link are also valid for the cycloconverter. Further discussion of the cycloconverter system and a comparison of the two systems are contained in part II, "Trade Study, System Description."

Hydraulic system description.—User end item functions for this configuration are identical to those of configuration II. All hydraulic power is generated by electric-motor-driven hydraulic pumps installed in the wheel well. These power source locations result in some changes in the hydraulic distribution system. Systems A and C are each powered by two 1.64×10^{-3} m^3/s (26 gpm) pumps, while system B contains two 2.02×10^{-3} m^3/s (32 gpm) pumps. The requirement for larger pumps in system B stems from the landing gear retraction demands, which must be satisfied in the event of a critical engine failure on takeoff. Theoretically the loss of electrical power to the pump drive can be reinstated in a matter of seconds by transferring the load to a different electrical load bus. However, the gear retraction takes only about 10 sec total, and the delay of 4-7 sec in reinstating power and the time required to accelerate the motor-pump combination from a standstill would be unacceptable. Therefore, both pumps in system B must be sized to provide 2.02×10^{-3} m^3/s (32 gpm). System flow demands and pump delivery characteristics are similar to those of configuration III. Normal practice would be to use the larger pump in all systems for standardization.

The electric-motor-driven pump arrangement has potential advantages. The loss of an engine does not necessarily mean any reduction in hydraulic pumping capacity due to pump loss. Only a temporary reduction in flow capacity would be experienced until the electric motor load can be transferred to a different load bus. The extent to which this can be accomplished depends on generator sizing and load switching arrangements. The use of electric motor pump drives gives some flexibility in the placement of the motor-pump combination and the number of units used. The units were located in an unpressurized area to eliminate the need for special provisions to prevent toxic vapors from entering the passenger cabin.

The emergency power unit in system C provides hydraulic power to primary flight control in the event of an all-engines-out condition.

Pneumatic system.—The pneumatic system is independent from the engine. Ram air is compressed by three electric-motor-driven compressors to provide wing anti-icing and cabin air conditioning. The delivered air pressure is that required to pressurize the cabin, meet duct system pressure losses, and provide sufficient heat of compression to accomplish wing anti-icing. Cooling of cabin supply air is done by motor-driven vapor cycle units. The system is shown schematically in figure 32.

System Weight Comparison

SPS weights were generated by using the weights established for the model 767-611 airplane by the ATTP study. These weights were modified to meet the particular requirements of the four selected SPS configurations with respect to equipment placement, power supply, equipment size, and degree of component integration. The weights are summarized in table 7 and are divided into four major categories: prime power source, electrical systems, hydraulic systems, and pneumatic systems. The weights shown are delta changes between configurations and do not show overall system weight for any one system. The individual system weight comparison, however, shows a more detailed weight breakdown in those areas where additional clarification is desired. Pertinent items in each category are discussed below.

TABLE 7.—SECONDARY POWER SYSTEM WEIGHT SUMMARY—MODEL 767-611

Item	System concept						
	Conventional technology			Dedicated APU		Internal engine generator IV	
	Bleed/shaft I	Shaft II					
		lb	(kg)	lb	(kg)	lb	(kg)
Prime power source							
Gearbox and shaft	Base	150	(68.2)	-335	(-152.5)	-885	(-402)
APU	Base	0		993	(451)	-54	(-24.6)
Electrical system	Base	0		585	(266)	2488	(1130)
Hydraulic system	Base	0		4	(1.8)	352	(160)
Pneumatic system[a]	Base	36	(16.4)	-569	(-258)	1256	(570)
APU fuel	Base			[b]450	(205)		
Total	Base	186	(84.5)	678	(308)	3157	(1435)

[a] Includes air conditioning system.

[b] Based on 1000-nmi trip, not included in total as OEW increase but accounted for as engine SFC increase.

Prime Power Source

The prime power source weight summary is shown in table 8. The significant differences between the four selected SPS configurations are in the propulsion engine gearbox arrangement, remote SPS accessory gearbox arrangement, and APU quantity. A typical gearbox arrangement is shown in figure 33. The APU weights are based on scaled versions of the Hamilton Standard ST6C-400 series unit which is in production.

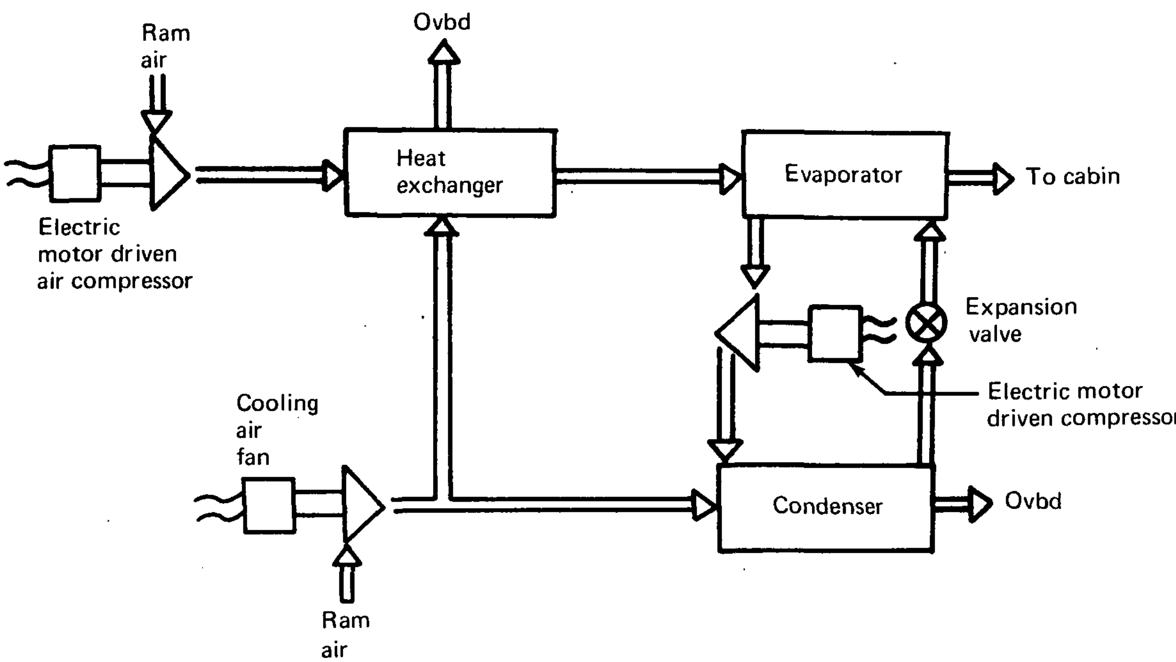
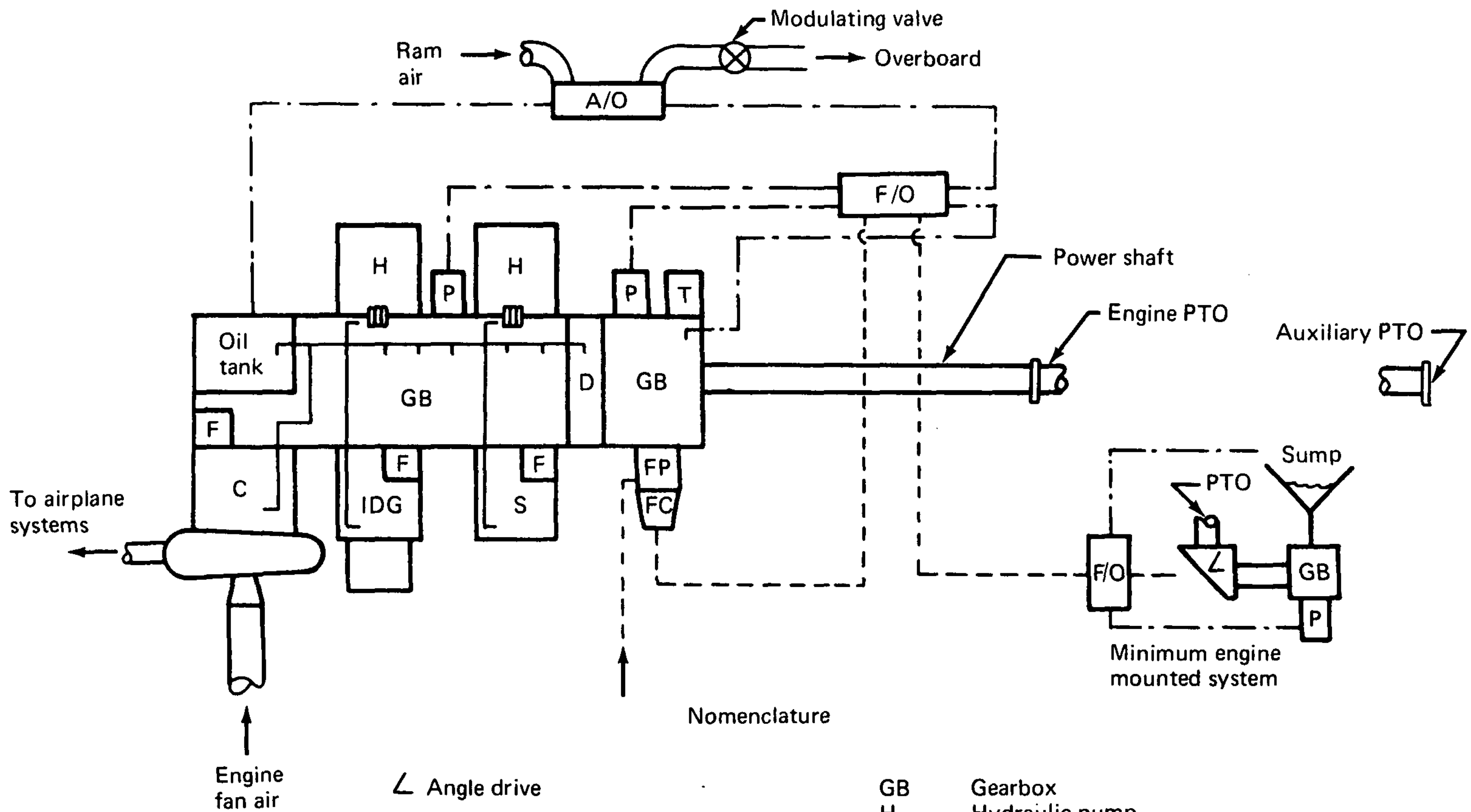


FIGURE 32.—VAPOR CYCLE COOLING UNIT

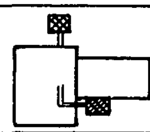
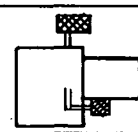
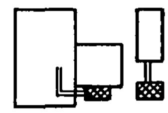
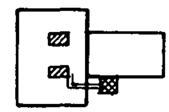


Nomenclature

∠	Angle drive
⫼	Spline
– – – –	Fuel lines
— - —	Lubricating oil lines
A/O	Air-to-oil heat exchanger
C	Compressor
D	Disconnect
F	Filter
FC	Fuel control
FP	Fuel pump
F/O	Fuel-to-oil heat exchanger
GB	Gearbox
H	Hydraulic pump
IDG	Integrated drive generator
P	Oil pump
PTO	Power takeoff
S	Starter
T	Tachometer

FIGURE 33.—TYPICAL REMOTE GEARBOX (SHAFT DRIVEN COMPRESSOR)

TABLE 8.–PRIME POWER SOURCE WEIGHT SUMMARY–MODEL 767-611

Item	Secondary power system concept							
	Conventional technology				Dedicated APU III		Internal engine generator IV	
	Bleed/shaft I		Shaft II					
Schematic: Gearbox ▩ Internal generator ▨								
	lb	(kg)	lb	(kg)	lb	(kg)	lb	(kg)
Minimum engine gearbox and drive								
Austere gearbox	225	(102.0)	225	(102.0)	225	(102.0)	225	(102.0)
Tower shaft	30	(13.6)	30	(13.6)	30	(13.6)	30	(13.6)
Added gearbox and drive weight								
– Generator/starter drive					300	(136.0)		
– Engine components drive							150	(68.0)
Remote gearbox								
Gearbox	750	(339.5)	900	(408.0)				
Shaft	75	(34.0)	75	(34.0)				
Alternate power takeoff	30	(13.6)	30	(13.6)				
Tower shaft	75	(34.0)	75	(34.0)				
Disconnect	75	(34.0)	75	(34.0)				
Cooler	30	(13.6)	30	(13.6)				
APU								
Engine	456	(206.5)	456	(206.5)	1449	(656.0)	402	(182.0)
Cruise trip fuel for 1000-nmi trip					[a]450	([a]204.0)		
APU gearbox for cooling unit					400	(181.0)		
Total weight	1746	(709.8)	1896	(859.3)	2404	(1088.6)	807	365.6
Delta weight	Base		+150	(+68.5)	+658	(+297.8)	-939	(-425.2)

[a]Not included in total as OEW increase but accounted for as increase in engine SFC

Electrical System

The electrical system weight is shown in table 9. The significant differences between the four selected SPS configurations are in the power generating systems. Typical weight-to-power parameters for major components are:

IDG	0.55 kg/kVA	(1.2 lb/kVA)
APU generator	0.36 kg/kVA	(0.8 lb/kVA)
Converter	0.23 kg/kVA	(0.5 lb/kVA)

TABLE 9.—ELECTRICAL SYSTEM WEIGHT SUMMARY—MODEL 767-611

Item	Secondary power system concept					
	Conventional technology I and II		Dedicated APU III		Internal engine generator IV	
	lb	(kg)	lb	(kg)	lb	(kg)
Integrated drive generator (IDG) with quick attach-detach	282	(128.0)				
Generator/starter drive with mounting (GSD)			585	(265.5)		
Internal engine generator (IEG)					945	(429.0)
Generator control unit (GCU)	31	(14.1)	38	(17.2)	(a)	(a)
Frequency converter					560	(254.0)
Control, protection CTS, and wiring	34	(15.4)	35	(15.9)	60	(27.2)
Generator feeders	170	(77.0)	263	(119.4)	707	(321.0)
Generator/drive heat exchanger and plumbing	108	(49.0)	162	(73.5)	[b]240	[b](109.0)
Rack cooling provisions	—	—	—	—	76	(34.5)
Installation—connectors	64	(29.0)	71	(32.2)	260	(118.0)
Contactors	41	(18.6)	70	(31.8)	174	(79.0)
APU generator and control panel	73	(33.1)	146	(66.2)	245	(111.0)
APU feeders	112	(50.8)	172	(78.0)	180	(81.6)
Hydraulic motor generator (HMG) with controls and wire	78	(35.4)	—	—	—	—
APU installation—connectors	42	(19.1)	78	(35.4)	76	(34.5)
Totals[c]	1035	(469.5)	1620	(735.1)	3523	(1598.8)
Delta weight	Base		585	(265.3)	2488	(1128.3)

[a]Part of converter.

[b]Part of engine cooling—increased for generator cooling.

[c]Total for airplane does not include components common to all systems.

The IEG-dc-link generator weight of 143 kg (315 lb) for 357 kVA was computed by NASA/Lewis for the engine idle speed condition. This size unit is capable of producing 400 kVA at cruise speed and cooling conditions. An IEG-dc-link generator weight of 143 kg (315 lb) is considered to be realistic for a 400-kVA unit.

All generators are assumed to be oil-spray-cooled machines and, with the exception of the IEG, operate at 1260 rad/sec (12 000 rpm).

All power-carrying lines are assumed to be aluminum wire except for a short length, 3.048 m (10 ft), in the engine high-temperature areas.

TABLE 10.–HYDRAULIC SYSTEMS WEIGHT SUMMARY–MODEL 767-611

Item	Secondary power system concept							
	Conventional technology		Dedicated APU				Internal engine generator	
	I and II		III		III[a]		IV	
	lb	(kg)	lb	(kg)	lb	(kg)	lb	(kg)
Distribution system	460[b]	(209)	360	(163)	414	(188)	274	(124)
Power generation[c]	268	(121)	372	(169)	646	(293)	699	(318)
Emergency power unit[d]	–	–	–	–	–	–	107	(48)
Totals[e]	728	(330)	732	(332)	1060	(481)	1080	(490)
Delta weight	Base		+4	(+2)	+332	(+151)	+352	(+160)

[a]Hydraulic power supply variation. Allows dispatch with one APU inoperable.

[b]An additional weight of 160 lb (72.5 kg) must be included in gearbox is engine mounted.

[c]Pumps, electrical feeders, and motors.

[d]Hydrazine unit with fuel supply.

[e]Total for airplane does not include components common to all systems.

TABLE 11.—WEIGHT OF TEFLON HYDRAULIC HOSE

Hose size	Medium pressure				High pressure				Weight of fluid in hose[a]
	Fire resistant		Abrasion resistant		Fire resistant		Abrasion resistant		
	Hose weight	Weight of two sets of end fittings	Hose weight	Weight of two sets of end fittings	Hose weight	Weight of two sets of end fittings	Hose weight	Weight of two sets of end fittings	
in. (mm)	lb/ft (kg/m)	lb (kg)	lb/ft (kg/m)	lb (kg)	lb/ft (kg/m)	lb (kg)	lb/ft (kg/m)	lb (kg)	lb/ft (kg/m)
1/4 (6.35)	0.163 (0.243)	0.215 (0.098)	0.134 (0.199)	0.215 (0.098)	0.260 (0.387)	0.282 (0.128)	0.220 (0.327)	0.302 (0.137)	0.0225 (0.0335)
3/8 (9.53)	0.212 (0.315)	0.400 (0.181)	0.175 (0.260)	0.405 (0.183)	0.380 (0.565)	0.456 (0.207)	0.336 (0.500)	0.506 (0.229)	0.0507 (0.0754)
1/2 (12.70)	0.299 (0.445)	0.740 (0.336)	0.254 (0.378)	0.744 (0.337)	0.522 (0.777)	0.890 (0.404)	0.485 (0.722)	0.916 (0.415)	0.0902 (0.134)
5/8 (15.86)	0.337 (0.501)	0.790 (0.358)	0.301 (0.449)	0.790 (0.358)	0.72 (1.071)	1.728 (0.784)	0.700 (1.042)	1.74 (0.789)	0.141 (0.210)
3/4 (19.05)	0.386 (0.574)	1.42 (0.644)	0.380 (0.565)	1.420 (0.644)	0.922 (1.372)	2.96 (1.342)	0.860 (1.280)	3.010 (1.365)	0.206 (0.306)
1 (25.4)	0.762 (1.134)	1.65 (0.748)	0.742 (1.104)	1.65 (0.748)	1.300 (1.934)	4.000 (1.814)	1.050 (1.562)	4.500 (2.041)	0.361 (0.537)
1-1/4 (31.75)	0.912 (1.357)	2.48 (1.125)	0.883 (1.314)	2.48 (1.125)	1.700 (2.529)	6.200 (2.812)	1.350 (2.009)	6.500 (2.948)	0.564 (0.839)
1-1/2 (38.10)	1.120 (1.666)	3.500 (1.587)	1.060 (1.577)	3.300 (1.497)	2.13 (3.169)	8.000 (3.628)	1.660 (2.470)	8.500 (3.855)	0.812 (1.208)

[a]Hydraulic fluid specific gravity 1.06

TABLE 12A.—PRESSURE LINES, TITANIUM 3Al-2.5V COLD WORKED

Item	Nominal tube size, in.					
	3/8	1/2	5/8	3/4	1	1-1/4
Outside tube diameter, in. (mm)	0.375 (9.53)	0.500 (12.70)	0.625 (15.86)	0.750 (19.05)	1.000 (25.40)	1.250 (31.75)
Tube wall thickness[a], in. (mm)	0.021 (0.533)	0.027 (0.686)	0.035 (0.889)	0.042 (1.067)	0.055 1.397)	0.070 (1.778)
Fluid weight per unit length[b], lb/ft (kg/m)	0.040 (0.059)	0.072 (0.11)	0.111 (0.165)	0.160 (0.238)	0.286 (0.425)	0.445 (0.661)
Tube weight per unit length[c], lb/ft (kg/m)	0.045 (0.067)	0.078 (0.11)	0.126 (0.188)	0.182 (0.270)	0.317 (0.472)	0.504 (0.751)
Total tube weight per unit length, lb/ft (kg/m)	0.085 (0.127)	0.150 (0.22)	0.237 (0.353)	0.342 (0.508)	0.603 (0.897)	0.949 (1.412)

[a]Calculated values based on allowable stress. Installation requirements and availability may dictate alternate choice.

[b]Hydraulic fluid specific gravity 1.06.

[c]Titanium specific gravity 4.49

TABLE 12B—RETURN LINES, ALUMINUM 6061-T6

Item	Nominal tube size, in.					
	3/8	1/2	5/8	3/4	1	1-1/4
Outside tube diameter, in. (mm)	0.375 (9.53)	0.500 (12.70)	0.625 (15.86)	0.750 (19.05)	1.000 (25.40)	1.250 (31.75)
Tube wall thickness[a], in. (mm)	0.016 (0.406)	0.016 (0.406)	0.018 (0.457)	0.020 (0.508)	0.026 (0.660)	0.033 (0.838)
Fluid weight per unit length[b], lb/ft (kg/m)	0.042 (0.063)	0.079 (0.118)	0.125 (0.186)	0.182 (0.271)	0.324 (0.482)	0.506 (0.753)
Tube weight per unit length[c], lb/ft (kg/m)	0.021 (0.032)	0.029 (0.043)	0.040 (0.060)	0.054 (0.080)	0.094 (0.139)	0.148 (0.221)
Total tube weight per unit length, lb/ft (kg/m)	0.063 (0.095)	0.108 (0.161)	0.165 (0.246)	0.236 (0.351)	0.418 (0.621)	0.654 (0.974)

[a]Calculated values based on allowable stress. Installation requirements and availability may dictate alternate choice.

[b]Hydraulic fluid specific gravity 1.06.

[c]Aluminum specific gravity 2.71.

Hydraulic System

The hydraulic system weight summary is shown in table 10. The significant differences between the four selected SPS configurations are in the distribution system, power generation, and use of an emergency power unit (EPU) for configuration IV. A typical distribution system layout is shown in figure 34.

The distribution layout and the line sizing and weight parameters in figures 35 and 36 and tables 11 and 12 were used to compute the distribution system differences. The power generation differences were computed from the pump and electrical motor weight parameters shown in figures 37 and 38.

The EPU weights were based on equipment supplier data. The fuel supply for the unit was based on the load profile shown in figure 39.

Pneumatic System

The pneumatic system weight summary is shown in table 13. The significant differences between the four selected SPS configurations are in the air supply and air conditioning system.

TABLE 13.–PNEUMATIC SYSTEMS WEIGHT SUMMARY–MODEL 767-611

Item	Secondary power system concept			
	Conventional technology		Dedicated APU III	Internal engine generator IV
	Bleed/shaft I	Shaft II		
Air supply	Base	36 lb (16.3 kg)	-387 lb (-175.5 kg)	544 lb (246.5 kg)
Air conditioning	Base	0	-42 lb (-19.0 kg)	852 lb[a] (386.0 kg)[a]
Starting system	Base	0	-140 lb (-63.5 kg)	-140 lb (-63.5 kg)
Total	Base	36 lb (16.3 kg)	-569 lb (-258.0 kg)	1256 lb (569 kg)

[a]Use of 100% recirculated cabin air on the ground would reduce this to 357 lb (162 kg).

Configuration III uses the powered bootstrap cooling machine which integrates the air compressor and cooling machine functions with the APU. This eliminates the precooler system, part of the cooling machine, and long high-pressure supply lines in the cabin which required acoustical treatment. Configuration IV, on the other hand, added an electrically driven air compressor and vapor cycle cooling unit, which resulted in a weight penalty.

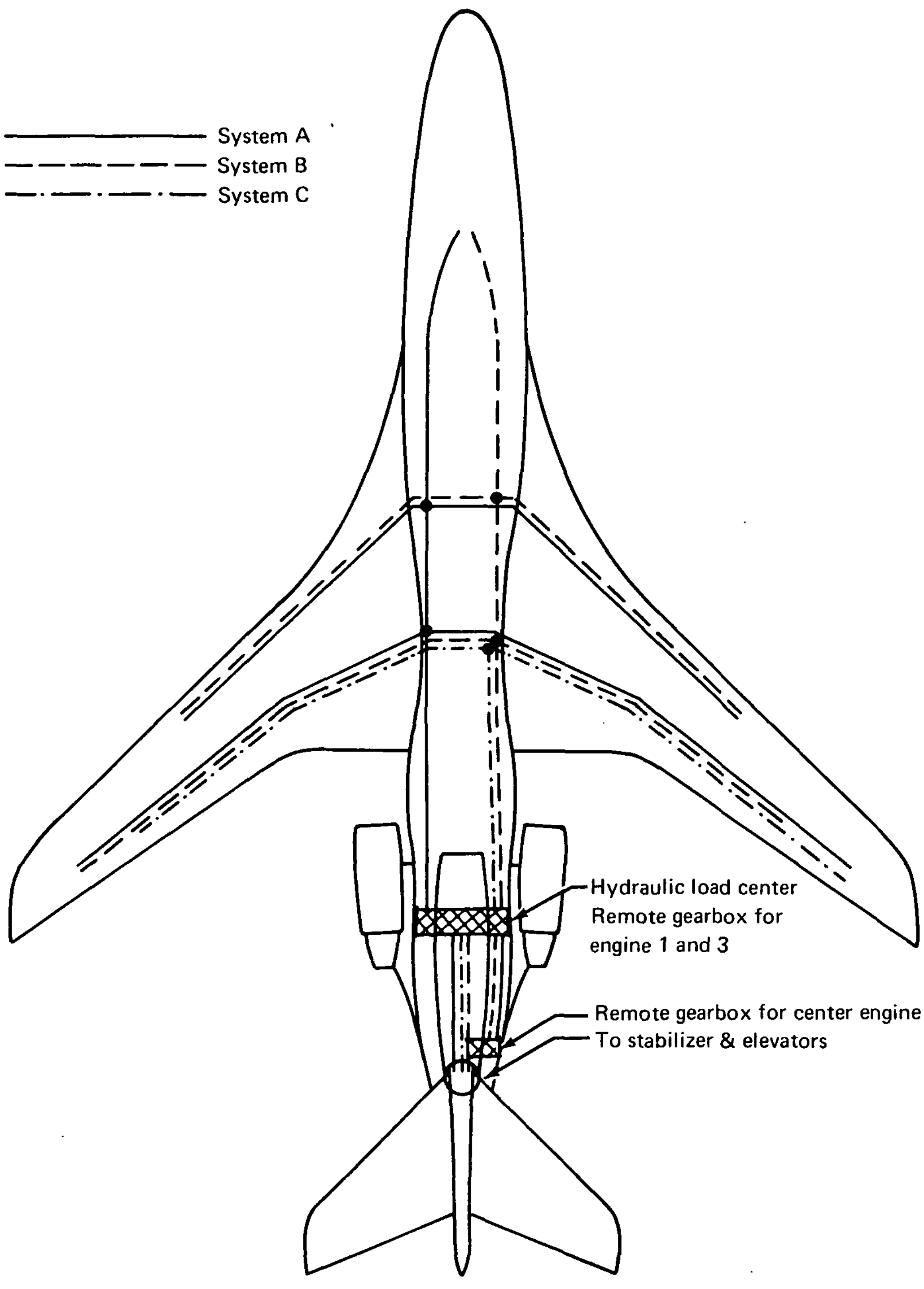


FIGURE 34.–PRESSURE LINE SCHEMATIC–CONFIGURATIONS I AND II

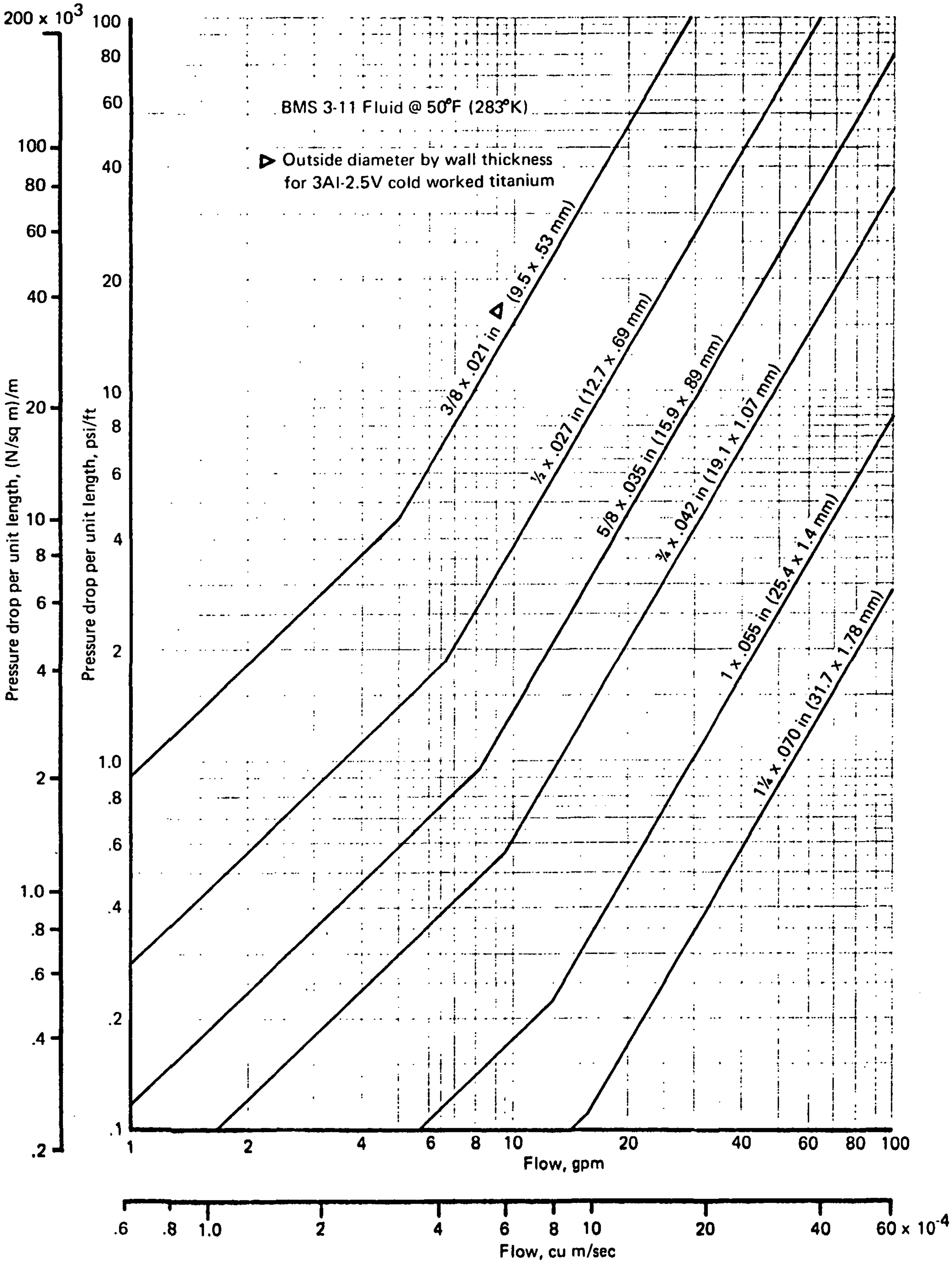


FIGURE 35.—PRESSURE LINE PRESSURE DROP PER UNIT LENGTH

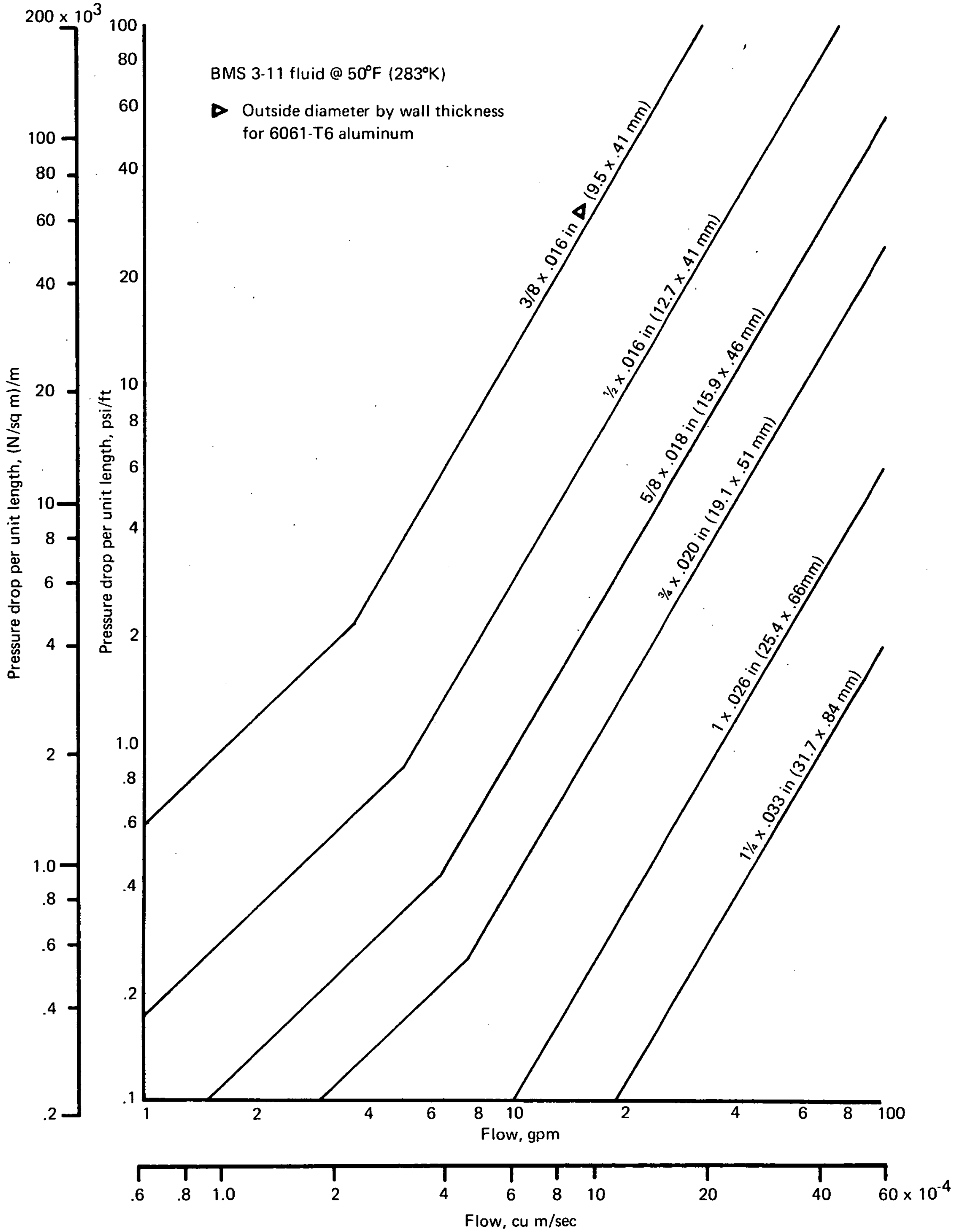


FIGURE 36.—RETURN LINE PRESSURE DROP PER UNIT LENGTH

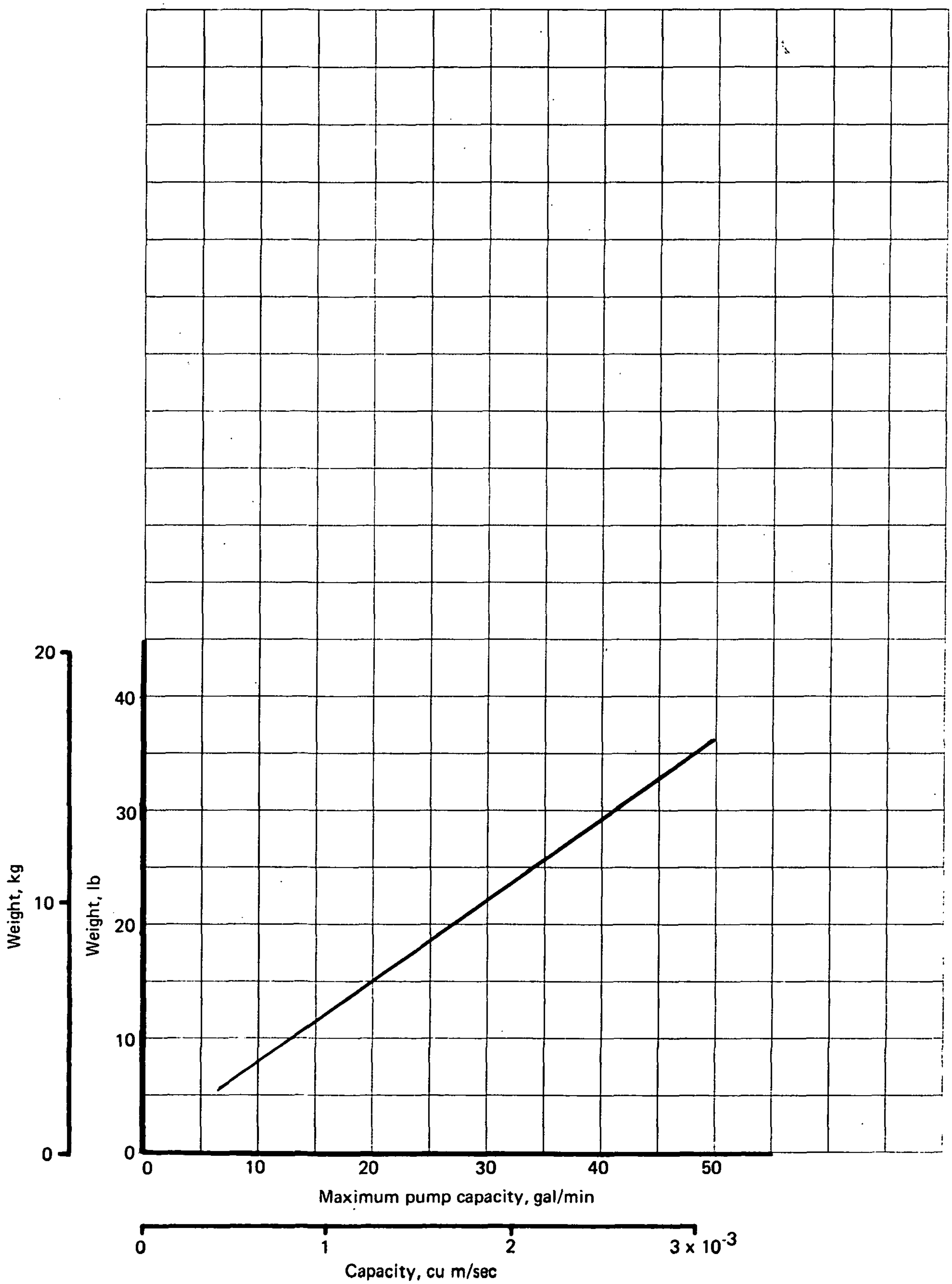


FIGURE 37.—HYDRAULIC PUMP WEIGHTS

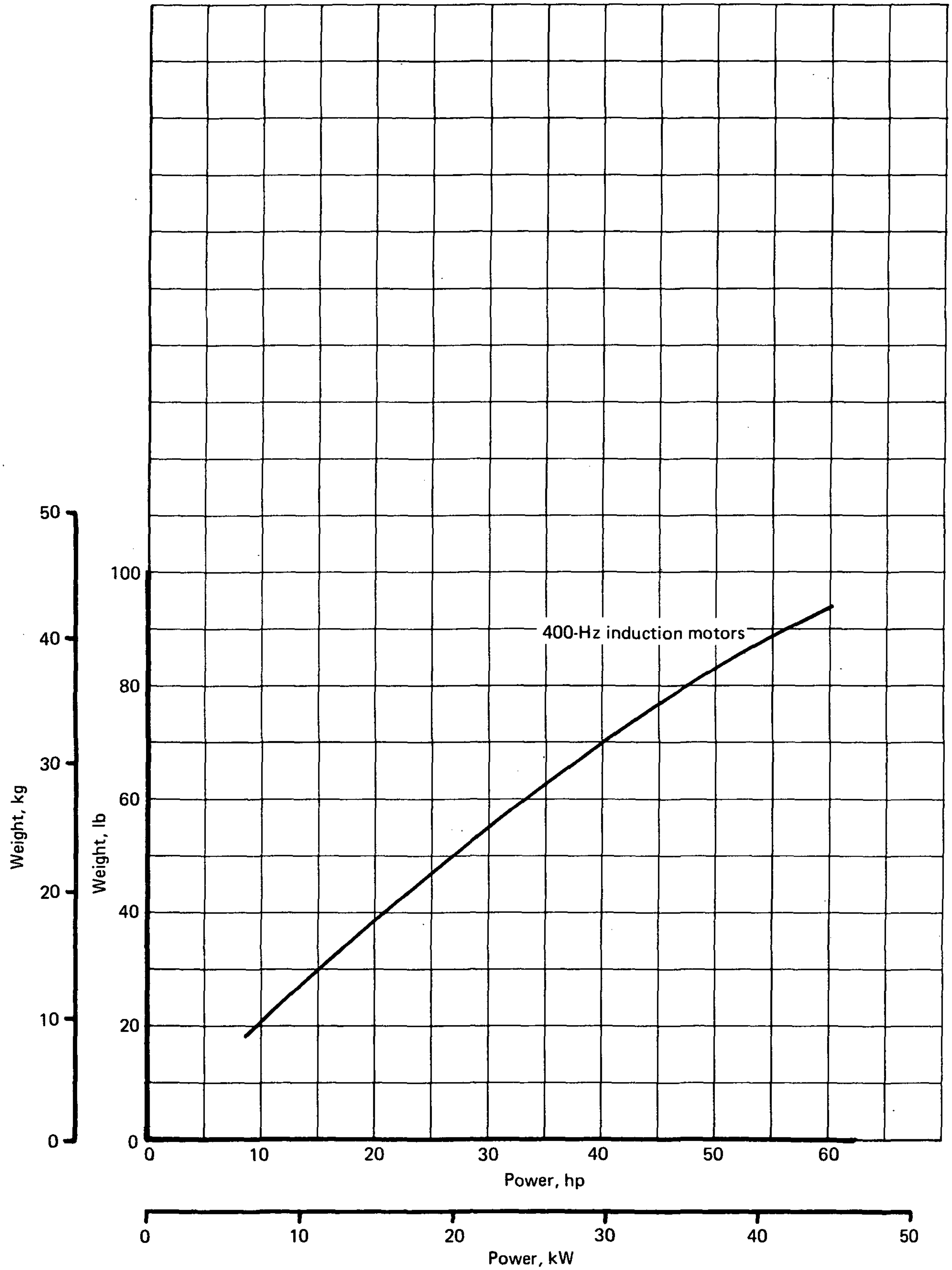


FIGURE 38.—ELECTRIC MOTOR WEIGHTS

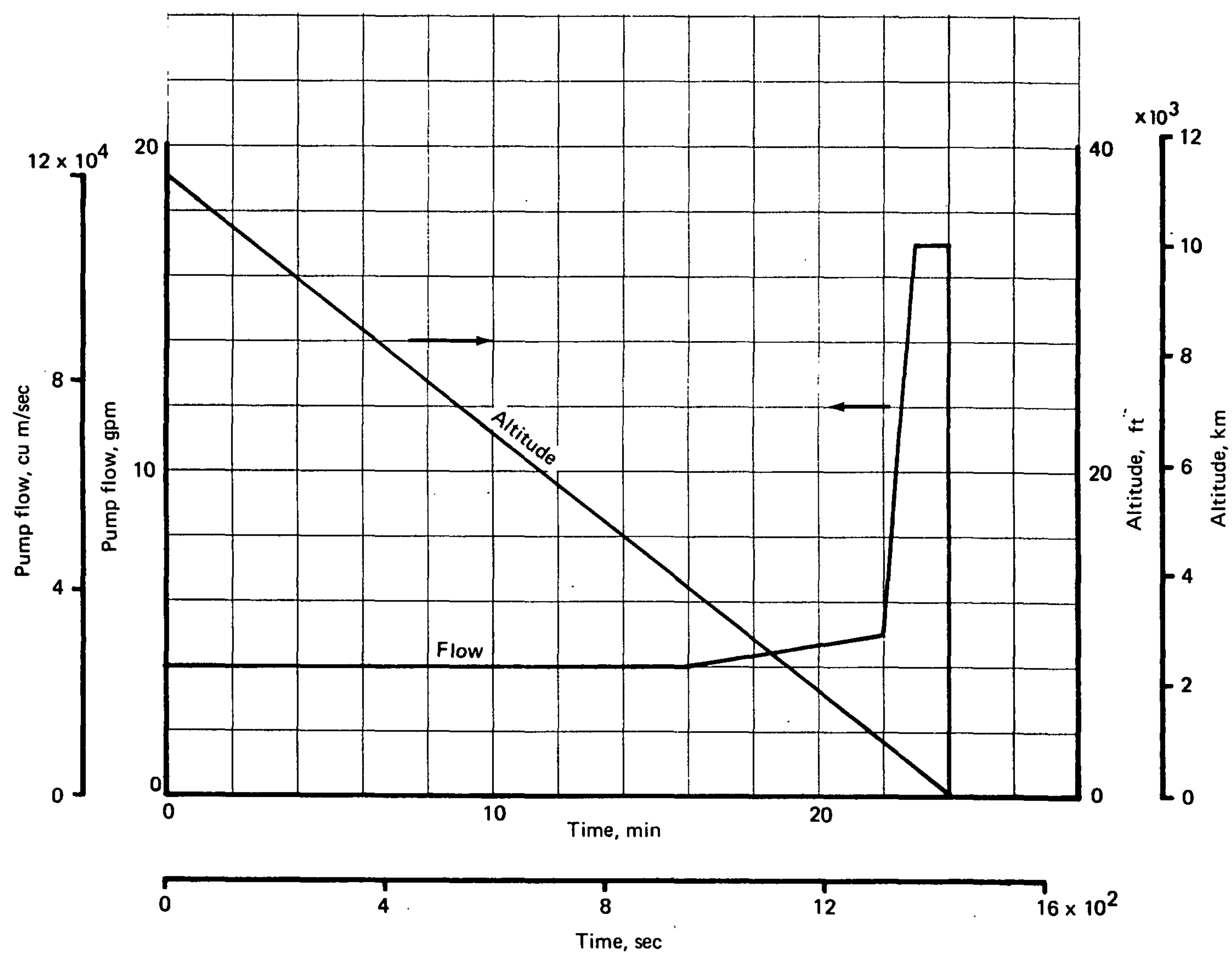


FIGURE 39.—EMERGENCY POWER UNIT LOAD PROFILE

The ice protection system weights were assumed to be the same for all four configurations.

RESULTS AND CONCLUSIONS

Results

The technical and economic payoff on a total-airplane basis for the four SPS configurations studied is shown in table 14. The results are presented as delta changes from the base SPS configuration I. Configuration I is the lowest risk system approach for the near-sonic airplane and could be initiated today for an airplane certification early in the 1975 to 1985 time period.

TABLE 14.–TECHNICAL AND ECONOMIC END RESULT STUDY SUMMARY–MODEL 767-611

OEW = 195 510 lb (88 680 kg) TOGW = 356 000 lb (161 480 kg)

Item	Unit	Secondary power system concept			
		Conventional technology		Dedicated APU III	Internal engine generator IV
		Bleed/shaft I	Shaft II		
Uncycled parameter (delta change)					
Equipment weight	% of OEW	Base	-0.10	-0.35	-1.61
Cruise SFC	%	Base	+0.42	+2.72	+1.82
Cruise thrust	%	Base	+1.75	+7.89	+2.05
Cruise drag	%	Base	0	-0.88	-0.62
Cycled TOGW for equivalent payload-range-performance objectives	%	Base	+0.41	+0.52	-1.96
	lb	Base	+1 470	+1 860	-7 000
	(kg)	Base	(+669)	(+845)	(-3 180)
Net total value of technology[b]	$/airplane	Base	+42 100	+68 591	-604 423

\+ Denotes payoff
\- Denotes penalty
[a]Reduction as a result of not extracting power from main engines
[b]Assumes equal maintenance costs

The equipment weight, cruise SFC, cruise thrust, and cruise drag differences between the four SPS configurations are also shown in table 14. The effect of these changes on a total-airplane basis can be best represented as a conversion of these parameters to a cycled takeoff gross weight parameter (table 15). The conversion factors were generated by the ATTP study and are shown in table 16a. Another measure of these changes on a total-airplane basis is to convert the changes to a "total value of technology" parameter. The total value of technology is the dollar savings realized over the

TABLE 15.—UNCYCLED PARAMETER CONVERSION TO CYCLED TAKEOFF GROSS WEIGHT FOR EQUIVALENT PAYLOAD - RANGE-PERFORMANCE OBJECTIVES—MODEL 767-611

OEW = 195 510 lb (88 680 kg) TOGW = 356 000 lb (161 480 kg)

Item	Unit	Secondary power system concept			
		Conventional technology		Dedicated APU III	Internal engine generator IV
		Bleed/shaft I	Shaft II		
Equipment weight	%	Base	–0.165	–0.577	–2.66
Cruise SFC	%	Base	+0.231	+1.495	+1.00
Cruise thrust[a]	%	Base	+0.346	+0.695	+0.465
Cruise drag	%	Base	0	–1.09	–0.769
Total	%	Base	+0.412	+0.523	–1.964
Effect on TOGW	lb	Base	+1470	+1860	–7000
	kg	Base	(+669)	(+845)	(–3180)

\+ Denotes payoff
\- Denotes penalty
[a]Reduction as a result of not extracting power from main engines

TABLE 16A.—CYCLED TAKEOFF GROSS WEIGHT SENSITIVITY FACTORS

1% change to parameter (uncycled)	% change to TOGW (cycled)		
	M = 0.90 design	M = 0.95 design	M = 0.98 design
OEW	1.30	1.42	1.65
Cruise drag	0.73	1.05	1.24
Cruise SFC	0.61	0.54	0.55
Cruise thrust	-0.12	-0.42	-0.54

TABLE 16B.—TOTAL VALUE OF TECHNOLOGY SENSITIVITY FACTORS

	Total value of technology	
	M = 0.84 design	M = 0.98 design
Uncycled OEW	$155/lb ($71/kg)	$199/lb ($91/kg)
SFC	$ 68 000/1%	$128 000/1%
Drag	$119 000/1%	$155 000/1%

life of the airplane for making the technology change. This conversion is shown in table 17, and conversion factors, also generated by the ATTP study, are shown in table 16b.

TABLE 17.–TOTAL VALUE OF TECHNOLOGY (1972 DOLLARS)–MODEL 767-611

Item	Secondary power system concept			
	Conventional technology		Dedicated APU III	Internal engine generator IV
	Bleed/shaft I	Shaft II		
Weight	Base	-37 000	-135 000	-629 000
Cruise SFC	Base	+53 700	+348 000	+233 000
Cruise thrust	Base	+63 500	+142 800	+ 79 700
Cruise drag	Base	0	-136 000	- 96 000
Equipment/engineering	Base	-38 100	-151 209	-192 123
Net total value[a]	Base	+42 100	+ 68 591	-604 423

+ Denotes payoff
– Denotes penalty
[a]Dollars per airplane—assumes equal maintenance costs.

The data used to generate the uncycled parameter changes are shown in tables 18, 19, and 20. In general, the user end cruise requirements (table 18) are converted to engine/APU power extraction by the efficiency factors (table 19). The engine/APU power extraction is then converted to specific fuel consumption and thrust factor by the penalty factors (table 19). The ram air penalty is converted to drag, assuming one momentum loss. All operating penalties are expressed in terms of specific fuel consumption, thrust, and drag.

Conclusions

The SPS configurations studied were established and traded on the basis of the model 767-611. This airplane included three engines located on the aft body section. An airplane with this arrangement can be expected to have a wider operational center-of-gravity range, with attendant balance and control problems, than an airplane with wing-mounted engines. The ATTP study ultimately selected the model 767-620 (two engines on the wings and one engine in the tail) in preference to the model 767-611 or the model 767-711 (four wing-mounted engines). It was beyond the scope of this study to evaluate the total airplane reconfiguration effect of adding weight to the aft section of the model 767-611 (dedicated APU concept). This may cast an element of doubt on the validity of ranking of the SPS configurations below:

a. Configuration III (dedicated APU)

b. Configuration II (conventional technology—all shaft power)

c. Configuration I (conventional technology—bleed/shaft power)

d. Configuration IV (IEG—all-electric SPS use).

TABLE 18.—SECONDARY POWER SYSTEM USER END CRUISE POWER REQUIREMENT—MODEL 767-611

Item	Secondary power system concept			
	Conventional technology		Dedicated APU III	Internal engine generator IV
	Bleed/shaft I	Shaft II		
Cabin air source, lb/min (kg/sec)				
Engine compressor bleed	288 (2.18)	–	–	–
Engine fan air	–	288 (2.18)	–	–
Ram air	–	–	288 (2.18)	288 (2.18)
Engine shaft power, hp (kW)				
Hydraulic system	58 (43.4)	58 (43.4)	–	–
Electrical system				
Electrical	174 (130)	174 (130)	174 (130)	174 (130)
Hydraulic	–	–	19 (14.2)	58 (43.3)
Pneumatic	–	–	–	412 (308)
APU shaft power, hp (kW)				
Hydraulic	–	–	39 (29)	–
Pheumatic	–	–	449 (335)	–

TABLE 19.—SECONDARY POWER SYSTEM POWER CONVERSION EFFICIENCY AND CRUISE ENGINE AND APU PENALTY FACTOR

A. Secondary Power System Power Conversion Efficiency	
Item	Efficiency, %
IEG power generation (engine shaft to load bus)	80
GSD and IDG power generation (engine shaft to load bus)	72
Shaft power	
Engine gearbox (remote installation)	97
APU gearbox	98
Electric motor	90

B. Cruise Engine and APU Penalty Factor[a]			
Item	Quantity	ΔSFC, %	ΔThrust, %
Compressor bleed	1 lb/sec	0.85	1.58
Fan air	1 lb/sec	0.46	0.47
Ram air	1 lb/sec		0.13
Engine shaft power	100 hp	0.4	0.98
APU SFC = 0.5 lb fuel/hr/hp-hr (1.77 kg fuel/sec/kW-sec)			

[a]See figure 11.

TABLE 20.—SECONDARY POWER SYSTEM CRUISE POWER REQUIREMENTS, TOTAL FOR AIRPLANE—MODEL 767-611

Item	Secondary power system concepts			
	Conventional technology		Dedicated APU III	Internal engine generator IV
	Bleed/shaft I	Shaft II		
Air source, lb/min (kg/sec)				
Engine compressor bleed	288 (2.18)			
Engine fan bleed		288 (2.18)		
Ram air			288 (2.18)	288 (2.18)
SFC, %	4.09	2.21	–	–
Thrust, %	7.59	2.26	–	–
Drag, %	–	–	0.624	0.624
Engine shaft power, hp (kW)				
Hydraulic system	60 (44.7)	60 (44.7)	–	–
Electrical system				
Electrical	242 (180.5)	242 (180.5)	242 (180.5)	217 (162)
Hydraulic	–	–	29 (21.6)	80 (59.6)
Pneumatic	–	–	–	572 (427)
Pheumatic system	–	365 (272)	–	
Total, hp (kW)	302 (225.2)	667 (497.2)	271 (202.1)	869 (648.6)
SFC,%	1.21	2.67	1.08	3.48
Thrust, %	2.96	6.54	2.66	8.50
APU shaft power[a]				
Hydraulic, hp (kW)	Not used	Not used	40 (298)	Not used
Pneumatic, hp (kW)	during	during	458 (342)	during
Total, hp (kW)	cruise	cruise	498 (372)	cruise
Ram air, lb/min (kg/sec)			236 (1.79)	
SFC,%			1.5	
Drag, %			0.26	
Totals				
SFC, %	5.3	4.88	2.58	3.48
Thrust, %	10.55	8.80	2.66	8.50
Drag, %	–	–	0.88	0.62

[a]APU fuel consumption is converted to equivalent engine fuel SFC increase. APU engine primary air consumption is charged with 50% momentum loss to compensate for installation losses.

The results of this study, however, are valid to support the following conclusions:

- The optimum SPS is very dependent upon the airplane configuration. The development of SPS building blocks should take priority over integrated system developments. The building blocks described in part III of this document should be vigorously pursued to ensure timely development of an advanced technology transport.

- The significant SPS trade items, on a total-airplane basis, are those affecting cruise SFC and cruise drag, as opposed to those items which reduce SPS weight.

- All engine shaft power extraction is preferred over the combined engine compressor bleed/shaft power extraction for the advanced technology engines under consideration at the present time.

- The dedicated APU SPS concept showed a promising payoff which can be further improved by replacing the GSD with an air turbine starter and resizing the electrical system.

- The IEG SPS concept should not be considered for airplane configurations where engine frontal area and resultant cruise drag may be reduced by other means.

PART II

INTRODUCTION

This part of the study is devoted to the investigation and evaluation of the technical and economic payoff of the IEG-powered SPS concept against the conventional, present-day, and competitive advanced SPS concept for the 1975 time period; i.e., 1975 engine program go-ahead resulting in a 1980 airplane certification.

Industry studies on the cycloconverter and dc-link VSCF systems indicated that for the 1975 period the generating capacity should be limited to engine start capability. As a result, an IEG SPS configuration was conceived that would utilize this electrical power capacity and still meet the SPS loads and requirements defined in part I. This configuration was traded against the modified conventional technology (similar to configuration I defined in part I) and the competitive generator starter drive (GSD) SPS configurations.

The GSD is an engine-pad-mounted generator drive which can be used to start the engine and also to generate electrical power. The concept was first proposed for the model 727 airplane, but could not be developed in time to be used. Later development and successful use of the axial gear differential constant speed drive, which is an integral part of the GSD, makes this concept a highly competitive item of moderate risk for the 1975 time period.

For this study the baseline airplane configuration was updated from the model 767-611 to the model 767-620 (fig. 40). This airplane was selected in the ATTP study as having the most potential. The SPS loads and requirements defined in part I are equally applicable to this airplane. The generating capacity required to start the engine (135 kVA per channel) is insufficient to power all SPS loads. A rating of 340 kVA per channel is required to power all SPS loads, as discussed in part I. The pneumatic power system is the same for all three SPS configurations, except for the deletion of the air turbine starter on the GSD and IEG configurations. This eliminated one system from the trade study and thus increased its validity.

The following ground rules were established to focus the study consistent with the man-hours available.

- APU to be a ground-operable unit only.
- IEG (excluding converter) reliability to be greater than 1.6×10^8 sec (50 000 hr) mean time between failures.

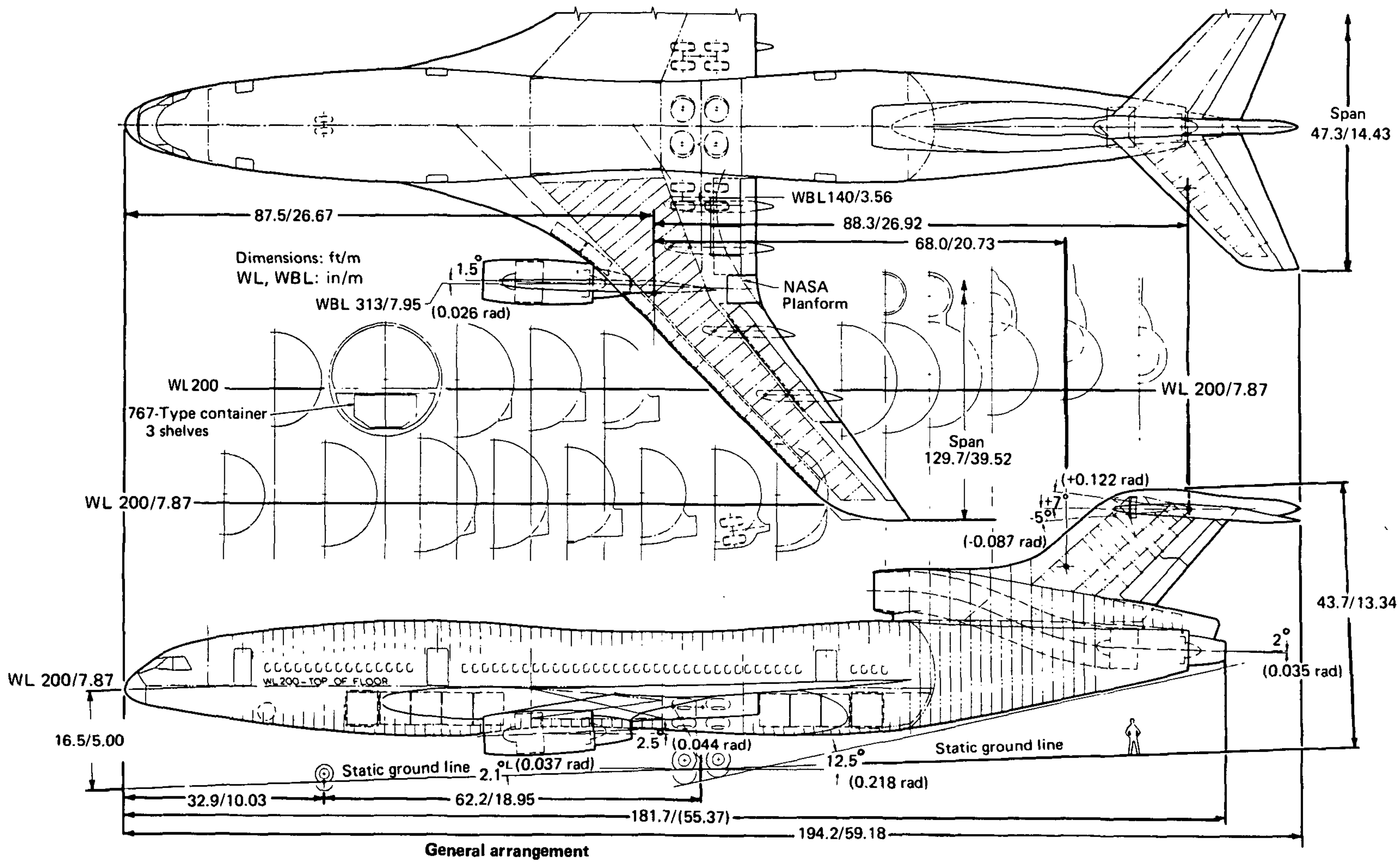


FIGURE 40.–MODEL 767–620

- Conventional-technology system to be designed for one-generator-inoperative dispatch. This dictates the need for a small fourth independent electrical power generating source to satisfy the safety requirement for AFCS (critical flight controls).

- IEG and GSD systems assume all generator/converters to be operative at takeoff. (This equipment is required to start engines.)

- Windmilling power to be sufficient to provide emergency power from propulsion-engine-driven hydraulic pumps.

- Loss of any individual source of power provided to hydraulic pump or loss of any single pump must not result in loss of any hydraulic system.

- Hydraulic system ground checkout to be possible without operating main propulsion engines or APU.

- Pneumatic power (engine compressor bleed air) usage to be limited to environmental systems.

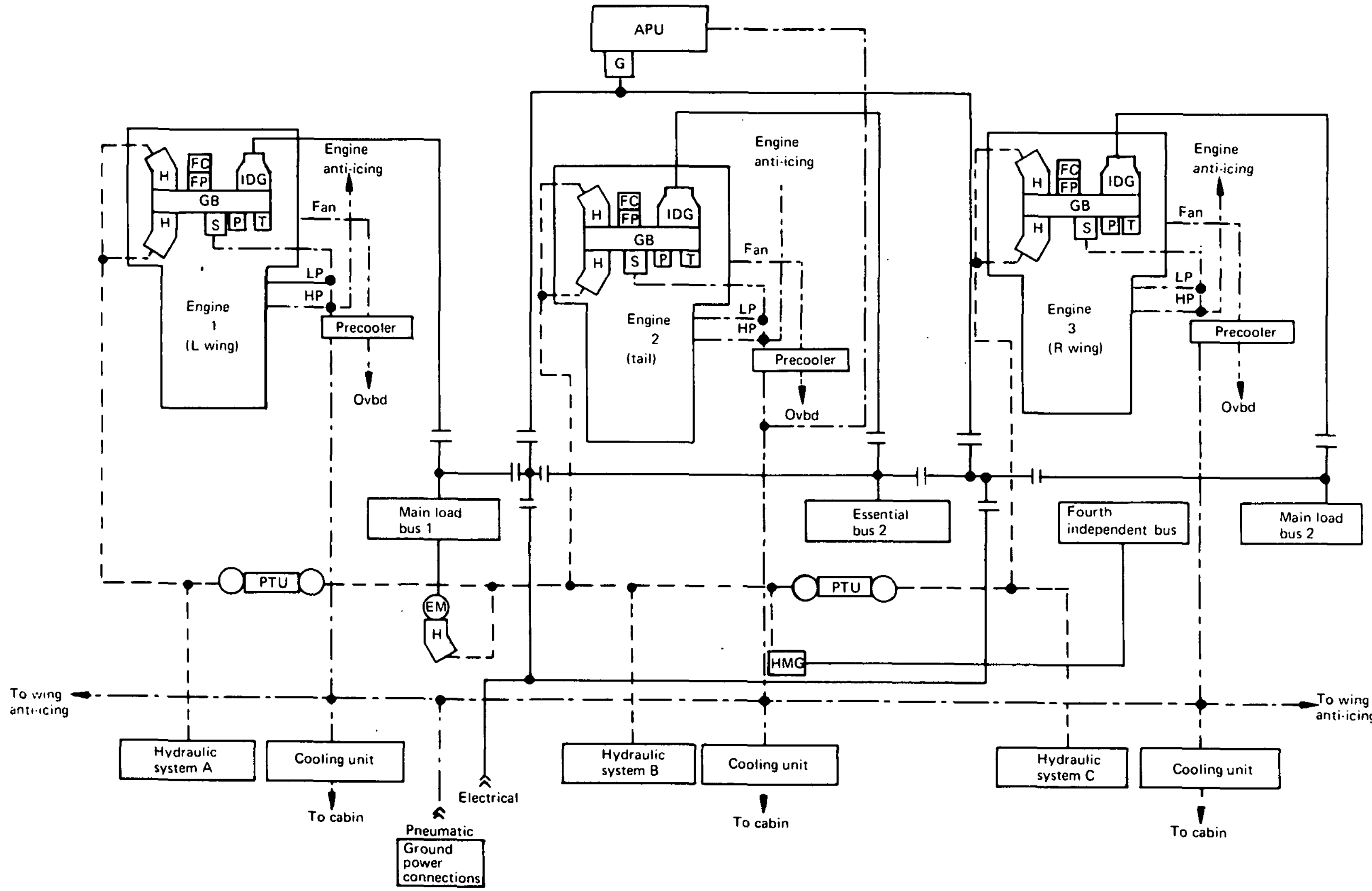
- User system reliability to be assumed equal for all three concepts.

TRADE STUDY

System Description

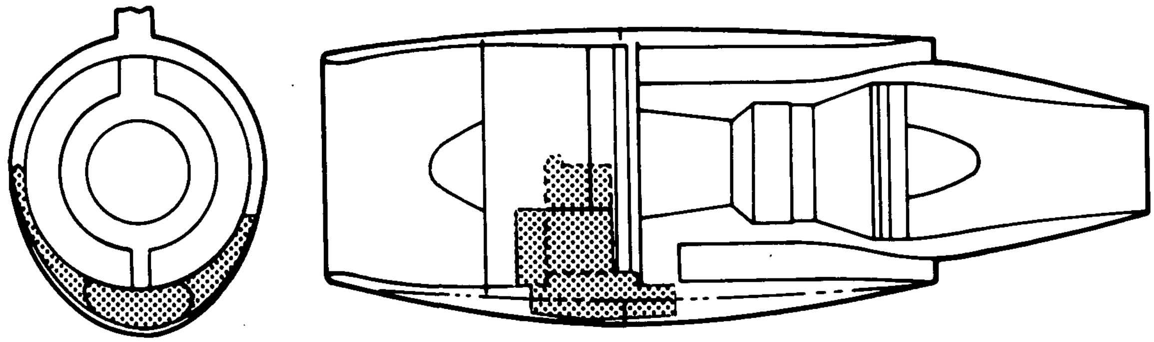
Conventional SPS (Fig. 41)

This configuration utilizes conventional technology components and is essentially the same as configuration I of part I. The major differences are the tailoring of the hydraulic and electrical distribution systems to the airplane configuration and the use of engine-mounted gearboxes to drive the airplane accessories. Two variations were studied. One was a gearbox installation on the lower fan case (chin mount) with the lower fan cowl shaped to provide aerodynamic smoothness over the accessories (fig. 42a). For the other, the gearbox was installed in the lower fan duct bifurcation section. The cowl diameter was increased slightly to account for the added bifurcation frontal area (fig. 42b). A description of each applicable SPS system follows to assist in identifying components shown in the diagrams and discussed in the weight trades.

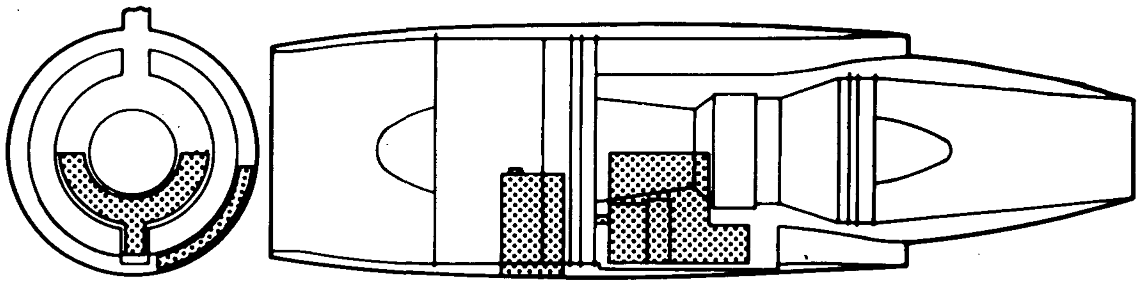


APU	Auxiliary power unit	G	Generator	HP	High-pressure bleed	P	Oil pump
Fan	Fan air bleed	GB	Gearbox	IDG	Integrated drive generator	PTU	Power transfer unit
FC	Fuel control	H	Hydraulic pump	LP	Low-pressure bleed	S	Air turbine starter
FP	Fuel pump	HMG	Hydraulic motor generator	Ovbd	Overboard	T	Tach generator
EM	Electric motor						

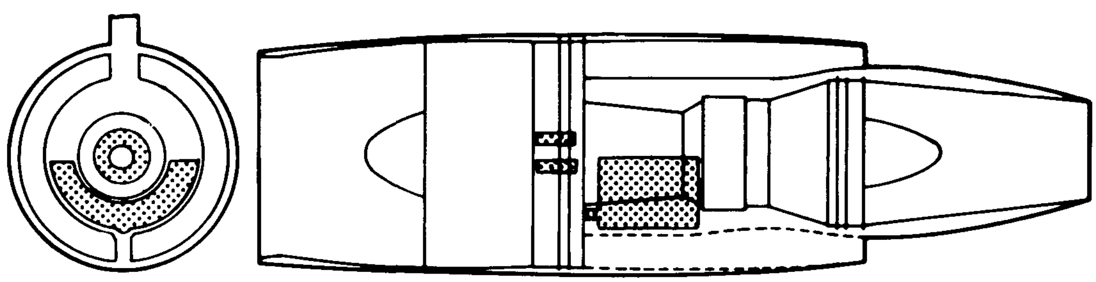
FIGURE 41.—CONVENTIONAL SECONDARY POWER SYSTEM



a. Conventional chin mounted accessories



b. Bifurcation mounted accessories



c. Internal engine generator-austere engine gearbox

Figure	Description	Maximum diameter		Bifurcation width			
				Upper		Lower	
		in	(m)	in	(m)	in	(m)
a	Convention chin mounted	89.0	(2.26)	13.0	(0.30)	6.0	(0.15)
b	Conventional technology	89.0	(2.26)	13.0	(0.30)	13.0	(0.30)
b	Advanced technology	88.0	(2.23)	13.0	(0.30)	13.0	(0.30)
c	Internal engine generator	83.0	(2.11)	13.0	(0.30)	3.0	(0.08)

FIGURE 42.—MODEL 767-620 NACELLE CONFIGURATIONS

Electrical system.—The electrical system is the same as that of configuration I of part I except for the following (see fig. 43):

- Only one HMG is provided to make the part II systems more equivalent for trade study purposes. This unit would operate only when one of the main generators was inoperative.

- Additional contactors were added to provide a greater degree of isolation. Essential bus 4 is normally powered by generator 2 and backed up by the HMG.

Hydraulic system.—The hydraulic functions are the same as those of configuration I of part I. The system arrangement was modified as follows:

- The hydraulic system was changed from cross-connected, engine-driven pumps to a configuration that uses power transfer units (PTU) to facilitate power transfer between systems. PTUs transfer power from an operating system into a system with an inoperative power source without fluid transfer between systems.

- One HMG was deleted (see electrical description, above) because the remaining HMG in system B can be powered by a PTU from either system A or C in the event of loss of the center engine.

- Each system has two 2.33×10^{-3} m^3/s (37 gpm) engine-driven pumps. System B has one 6.3×10^{-4} m^3/s (10 gpm) pump for system checkout. The PTUs allow checkout of systems A and C.

Generator/Starter Drive SPS (Fig. 44)

This configuration utilizes advanced technology engine accessories (e.g., electronic fuel control, high-speed oil pump), GSD, and hydraulic pumps. These accessories are installed on lower fan duct bifurcation mounted gearboxes (fig. 42b).

Electrical system.—The generator/starter drive configuration is similar to that shown in part I except for the APU generators. The bus arrangement and generator ratings are shown in figure 45. Each generator can furnish approximately two-thirds of the total electrical load and can assume the loads of both its own bus and the bus of a failed generator. Standby and dc systems are the same as those shown in figure 43.

Hydraulic system.—Each hydraulic system has one engine-driven and one electrically driven hydraulic pump. This arrangement precludes the need for an emergency power unit, which saves

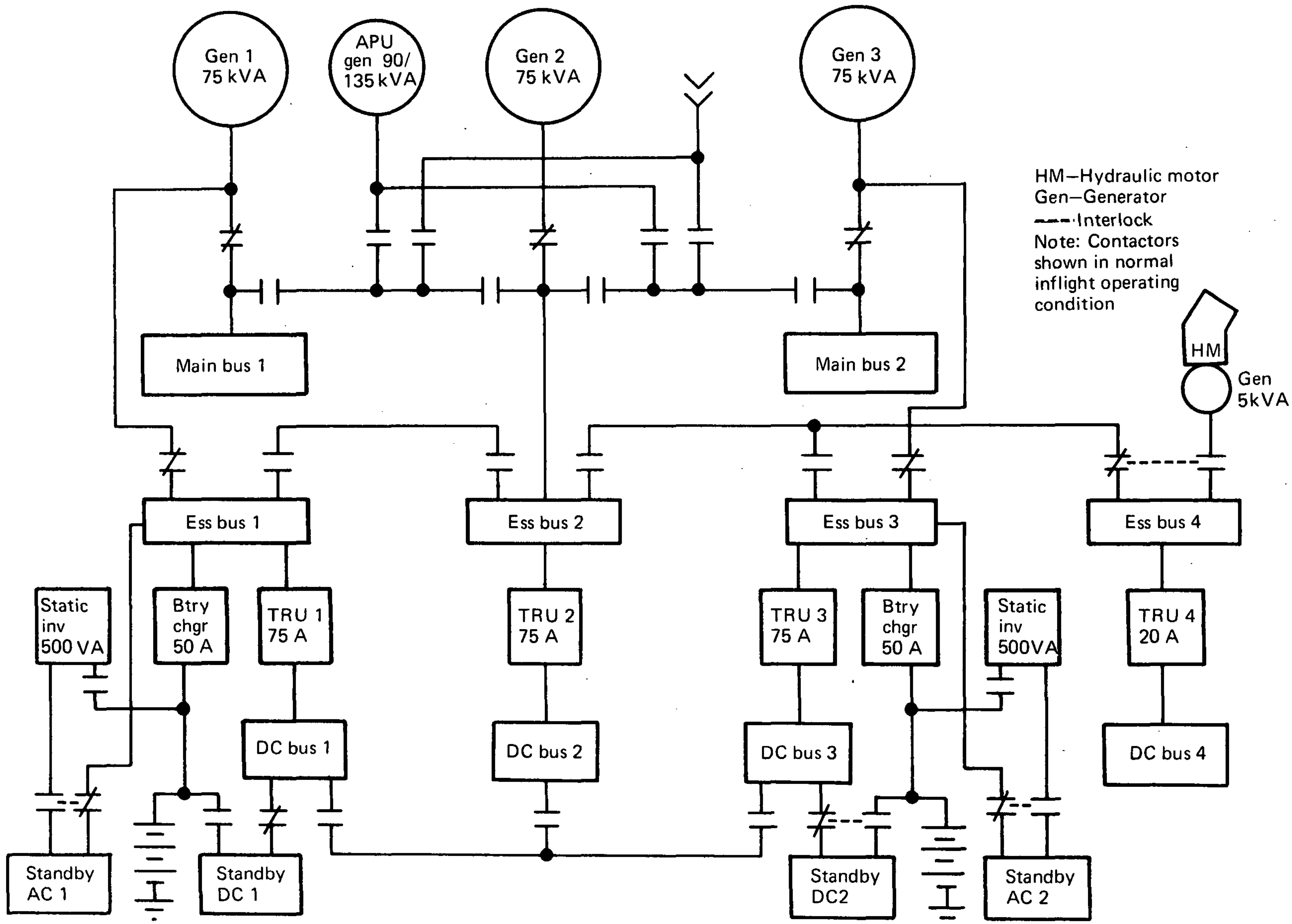
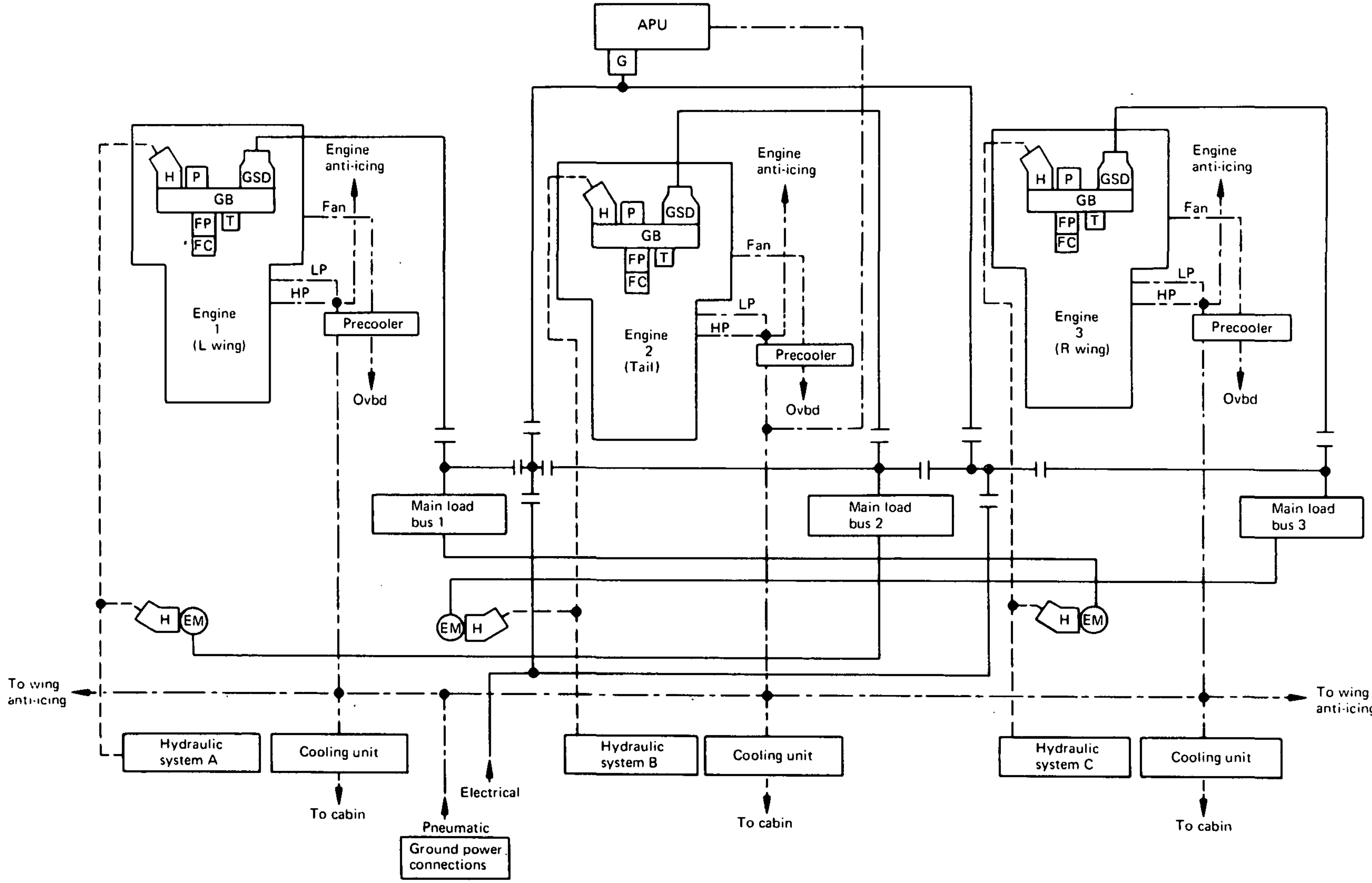


FIGURE 43.—ELECTRICAL SYSTEM DIAGRAM—CONVENTIONAL CONFIGURATION



APU	Auxiliary power unit	GSD	Generator/starter drive	P	Oil pump
EM	Electric motor	H	Hydraulic pump	FP	Fuel pump
Fan	Fan air bleed	HP	High-pressure bleed	FC	Fuel control
G	Electric generator	LP	Low-pressure bleed	T	Tach generator
GB	Gearbox	Ovbd	Overboard		

FIGURE 44.—GENERATOR/STARTER DRIVE SECONDARY POWER SYSTEM

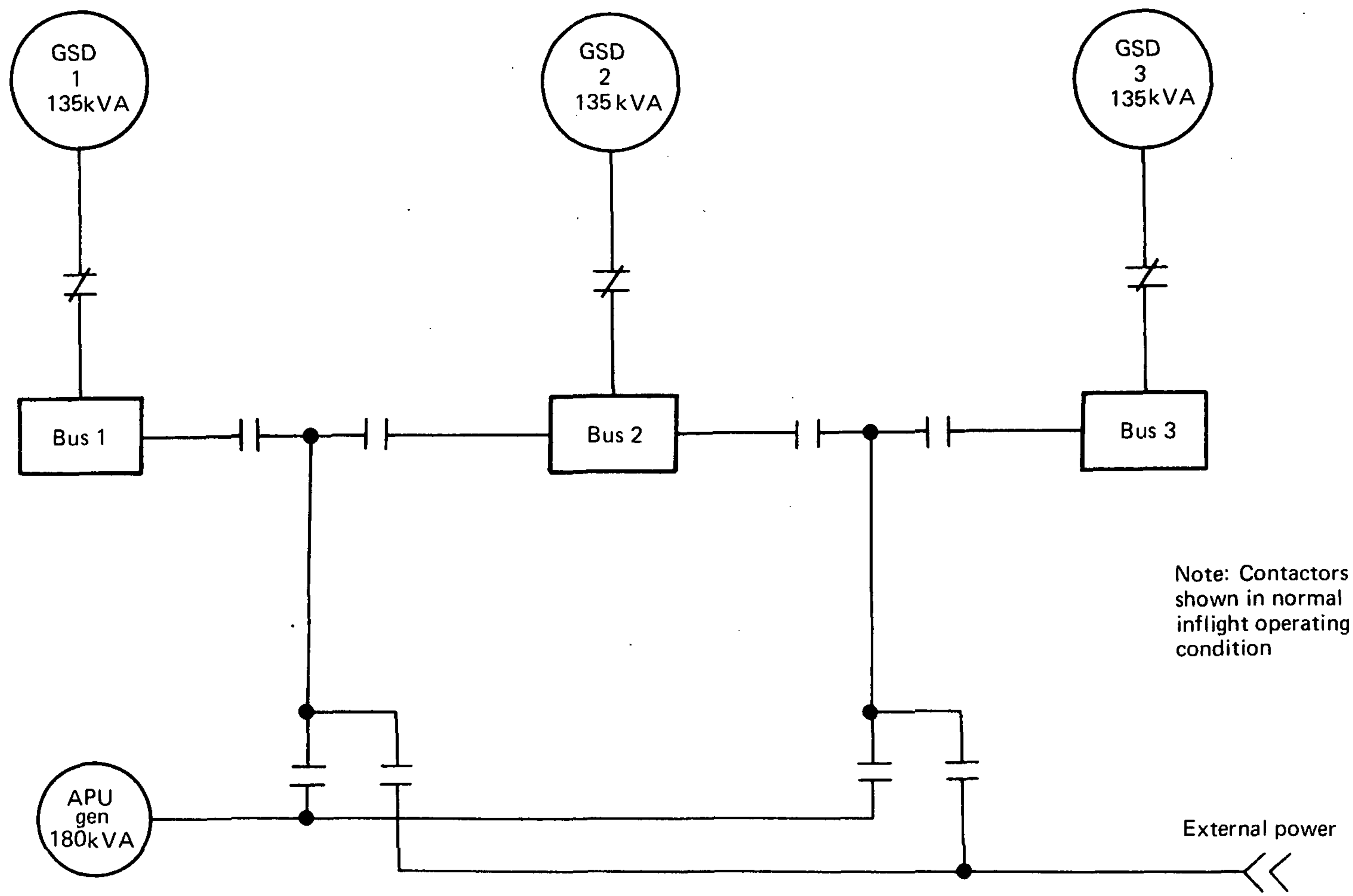


FIGURE 45.—SIMPLIFIED DIAGRAM—GENERATOR/STARTER DRIVE CONFIGURATION

56.6 kg (125 lb). The maximum capacity of each pump is 2.02×10^{-3} m^3/s (32 gpm). The electric motors are connected to the load buses so that loss of one propulsion engine will not cause the loss of one hydraulic system. Also, the power to the electric motor can, in most cases, be restored by switching to a different load bus. The gear retraction load requirement on takeoff is met by the electric-motor-driven pump in the event that the center engine or its pump fails. The hydraulic system flow demands and pump capacities are shown in figure 46.

Internal Engine Generator SPS (Fig. 47)

This system is similar to that of configuration IV of part I except that the IEG is sized for engine starting (135-kVA versus 340-kVA ratings). This capacity in the generating mode provides sufficient power for all normal hydraulic and electric loads. Both cycloconverter and dc-link versions of the IEG were used.

Electrical system.—For both VSCF systems an output rating of 135 kVA resulted when the generator and converter were sized for engine starting. This correlated closely with the 140-kVA rating calculated by NASA/Lewis. This correlation establishes a high degree of confidence in the information used in the evaluation. The data used in this study were provided by electrical equipment suppliers and are shown in table 21. The load bus arrangement is shown in figure 48.

TABLE 21.—PARAMETERS USED FOR IEG STUDIES

Parameter	Generator		Converter		Feeders	
	Cycloconverter	DC link	Cycloconverter	DC link	Cycloconverter	DC link
Continuous rating, kVA	175	150	135	135	Sized for application	
Weight, lb (kg)	175 (79.4)	165 (72.5)	95 (43)	155 (70.3)		
lb/ft (kg/m)					2.0 (2.97)	1.5 (2.23)
Diameter, in. (m)	13 (0.33)					
Lenght, in. (m) (includes mounting)	16 (0.406)					
Volume, cu in. (cu m)			2000 (0.033)			
Losses, kW	23	23	0	11	3	
Cooling	Oil	Oil	Air or cold plate			

- Generator design—To meet the extended MTBF required, it was necessary to increase the copper and iron in the generator. This reduces internal temperatures and extends generator insulation life. Based on absence of bearings and seals, conservative design of generator electromagnetics, and on test data on rotating rectifiers that have operated at gravitational forces greater than those to be encountered in the IEG, 1.8×10^8 to 3.6×10^8 sec (50 000 to 100 000 hr) MTBF is feasible.

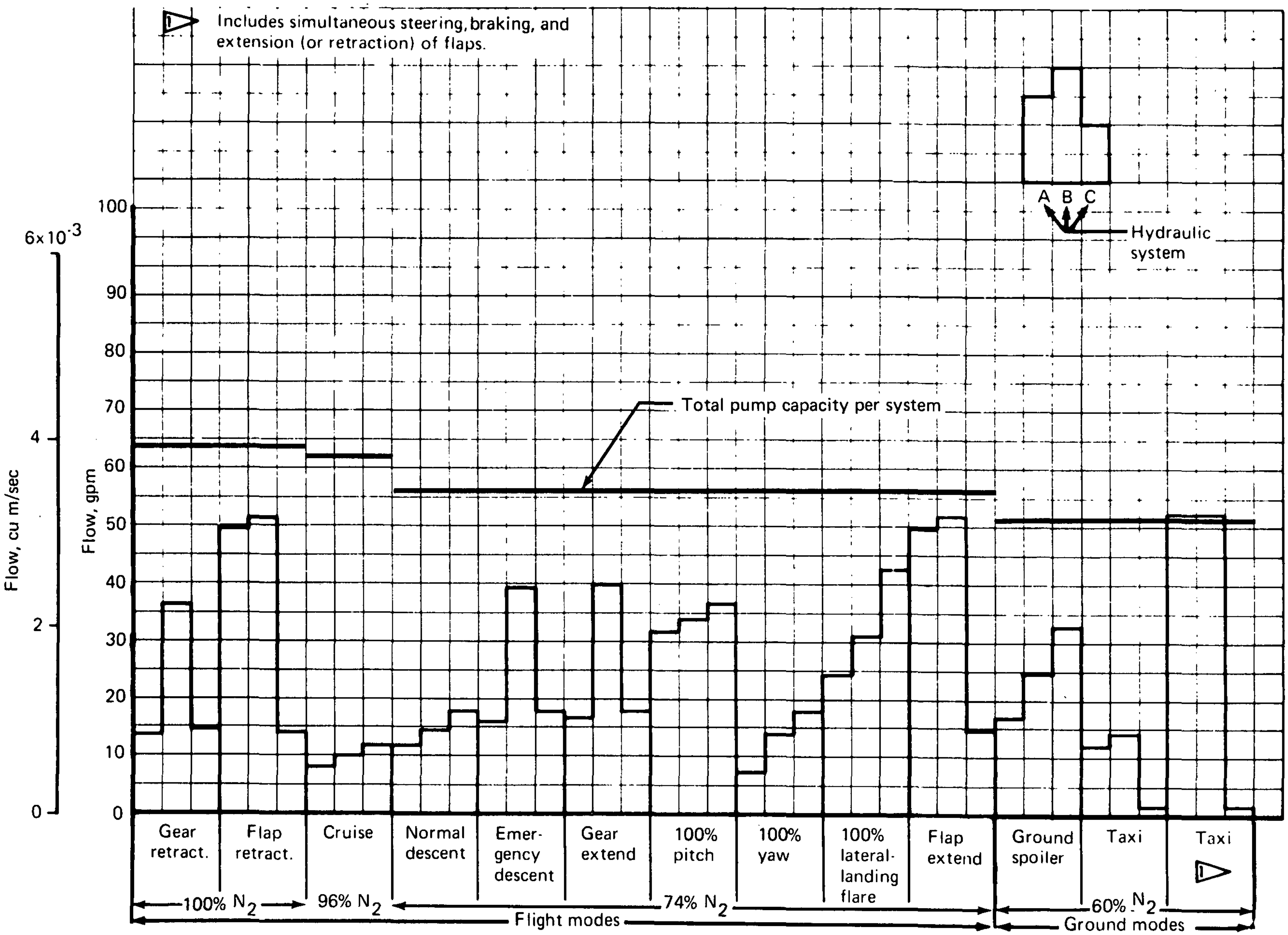
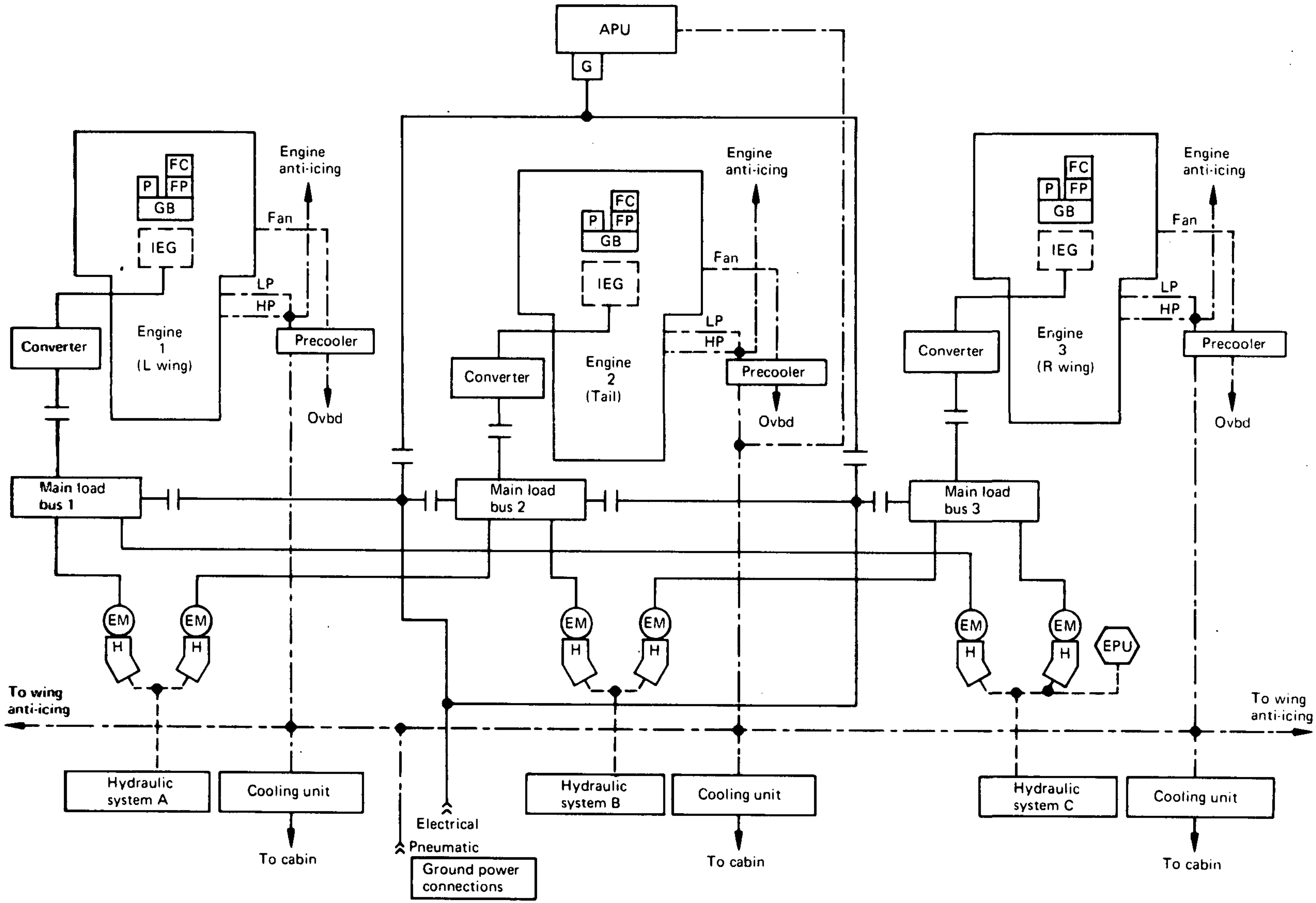


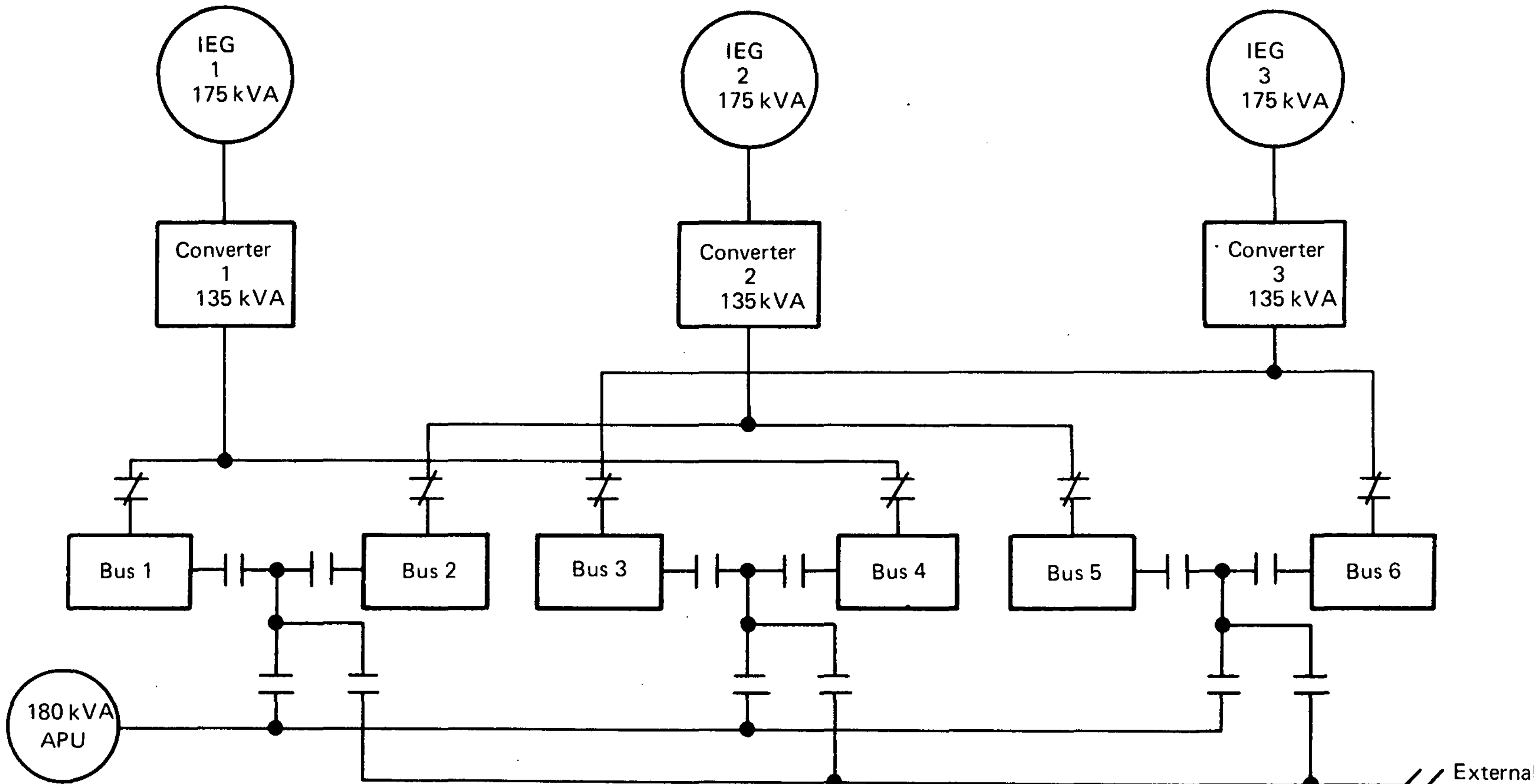
FIGURE 46.–HYDRAULIC FLOW DEMAND AND PUMP CAPACITY–GSD CONFIGURATION



APU	Auxiliary power unit	G	Electrical generator	LP	Low-pressure bleed
EM	Electric motor	GB	Gearbox	Ovbd	Overboard
Fan	Fan air bleed	H	Hydraulic pump	P	Oil pump
FP	Fuel pump	HP	High-pressure bleed	EPU	Emergency power unit
FC	Fuel control	IEG	Internal engine generator		

Note: DC, standby, and essential systems not shown.

FIGURE 47.—INTERNAL ENGINE GENERATOR SECONDARY POWER SYSTEM



Generator out	Buses picked up by remaining generators			Buses not energized
	Gen 1	Gen 2	Gen 3	
None	1 & 4	2 & 5	3 & 6	–
1	–	1,2,5	3,4,6	–
2	1,2,4	–	3,5,6	–
3	1,3,4	2,5,6	–	–
1 & 2	–	–	3,4,5,6	1 & 2
2 & 3	1,2,3,4	–	–	5 & 6
1 & 3	–	1,2,5,6	–	3 & 4

Note: Contactors shown in normal inflight operating condition

FIGURE 48.—SIMPLIFIED DIAGRAM—CYCLOCONVERTER IEG CONFIGURATION

- Principle of operation (generating mode)—The cycloconverter VSCF generator for this application is a nine-phase machine. Over the normal operating speed range, output frequency to the converter is 1200 to 2400 Hz. Six silicon controlled rectifiers (SCRs) are connected to each generator phase and are programmed to switch portions of the high-frequency power to the output in a sequence that approximates a constant-frequency (400-Hz) sine wave output. Each of the three output phases is regulated individually and filtered to obtain required power quality.

  Basic principles of operation of the dc-link approach were discussed in part I.

  Operation of VSCF systems in the starting mode is similar for both dc-link and cycloconverter approaches.

  To provide synchronous motor torque to the engine, multiphase armature currents of proper phase with respect to the rotor must flow in the stator windings, thereby causing a rotating field which reacts with the field flux of the salient pole machine. The torque is proportional to the product of the rotor magnetic field and armature flux linkages and is also dependent on the angular relationship between the rotating mmf wave and the position of the field pole (i.e., the power angle). The most efficient motor operation is obtained with constant field flux held near the saturation level with the torque controlled by the armature current control.

  At very low speeds, the armature current is obtained with very little voltage, since there is little back emf in the motor. The low-frequency alternating current is obtained by normal converter action (stepdown frequency changer) with commutation of the SCRs from the 400-Hz supply. The intelligence to program the armature currents in proper phase relationship to the rotor will most likely be provided by a shaft position indicator. As the speed of the motor increases, a larger voltage is automatically programmed to the motor to overcome the increasing back emf. Armature current (and hence torque) is thus held approximately constant by current limit circuitry as the motor accelerates. At higher motor speeds, the back emf assists with SCR commutation, and a gradual shift in commutation mode takes place. Finally, at high speeds (near engine-idle speed), commutation is supplied almost entirely from the back emf of the motor. This gradual transition in commutation mode occurs smoothly without loss of torque at any condition.

  The exact torque speed characteristics of the starting system can be tailored to provide optimum starting. The maximum torque available at high speeds is somewhat reduced over that available at low speeds, but with self-assist of the engine after lightoff, very fast starting should be achievable.

The dc-link VSCF approach differs from the cycloconverter principally in that power from the generator is first rectified to dc.

When operated in the starting mode, the dc-link system must be "turned around" so that the power source (ground cart or APU) is feeding into the rectifier section. The diagram of figure 31 indicates the switching that must be provided so that the dc-link converter can convert the input power voltage and frequency required by the generator when operated as a motor. At this time, operation of the generator as a synchronous motor from zero speed is not considered by the equipment supplier to be the best approach for a dc-link system. Operation as an induction motor is the preferred scheme; however, this involves a means to short the main field windings during the starting sequence, which adds complexity to the generator. A mechanical (centrifugal) switch could be the simplest means to accomplish field shorting, or solid-state components operated by magnetic coupling might be used. In either case, components are added which increase the problem of achieving the high level of reliability required of the generator.

The cycloconverter VSCF can conduct power in both directions and can operate in the engine starting mode without undue complications. No special "start" bus switching arrangements are required, thus permitting an uncomplicated bus and bus protection arrangement. During engine starting, 400-Hz power is supplied to the constant-frequency terminals of the converter from either an external source or the onboard APU. The 400-Hz power is converted to the variable frequency and variable voltage required by the generator operating as a motor.

Hydraulic system.—The hydraulic system is the same as that of configuration IV of part I.

System Weight Comparison

SPS weights were generated for the model 767-620 airplane. These weights were modified to meet the particular requirements of the selected SPS configurations with respect to equipment placement, power supply, equipment size, and degree of component integration. The weights shown in table 22 reflect delta changes between configurations and do not show overall system weight for any one system. Pertinent items are discussed below.

Propulsion Engine Gearbox

The engine gearbox weight summary is shown in table 23. Significant differences existing between the SPS configurations are in the propulsion engine gearbox arrangement, location, and size for the traded configurations.

TABLE 22.–SECONDARY POWER SYSTEM WEIGHTS[a]–MODEL 767-620

Item	Weight			
	System		Delta	
	lb	(kg)	lb	(kg)
Conventional technology				
Hydraulic	1081	(490)		
Electrical	876	(397)		
Pneumatic	Base	(Base)		
Fan cowl[b]				
Chin mount	Base	(Base)		
Bifurcation mount	120	(54.4)		
Gearbox	Base	(Base)		
Total				
Chin mount	1957	(887)	Base	(Base)
Bifurcation Mount	2077	(941.4)	120	(54.4)
Generator starter				
Hydraulic	1055	(478.0)		
Electrical	1430	(648.3)		
Pneumatic	-170	(-77.0)		
Fan cowl[b]	74	(33.5)		
Gearbox	-165	(-74.8)		
Total	2224	(1008.0)	267	(121)
Internal engine generator				
Hydraulic	1061	(481.5)		
Electrical				
Cycloconverter	1902	(862)		
DC link	1952	(884)		
Pneumatic	-170	(-77.0)		
Fan cowl[b]	-318	(-144)		
Gearbox	-450	(-250)		
Total				
Cycloconverter	2025	(918)	68	(31)
DC link	2075	(941.5)	118	(53.5)

[a]Total for airplane does not include components common to all systems.

[b]Weight for two wing-mounted engines and tail engine assumed to be same for all concepts.

TABLE 23.–GEARBOX WEIGHT COMPARISON–MODEL 767-620

Item	Secondary power system concept							
	Conventional technology				Engine pad-mounted generator/ starter		Internal engine generator	
	Chin mount Installation		Bifurcation mount installation					
Gearbox location	Lower fan case		Lower fan duct bifurcation		Lower fan duct bifurcation		Lower primary core	
	lb	(kg)	lb	(kg)	lb	(kg)	lb	(kg)
Gearbox	750	(339.5)	750	(339.5)	600	(272.0)	345	(156.3)
Copler	30	(13.6)	30	(13.6)	30	(13.6)	30	(13.6)
Tower shaft	75	(34.0)	75	(34.0)	60	(27.2)	30	(13.6)
Total for airplane	855	(387.1)	855	(387.1)	690	(312.8)	405	(183.5)
Delta weight	Base		0	0	-165	(-74.3)	-450	(-203.6)

Electrical System

The electrical system weight is shown in table 24 for components that differ between configurations. Weights are listed for two versions of the IEG; the cycloconverter type and the dc-link type. Generator feeder data are for aluminum conductors (spliced to short lengths of copper in the engine area) for all configurations except the cycloconverter version of the IEG, which are all copper in seven-conductor shielded bundles. Aluminum cables for the cycloconverter VSCF system are under investigation and may be ready for application by 1975; however, insufficient information is available now to provide accurate weight estimates.

Hydraulic System

The hydraulic system weight comparison is shown in table 25. The major difference in the configurations from those of part I is caused by the change in baseline airplane.

The reservoirs for systems A and C of the conventional SPS configuration were located in the wheel wells for space consideration. This resulted in a significant weight penalty because of the long suction lines.

The GSD distribution system is essentially similar to the conventional system except for the changes required to accommodate the large electric-motor-driven pumps. In systems A and C the electric motor pumps and reservoirs are in the wheel well; for system B they are located in the tail.

TABLE 24.—ELECTRICAL SYSTEM WEIGHT SUMMARY—MODEL 767-620

Item	Secondary power system concepts							
	Conventional technology		Generator/starter drive		Internal engine generator			
					Cycloconverter		DC link	
	lb	(kg)	lb	(kg)	lb	(kg)	lb	(kg)
Integrated drive generator	282	(128)						
Generator/starter drive with mounting			636	(288)				
Internal engine generator (IEG)					525	(238)	495	(224)
Converter					285	(129.3)	465	(211)
Generator control unit and CTs	31.5	(14.3)	37.5	(17)			21	(9.5)
DPCTs and wiring	6.0	(2.7)	6	(2.7)	6	(2.7)	12	(5.4)
Generator control wiring	28.0	(12.7)	29	(13.1)	31	(14.0)	31	(14)
Heat exchanger (plumbing, oil, etc.)	108.0	(49.0)	126	(57)	120	(54.4)	108	(49)
Generator, feeders	117.0	(53.0)	207	(94)	417	(189.2)	261	(118)
Rack cooling provisions	—	—	—	—	38	(17.2)	38	(17.3)
Installation—connectors	58.0	(26.3)	69	(31.3)	130	(59.0)	127	(57.5)
APU generator and control	73.0	(33.1)	140	(63.5)	140	(63.5)	140	(63.5)
APU feeders	86.0	(39.0)	128	(58.0)	128	(58.0)	128	(58.0)
Contactors	39.3	(17.8)	44	(20.0)	72	(32.6)	116	(52.6)
Hydraulic motor generator and controls	39.0	(17.7)	—	—	—	—	—	—
Bus protection panel	9.0	(4.1)	8	(3.6)	10	(4.5)	10	(4.5)
Totals[a]	876.8	(397.7)	1430.5	(648.2)	1902	(862.7)	1952	(885.3)
Delta weight	Base		553.7	(250.9)	1025.2	(464.9)	1075.2	(487.6)

[a]Total for airplane does not include components common to all systems.

TABLE 25.—HYDRAULIC SYSTEMS WEIGHT SUMMARY—MODEL 767-620

Item	Secondary power system concept					
	Conventional technology[a]		Generator/starter drive		Internal engine generator[a]	
	lb	(kg)	lb	(kg)	lb	(kg)
Distribution system	759	(344)	594	(269)	255	(116)
Power generation[b]	222	(101)	461	(209)	699	(318)
Power transfer units	100	(45)	—	—	—	—
Emergency power unit[c]	—	—	—	—	107	(48)
Total[d]	1081	(490)	1055	(478)	1061	(482)
Delta weight	Base		−26	(−12)	−20	(−9)

[a]Weights applicable to both variations of this configuration
[b]Pumps, electrical feeders, and motors
[c]Hydrazine unit with fuel supply
[d]Total for airplane does not include components common to all systems

The general layout of the pressure line distribution system for the IEG configuration is shown in figure 49. All hydraulic pumps, reservoirs, and suction lines are within the hydraulic load center located in the wheel well.

Pneumatic System

The pneumatic system weight differences are also shown in table 22 and reflect the deletion of the pneumatic starting system on the GSD and IEG configurations.

Engine Fan Cowl

The fan cowl weight differences shown in table 26 reflect the modifications required to accommodate the various accessory installations.

TABLE 26.—FAN COWL WEIGHT COMPARISON (PER ENGINE)—MODEL 767-620

Item	Secondary power system concept							
	Conventional technology				Engine pad-mounted generator/starter		Internal engine generator	
	Chin mount installation		Bifurcation mount installation					
	lb	(kg)	lb	(kg)	lb	(kg)	lb	(kg)
Fan cowl	1409	(639)	1391	(631)	1368	(620.5)	1231	(558.1)
Actuator	—	—	70	(31.7)	70	(31.7)	20	(9.1)
Bifurcation structure	2	(0.9)	10	(4.5)	10	(4.5)	1	(0.5)
Total	1411	(639.9)	1471	(667.2)	1448	(656.7)	1252	(567.7)
Delta weight	Base	(Base)	+60	(+27.3)	+37	(+16.8)	−159	(−72.2)

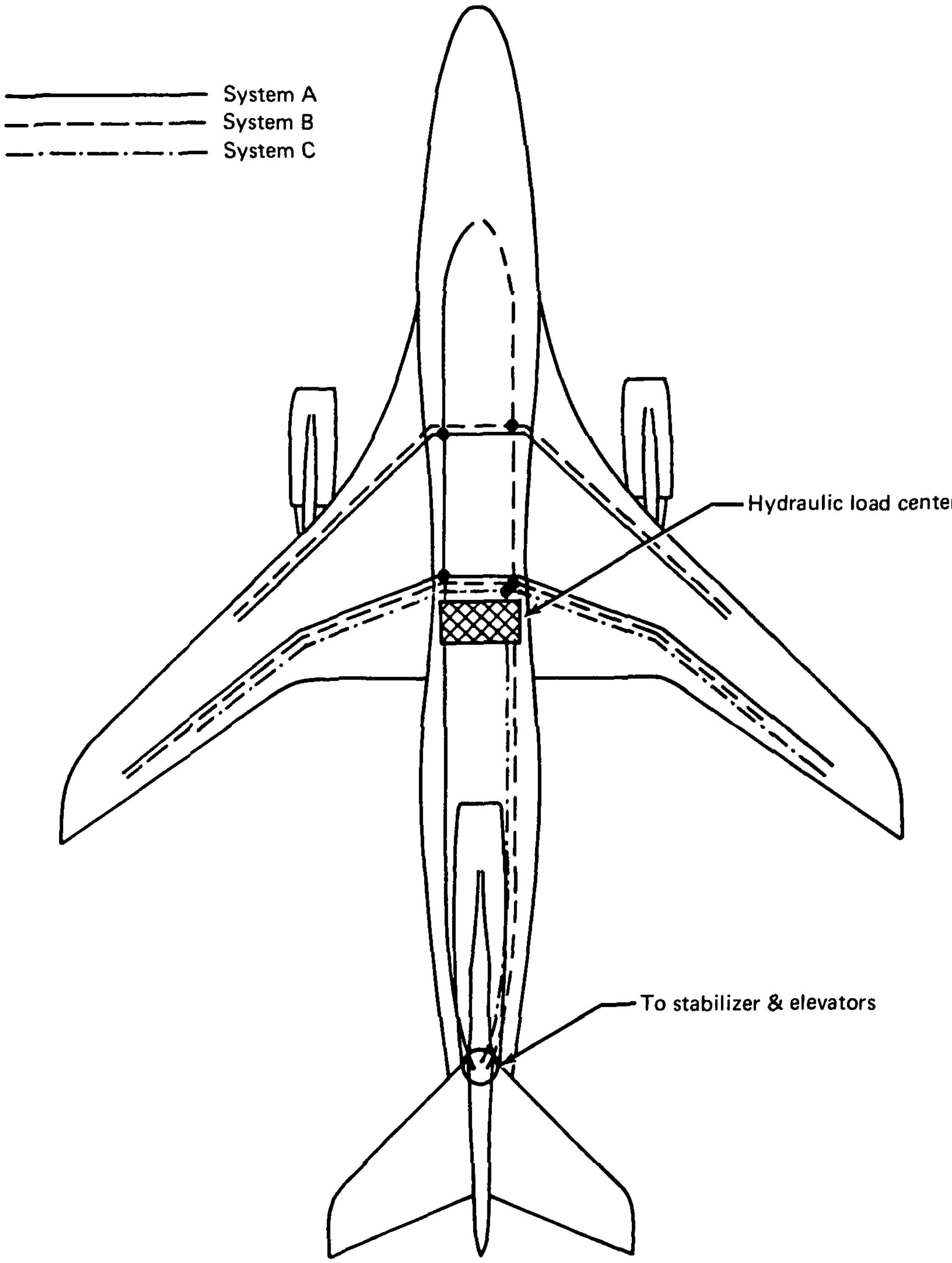


FIGURE 49.—PRESSURE LINE SCHEMATIC—IEG CONFIGURATION

Reliability Analysis

General

The reliability of the three configurations was considered from the standpoint of (1) engine starting function, (2) various success criteria of power generation, and (3) dispatch reliability.

In addition to the basic ground rules, the following requirements related to the reliability analysis were added:

- Since the APU is similar in all three configurations and is not operated in flight, it was eliminated from the reliability study.
- The fourth independent electrical power source on the conventional technology configuration was eliminated from the power generation reliability study. In the dispatch reliability study it was considered because of its significant effect.
- A flight time of 3600 sec (1 hr) was assumed for all configurations.
- All components were assumed to be operable in all configurations at airplane dispatch, i.e., at the beginning of each flight, for the power generation study.
- Multiple system losses due to crew error or other reasons outside of equipment failure were not considered in this study.

The reliability analysis results are summarized in table 27.

Discussion

The GSD and IEG components are used both to start engines and to generate electrical power. Thus, the analysis of all three configurations considers both these functions. A third function, that of switching from starting to generating, has been combined with the engine starting function. The total start-generate reliability is a simple series combination of the two individual reliabilities as illustrated below:

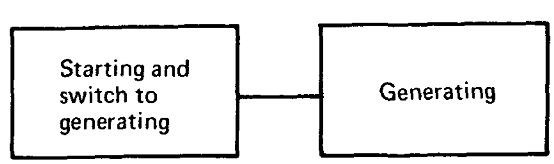


TABLE 27.—SECONDARY POWER SYSTEM RELIABILITY RATING —MODEL 767-620 AIRPLANE BASELINE

Item	System concept			
	Conventional technology[a]	Engine pad-mounted generator/ starter	Internal engine generator[a]	Actual for model 727
Probability of not starting one of the engines due to failure of the starting system (per engine start)	200×10^{-6}	200×10^{-6}	200×10^{-6}	240×10^{-6}
Once started, probability of experiencing any degradation below 100% electrical ac power on all three buses per 1-hr (3600-sec) flight	750×10^{-6}	900×10^{-6}	870×10^{-6}	
Total start-generate mission using above generate criteria (sum of above) per 1-hr (3600-sec) flight	950×10^{-6}	1100×10^{-6}	1070×10^{-6}	
Once started, probability of experiencing any degradation below 100% hydraulic power on all three systems per 1-hr (3600 sec) flight	1600×10^{-6}	1700×10^{-6}	1500×10^{-6}	

[a]Applicable to both variations

Starting and switching functional reliability.—The conventional technology configuration has a pneumatic starter system which is completely separate from the electrical generating equipment. Pneumatic starter system problems were reviewed to determine the potential reliability of the conventional technology configuration. The resulting predictions further dictated the allowable failure rates for the starting hardware on the other two configurations when used in their starting modes of operation.

About two-thirds of the starting problems identified in the study directly involve the pneumatic starting system. The starter itself is only a small percentage of this.

A realistic goal for the pneumatic starting system for any new airplane is a failure rate of about $200/10^6$ starts. This rate applies to the conventional technology configuration as well as to the starting hardware of the other two configurations during their starting modes of operation. The goal of $200/10^6$ starts is judged to be realistic for the GSD, based on airline experience with similar components.

The IEG system encompasses two components during the starting function, i.e., the generator and converter. The converter was allocated a $190/10^6$ failure rate, leaving $10/10^6$ for the generator during the start function. Note that a rate of $10/10^6$ is equivalent to 100 000 mean starts between failures. If these rates can be achieved on the IEG configuration, then the starting functional reliability is in balance with the other two configurations. It may be difficult to achieve the rate of $10/10^6$ for the generator, and if so, the potential starting reliability of this configuration would not be considered on a par with the other two configurations.

Generating functional reliability.—Nine different success criteria were identified and an analysis of each was made to establish the probability of success. These are identified in table 28.

TABLE 28.—SUMMARY OF RELIABILITY ANALYSIS RESULTS—MODEL 767-620 AIRPLANE BASELINE

Per-Flight Probability (After Starting) of Experiencing Any Degradation Below That Defined as System Success Criterion

System success criteria	Secondary power system concept		
	Conventional technology[a]	Engine pad-mounted generator/ starter	Internal engine generator[a,b]
1. 100% ac power on all three buses	0.75×10^{-3}	0.90×10^{-3}	0.87×10^{-3} (0.95×10^{-3})
2. 100% ac power on one bus and no less than 1/2 loss to other two (two buses share one generator)	0.32×10^{-5}	0.33×10^{-5}	0.33×10^{-5} (0.33×10^{-5})
3. 100% ac power on at least two of three buses	0.19×10^{-6}	0.27×10^{-6}	0.25×10^{-6} (0.30×10^{-6})
4. 100% ac power on at least one of three buses	0.16×10^{-10}	0.27×10^{-10}	0.25×10^{-10} (0.32×10^{-10})
5. No less than 1/2 ac power loss on bus 1 (one specific bus)	0.10×10^{-5}	0.10×10^{-5}	0.10×10^{-5} (0.10×10^{-5})
6. 100% hydraulic power capability on all three systems	0.16×10^{-2}	0.17×10^{-2}	0.15×10^{-2}
7. 100% hydraulic capability on two systems and no less than 1/2 loss to remaining system	0.30×10^{-3}	0.30×10^{-3}	0.30×10^{-3}
8. 100% hydraulic capability on at least two of three systems	0.81×10^{-6}	0.94×10^{-6}	[c]3.7×10^{-6}
9. 100% hydraulic capability on at least one of three systems	0.14×10^{-9}	0.18×10^{-9}	[c]1.6×10^{-9}

[a]Applicable to both variations.

[b]The values in parentheses are the results using an IEG failure rate of $35/10^6$ instead of $10/10^6$.

[c]Bus failure rate chosen significantly affects results.

The computation was accomplished using predeveloped computer programming. The resultant numbers in table 28 refer to the probability of failure to meet the success criteria as defined. The probability is based on a 3600-sec (1-hr) flight.

For ease of consideration, one might think of the probabilities as system failure rates; e.g., 0.75×10^{-3} probability during a single flight of experiencing some type of degradation below that defined. This implies that about 0.75 failures in 1000 flights or 7.5 flights out of every 10 000 will be affected. In this example the majority of these cases will be the need to supply three buses from two generators. This criterion is of concern when three independent electrical sources are desirable.

The first five success criteria listed in table 28 involve the electrical system. The remaining four involve the hydraulic system. After considering the values in the table, it is evident that the conventional technology configuration gets top rating for generating functional reliability. The other two configurations would be rated about equal. This assumes that an IEG component failure rate of $10/10^6$ (100 000 MTBF) can be achieved. If the component failure rate were higher during the generating mode, e.g., $35/10^6$, then the IEG configuration would take a third place in the reliability rating. The values in parentheses in the IEG column of table 28 are the results using the $35/10^6$ rate in place of the $10/10^6$ rate for the IEG component.

The reliability spread for the three configurations, even with the use of higher IEG rate, is not considered overly large, since many of the component rates are only best estimates and their variance can affect the results; e.g., the rate for electrical shorts surrounding the electrical bus itself affects the hydraulic system loss probabilities in the IEG configuration. However, sufficient reliability spread is indicated by the analysis to establish the comparative rating on the three systems.

In conclusion, the generating reliability levels for the three configurations are relatively balanced, with the conventional technology configuration rated best, the GSD next best, and the IEG third.

Dispatch reliability.—The conventional technology configuration has a fourth independent electrical bus which was eliminated from the previous reliability considerations due to its small capacity. This bus gives the conventional technology configuration a significant advantage in dispatch reliability over the other two configurations, since it allows dispatch with one generating system out. For both of the other two configurations, all generating systems must be fully operable for dispatch.

Maintainability Analysis

The maintainability rating of the systems was made on the basis of judgment of potential maintenance costs resulting from premature component removals, component accessibility, remove and replace capability, and the need for fault isolation. A weighting factor established the relative importance of each consideration from a maintainability standpoint. The results are summarized in table 29. Typical premature removal rates used to judge the systems are shown in table 30.

TABLE 29.—SECONDARY POWER SYSTEM MAINTAINABILITY RATING —MODEL 767-620 AIRPLANE BASELINE

Item	System concept			
	Conventional technology[a]	Engine pad-mounted generator/starter	Internal engine generator[a]	Relative weighting[b]
Premature removal rate	1.0	1.3	1.5	20
Accessibility	1.0	0.9	0.6	30
Remove/replace capability	1.0	0.8	0.7	35
Need for fault isolation capability	1.0	1.1	1.2	15
Weighted average rating[c]	1.0	0.98	0.9	

[a]Applicable to both variations.

[b]Based on a total of 100.

[c]Relative to conventional technology concept, values less than 1 require more maintenance.

TABLE 30.—PREMATURE REMOVAL RATES PERTAINING TO ENGINE STARTING AND SECONDARY POWER[a]—MODEL 767-620 AIRPLANE BASELINE

Major system components	Basic component removal rate	Secondary power system concept		
		Conventional technology[b]	Engine pad-mounted generator/starter	Internal engine generator[b]
IDG and associated parts	0.25	0.75		
Engine-driven hydraulic pumps	0.50	3.00	1.50	
Electrically driven hydraulic pump	0.40	[c]0.20	1.20	2.40
Power transfer units	[c]0.20	0.40		
Hydraulic motor generator	0.40	[c]0.20		
Pneumatic starter	0.15	0.45		
Starter valve	0.50	1.50		
GSD and associated parts	0.60		1.80	
IEG	0.02			0.06
IEG converter	0.30			0.90
Emergency power unit	0.40			[c]0.20
Total		6.50	4.50	3.56
Factor[d]		1	1.3	1.5

[a]Removal rate based on 1000 flight hours (3.6 x 10^6 flight seconds)

[b]Applicable to both variations

[c]*One-half rate for part time operation*

[d]Greater than 1 is improvement (range of 0 to 2)

The conventional technology configuration has the best overall maintainability design, primarily because of the lowest component weights, easy access, and most lenient deferred maintenance plan. The GSD is considered less maintainable than the conventional technology configuration because the GSD is a heavy engine-mounted component. There is limited deferred maintenance capability, and there is a high dependency on the APU for starting electrical power. The IEG configuration is considered to have the lowest overall maintainability rating for the following reasons:

- The extremely limited line maintenance capability for primary generator/starter components
- The adverse converter weight
- No generator/starter deferred maintenance capability
- Need for a spare engine to correct a generator/starter failure or engine-out ferry to a location of a replacement engine
- High dependency on the APU for starter electrical power, assuming no suitable ground power source is available.

RESULTS AND CONCLUSIONS

Results

The technical and economic end result payoff on a total-airplane basis for the three SPS configurations is shown in table 31. The results are presented in terms of delta change from the conventional technology chin-mount installation SPS configuration. The maintainability ratings were established on the basis of premature component removals, component accessibility, remove and replace capability, and the need for fault isolation. The reliability ratings were established on the basis of engine starting function, various success criteria of electrical and hydraulic power generation, and dispatch reliability.

The equipment weight, cruise SFC, cruise thrust, and cruise drag differences for the three SPS configurations are also shown in table 31. The effect of these uncycled changes on a total-airplane basis can be best represented by converting these parameters to cycled takeoff gross weight parameter, which is shown in table 32. Another measure of these changes on a total-airplane basis is to convert the changes to total value of technology. This result is shown in table 33. The conversion factors were identified in table 16.

TABLE 31.—TECHNICAL AND ECONOMIC END RESULT STUDY SUMMARY—MODEL 767-620

OEW = 184 840 lb (83 840 kg) TOGW = 340 600 lb (154 490 kg)

Item	Unit	Secondary power system concept				
		Conventional technology		Engine pad-mounted *generator*/starter	Internal engine generator	
		Chin mount installation	Bifurcation mount installation		Cycloconverter	DC link
Uncycled parameters (delta changes)						
Equipment weight	% of OEW	Base	-0.065	-0.144	-0.037	-0.063
Cruise SFC	%	Base	0	-0.02	-0.03	-0.03
Cruise thrust[a]	%	Base	0	-0.04	-0.07	-0.07
Cruise drag	%	Base	+1.7	+1.8	+2.3	+2.3
Cycled TOGW for equivalent payload-range-performance objective	%	Base	+2	+1.95	+2.77	+2.73
	lb	Base	+6800	+6640	+9450	+9300
	(kg)	Base	(+3100)	(+3020)	(+4300)	(+4230)
Net total value of technology[b]	$/airplane	Base	+236 100	+192 625	+249 210	+281 810
Maintainability assessment relative to base		Base (100%)	May not be acceptable to airlines at this time	98% as good	90% as good	
Reliability assessment	All concepts are considered adequate for airline use.					

\+ *Denotes payoff*
\- Denotes penalty

[a]Reduction as a result of not extracting power from main engines.

[b]Assumes equal maintenance costs.

TABLE 32.—UNCYCLED PARAMETER CONVERSION TO CYCLED TAKEOFF GROSS WEIGHT FOR EQUIVALENT PAYLOAD-RANGE-PERFORMANCE OBJECTIVES—MODEL 767-620

OEW = 184 840 lb (83 840 ks) TOGW = 340 600 lb (154 490 kg)

Item	Unit	Secondary power system concept				
		Conventional technology		Engine pad-mounted generator/ starter	Internal engine generator	
		Chin mount installation	Bifurcation mount installation		Cycloconverter	DC link
Equipment weight Cruise SFC Cruise drag	% % %	Base	−0.11 0 +2.1	−0.238 −0.011 +2.2	−0.061 −0.017 +2.85	−0.104 −0.017 +2.85
Total	%	Base	+2%	+1.95	+2.77	+2.73
Effect on TOGW	lb (kg)		+6800 (+3090)	+6640 (+3030)	+9450 (+4300)	+9300 (+4210)

\+ Denotes payoff

\- Denotes penalty

TABLE 33.—TOTAL VALUE OF TECHNOLOGY (1972 DOLLARS)—MODEL 767-620

Item	Secondary power system concept				
	Conventional technology		Engine pad-mounted generator/ starter	Internal engine generator	
	Chin mount installation	Bifurcation mount installation		Cycloconverter	DC link
Weight Cruise SFC Cruise drag Equipment/engineering	Base Base Base Base	− 23 900 0 +263 500 − 3 500	− 53 000 − 2 560 +279 000 − 30 815	− 13 500 − 3 840 +356 500 − 89 950	− 23 500 − 3 840 +356 500 − 47 350
Net total value[a]	Base	+236 100	+192 625	+249 210	+281 810

\+ Denotes payoff
\- Denotes penalty
[a]Dollars per airplane—assumes equal maintenance costs

The cruise power extraction is shown in table 34. The differences are minimal and reflect the losses associated with energy conversion from one form to the other.

The projected effect of the nacelle configuration on the 767-620 airplane cruise drag is shown in figure 50. The drag rise associated with different nacelle configurations decreases as the airplane Mach number is reduced below the design Mach number. However, the drag associated with a chin-mounted installation at lower design airspeeds cannot be ignored. The effect of the four different nacelle configurations on an airplane designed for Mach 0.9 operation may be similar to that shown for the Mach 0.98 airplane, because a lower cowl fineness ratio will be used on the Mach 0.9 airplane. If one assumes that the drag penalty difference between the chin-mounted installation and the IEG concept remains the same at lower design Mach numbers, the uncycled parameter changes shown in table 31 remain the same. Its effect of TOGW and total value of technology, however, will be as shown in table 35.

Conclusions

The following conclusions are made, with the assumption that appropriate research and development items presented in part III are accomplished before the time period stated:

- The engine-pad-mounted GSD configuration is the best choice, assuming a 1975 engine go-ahead.

- The IEG configuration offers significant potential for a 1978 engine go-ahead. The cycloconverter variation represents a lower degree of risk considering its level of development.

The conventional technology engine fan duct bifurcation SPS configuration shows a substantial payoff potential with minimum risk and merits further study. This design is purely conceptual and requires that adequate access to the components be provided for airline acceptance. Unless a suitable maintenance approach can be developed, the potential payoff could be eliminated. The added study should include consideration of accessories with advanced technology envelopes (reduced size). The design is also affected by the engine size. Engine size below 12 300 kgf (27 000 lb) does not appear to have adequate space in the bifurcation area without affecting engine design.

The predicted reliability for the GSD and IEG concepts is comparable to that for the conventional technology concept and is considered to be acceptable for airline use.

The maintainability aspects of the GSD concept are comparable to those for the conventional technology concept. The IEG concept is ranked third in preference and is judged to be 90% as good as the conventional technology concept. This will reduce the potential dollar savings. An in-depth dollar cost assessment of maintainabilities was beyond the scope of this study.

TABLE 34.—SECONDARY POWER SYSTEM CRUISE POWER EXTRACTION PER ENGINE—MODEL 767-620

Item	System concepts		
	Conventional technology	Engine pad-mounted generator/starter	Internal engine generator
Engine bleed air extraction, lb/min(kg/sec)	96 (0.727)	96 (0.727)	96 (0.727)
Engine shaft horsepower extraction, hp(kW)			
Hydraulic	19.7 (14.7)	9.9 (7.4)	0 (0)
Electrical	80.5 (60.0)	95.3 (71.0)	107 (79.9)
Total	100.2 (74.7)	105.2 (78.4)	107 (79.9)
Delta	Base (Base)	+5 +(3.7)	+7 +(5.2)

Motor efficiency = 0.9
Generator/starter efficiency = 0.72
Internal engine generator system efficiency = 0.8
Gearbox efficiency = 0.98

TABLE 35.—MACH NUMBER EFFECT ON TECHNICAL AND ECONOMIC END RESULT PAYOFF—MODEL 767-620

Item	Secondary power system concept				
	Conventional technology		Engine pad-mounted generator/starter	Internal engine generator	
	Chin mount installation	Bifurcation mount installation		Cycloconverter	DC link
Cycled TOGW for equivalent payload-range performance objectives, %					
at Mach 0.98	Base	2.0	1.95	2.77	2.73
at Mach 0.95	Base	1.7	1.68	2.35	2.31
at Mach 0.90	Base	1.16	1.11	1.61	1.58
Net total value of technology[a]					
at Mach 0.98	Base	236 100	192 625	249 210	281 810
at Mach 0.95	Base	223 700	180 645	230 740	263 840
at Mach 0.90	Base	203 600	162 005	204 210	238 110

[a]Dollars per airplane—assumes equal maintenance costs

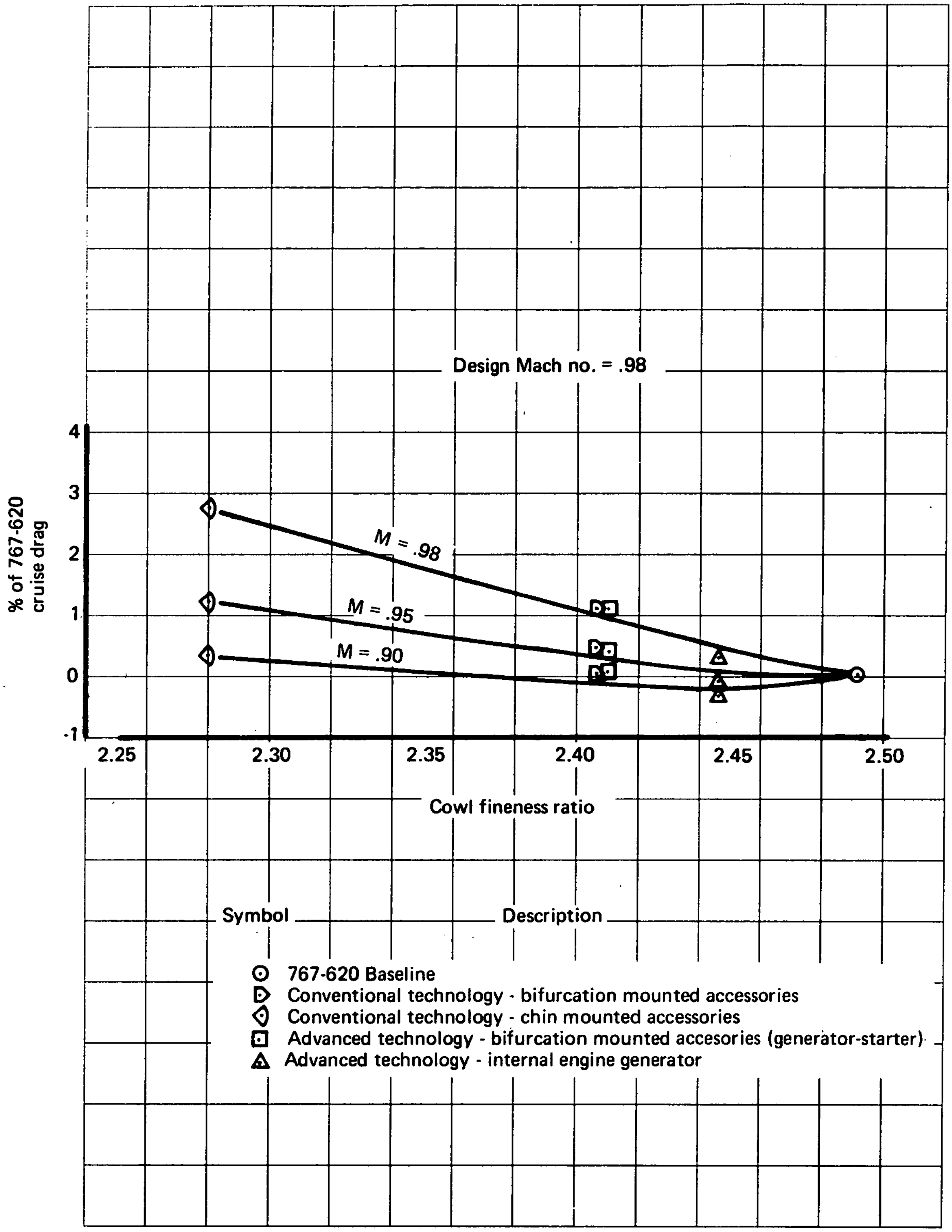


FIGURE 50.—NACELLE DRAG PENALTY

PART III

INTRODUCTION

This part of the report contains the recommended overall and individual research and development programs for advancement of those technologies identified in this study. Identification of potential payoff, assessment of the state of readiness of applicable technologies, and specific recommended action including estimated cost and schedules are provided.

A significant portion of the research and development activities outlined herein was shown in the final report on contract NAS1-10703, "Study of the Application of Advanced Technologies to Long-Range Transport Aircraft." This was done to provide continuity and to establish proper priorities relative to proposed programs in other technological areas. In the event the reports on contracts NAS1-10703 and NAS1-10893 do not exhibit identical data, this report shall take precedence because it reflects the most current effort.

The tendency to use off-the-shelf SPS equipment (electrical, hydraulics, pneumatics) as a cost saving expedient for prototype airplane designs has resulted in unnecessary penalties on the follow-on production airplanes. Typically, this equipment is of older design, and the SPS becomes a conglomerate collection of obsolete components, rather than an advanced integrated system. This approach does not promote an optimum SPS compatible with available technology and new airplane designs.

If development of the SPS elements is delayed until the production phase of each airplane, costly retrofit fixes to service problems and expensive maintenance will result. This has been a problem for current transport airplanes. Specific areas of research and potential payoff have been identified for a realistic program to permit timely incorporation of advanced technology SPS elements in future transport designs. The multiplicity of items, which include power source, electrical systems, hydraulic systems, and pneumatic systems, obviously require some priority rating. This is included in the Recommendations section.

RESEARCH AND DEVELOPMENT PROGRAMS

Gearbox and Ancillary Equipment

The subject technology improvements applied to the SPS have the potential to reduce airplane system weight by 68 to 90.7 kg (150 to 200 lb). This is equivalent to a cycled takeoff gross weight of 206 to 275 kg (475 to 605 lb). In addition, these improvements will contribute to reduced maintenance costs and improved reliability.

State of Readiness

Contractor research, engineering work on the United States supersonic transport (SST), and government-funded studies related to SPS have identified items that offer significant potential gains. These include integrated bevel gearing/gearbox design, integrated accessory and gearbox gearing, constant-speed gearboxes, two-speed gearboxes, and high-speed shafting. Current airplanes use state-of-the-art shaft-driven gearboxes incorporating primarily spur gearing. The SST program initiated development work on high-speed, high-power transmission shafts and gearing. Industry proposals concurred that development of high-speed shafting and gearboxes with bevel gearing could substantially reduce weight and gearbox envelope.

Recommended Action

It is recommended that a program to reestablish development of high-speed shaft drives and integrated bevel gearing development be initiated. This program, rather than being restricted to a specific airplane configuration, should be aimed at developing the criteria to be used for future drive systems. The program would consist of two phases.

Phase I.—Initiate an analysis to define gearbox gearing arrangements and drive shaft speeds. This would include possible arrangements of airplane accessories and pad output speeds. The relative drive shaft speeds and gearbox definition should include, as a minimum, input speeds of up to 2100 rad/sec (20 000 rpm), shaft power transmission of up to 373 kW (500 hp), and acceptance of typical advanced technology accessories.

Phase II.—Establish procurement and conduct tests on selected shaft and gearing arrangements that offer the highest potential payoff. This information will be used to establish basic design information for future airplane applications, cost, and schedules.

A program plan is shown in figure 51. Phases I and II are each of 12 months' duration. The predicted total program cost of $600 000 is divided into $100 000 for shaft development and $500 000 for gearing criteria development. The phase I effort will define the follow-on work.

Discussion

Concepts that offer the potential for reducing gearbox and associated SPS component weights are discussed below.

Integrated bevel gearing/gearbox design.—Current gearbox design uses straight spur gearing to match output pad speed to gearbox input speed. Use of bevel gear sets offers a significant potential reduction in gearbox size and weight because of the many optional accessory locations available.

Integrated accessory and gearbox gearing.—Those accessories requiring a high-speed-ratio gearbox for the component (e.g., air turbine engine starter) could have the main gearbox incorporate part of the component gear ratio, thus reducing the component gearbox. The pad interface could be integrated with the gearbox so that the component would contain its mating drive gear, thereby eliminating the requirement for a splined interface. This would normally result in a net weight and size reduction. The degree of integration acceptable and benefits derived for a particular design will depend on the size and quality of accessory components.

Constant-speed gearboxes.—The constant-speed drive (CSD), which currently drives only the generator, could be revised to drive the entire gearbox. Studies to date have shown no particular advantage for this type of drive. The CSD required to drive the gearbox is relatively heavy and less efficient than a straight gear drive. In addition, there is essentially nothing to be gained by driving the hydraulic pumps at constant speed, because the study engine does not have a wide speed range between flight idle and takeoff power (e.g., 74% to 100%). Development work in this area does not appear to be justified for this narrow speed range engine. However, there would be potential gains with wider speed ranges.

Two-speed gearboxes.—A clutch device and necessary controls could be incorporated into the gearbox to permit shifting of gear ratios. This arrangement offers potential weight reduction if used in conjunction with a suitable shaft-driven air compressor. For a narrow speed range engine, it might also be beneficial to drive the generator directly and convert only a small quantity of electric power to 400 Hz, and to operate most of the electrical loads with the frequency variation that results from the speed shift. Consideration of this option is directly dependent on the compatibility of the electrical load equipment. The potential gain in use of a two-speed gearbox to drive hydraulic pumps would be small on the narrow speed range engine. This approach would be more attractive on engines with wider speed ranges from idle to full throttle.

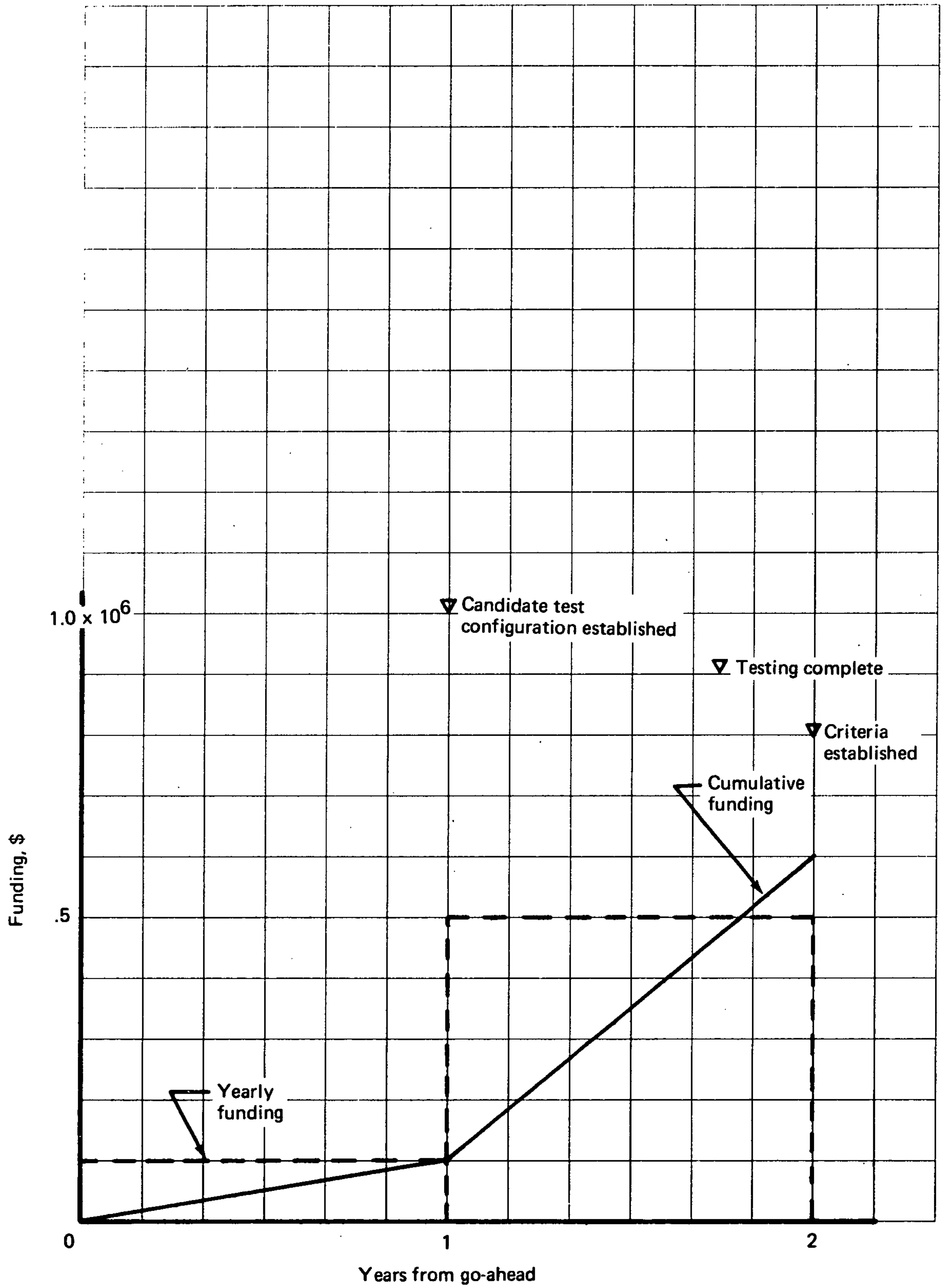


FIGURE 51.–GEARBOX AND ANCILLARY EQUIPMENT PROGRAM PLAN

High-speed shafting.—High-speed shafting coupled with high-speed gearboxes and accessory components represents a potentially significant reduction in engine frontal area and system weight. Gearbox size is dictated by accessory size and spacing, power extracted, and input and output speeds (gear sizing), accessory weight, and gearbox location. Smaller high-speed accessories with high input drive shaft speeds would allow smaller gearing, tighter spacing of accessories, and reduced gear-box loads from accessory overhung moments. The operating life and environmental requirements for high-speed airplane accessory drive shafting are beyond the presently applied technology. Shaft speed must run through, and possibly at, second and third critical frequencies. In addition, a high degree of radial and axial misalignment is experienced. Crowned splines can accept a high degree of both axial and radial misalignment, but development is insufficient to establish that life and service requirements could be met. A developmental accessory drive for the SST incorporated crowned splines on the main drive shaft. These splines were cooled and lubricated by a continuous oil bath. The total running time was too low to establish firm conclusions. Flexible diaphragms can provide the life requirements but are limited on axial misalignment and flexure per diaphragm. Therefore, some form of slip joint in the shaft and a series of flexible diaphragms are usually required. This can limit life and is potentially heavy.

Advanced Electrical Systems

Potential Payoff

The development and application of advanced design techniques and hardware to the electrical system offers the potential benefits per airplane listed in table 36.

TABLE 36.—ADVANCED ELECTRICAL SYSTEMS POTENTIAL BENEFITS[a]

Item	Delta cost, dollars in thousands	Delta equipment weight	Other benefits
Automatic electric energy management	+ 25 to + 50	–150 to –300 lb (– 68 to –136 kg)	Reduced crew workload Automated systems
Advanced wiring techniques	-150	–200 to –300 lb (– 91 to –136 kg)	Reduced maintenance Flexibility for changes
Fail-operative power system	- 5 to - 7	–100 to –150 lb (– 45 to – 68 kg)	Increased safety Improved reliability
Subtotal	-132	–450 to –750 lb (–204 to –340 kg)	
Generator/ starter drive SPS configuration	- 31	+267 lb (+121 kg)	1.95% cycled TOGW reduction

[a]The combined benefit would result in a cycled TOGW reduction of 2.5% or 8500 lb (3850 kg)

State of Readiness

The results of concept evaluation studies and initial hardware development programs provide high confidence that the projected payoffs can be realized. The contractor and others in industry have laboratory-tested automatic electric energy management systems (AEEMS) employing such concepts as automatic load management with programmable logic, signal multiplexing, single point data entry and display, and remote power controllers, as shown in figure 52. In addition, application studies for systems employing various combinations of these concepts have been made. Study results show feasibility and high payoff potential, but also indicate the need for significant improvements in system design and hardware prior to application.

Research programs involving aluminum wire, small-gage wire (SGW) harnesses (reinforced), and flat conductor cable (FCC) have confirmed the feasibility of advanced wiring techniques and the projected weight and cost benefits. Initial application studies and limited hardware testing have been completed.

Contractor-funded research has been conducted to study a large number of candidate electrical power system configurations compatible with the requirements of fail-operative automatic flight control systems. Analysis of these reveals the need for expanded effort to reduce cost and weight and to develop a variety of system components to permit realization of the required reliability and safety.

The generator/starter drive (GSD), a generator integrated with the constant speed drive (CSD) with engine starting capability, was developed for the model 727 airplane. Tight scheduling prevented development to the application stage, but the axial gear differential design, developed for the GSD, is now universally used in modern constant-speed generator drives.

Recommended Actions

The three-phase program shown in figure 53 is proposed.

Phase I.—This effort would be primarily analytical and would evaluate electrical system configuration, define trades of selected concepts, initiate hardware evaluation, and identify development items for the next phase.

Phase II.—This phase would continue selection of hardware to be designed, built, and tested in the laboratory. Probable candidates are:

- Hybrid remote power controllers (0.5- to 75-A ratings)

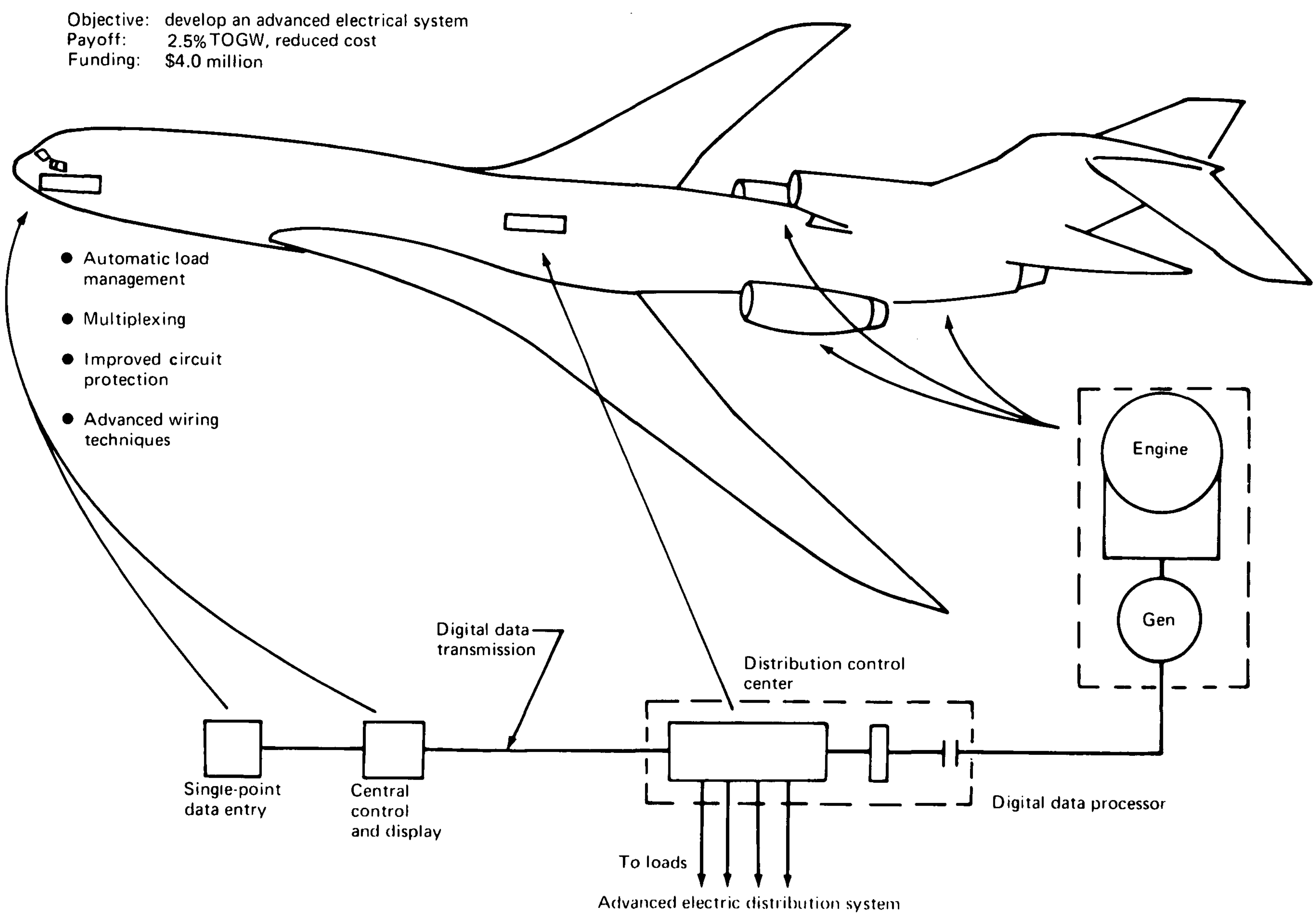


FIGURE 52.—AIRPLANE AUTOMATIC ELECTRIC ENERGY MANAGEMENT

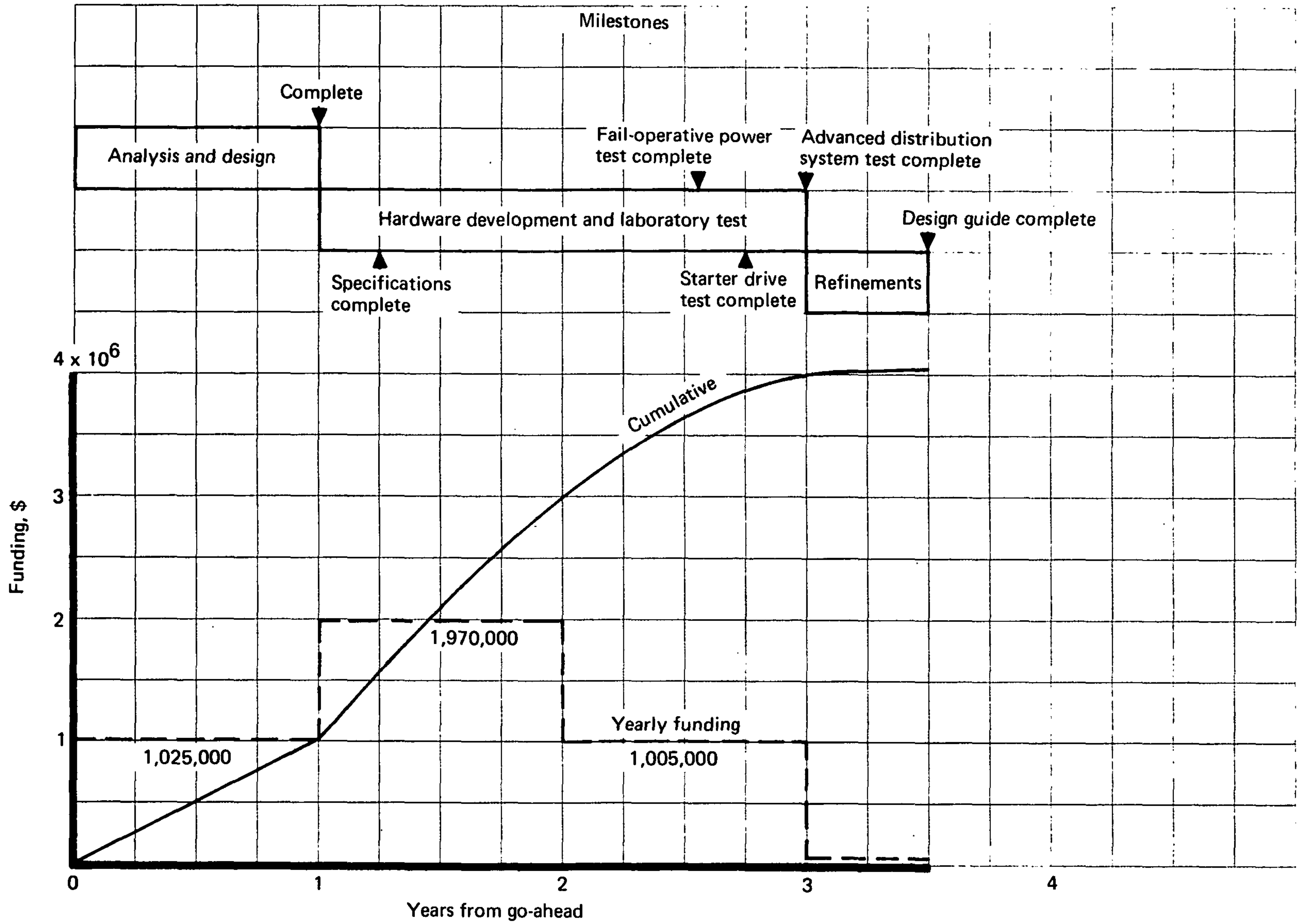


FIGURE 53.—ADVANCED ELECTRICAL SYSTEM PROGRAM PLAN

- Advanced wiring techniques—small-gage wiring harness, flat conductor cable, distribution/interconnection centers

- Fail-safe load transfer system

- Fail-safe APU intertie system

- Automatic electrical energy management system

- Integrated GSD and controls

A firm selection of equipment must await completion of phase I.

Phase III.—Problems uncovered by the hardware test program would be corrected and a design guide prepared for applying new wiring techniques.

Cost and Schedules

The recommended development program outlined herein would require 3.5 calendar years for completion and a total funding of $4.05 million.

Discussion

Automatic electric energy management system.—The principal objective of research in this area is to apply advanced techniques to the electrical system management. These techniques include multiplexing of control and information signals, automated control, remote power controllers (RPCs), failure detection and annunciation, and others. These objectives are dependent on the development of certain key components:

- Automatic systems management and control (ASMC) system

- RPCs, both hybrid and solid state

- Advanced crew controls and displays

Benefits to be derived from successful development of these include:

- Up to 136 kg (300 lb) weight savings per airplane through remote location of the electrical load center and use of RPCS

- Improved fault clearing with minimum transients
- Automatic control of the electrical generating system and power distribution
- Reduced wiring and wiring complexity
- Reduced crew workload
- Complete system status information and failure identification
- Reduction in flight cabin space requirements

Practical implementation of automatic systems management can be accomplished with conventional sensing and logic signals transmitted via conventional signal wiring. To realize the maximum benefits, however, a multiplex system must be developed that is capable of handling the large number of signals required with high reliability.

The application for RPCs arises primarily from the wire weight benefits to be gained by distributing power from a load center located in an area remote from the flight crew station. The FAR requirement that circuit breakers be resettable in flight results in the need for remote control of circuit protective devices. A remote thermal circuit breaker with electromagnetic control (RCCB) has been developed for this application and is in use in some newer airplanes. These RCCBs are relatively heavy, and extensive use in an airplane could erase the potential savings in wire weight. A conventional electromechanical relay with solid-state overload current sensing/tripping circuits ("hybrid" RCCB) appears potentially more attractive from a cost and weight standpoint. Development of these has reached a state where feasibility of their application appears assured; however, these RCCBs include backup circuit protection in the form of fusible links which are not compatible with some circuit applications. With a continuation of the development program on the "hybrid" RCCB, qualified units could be available within 1 year and would be competitive replacements for the presently used relay/thermal circuit breaker combination.

The "hybrid" RCCB may be an interim device. The development of a solid-state remote power controller (RPC) could make the RCCB obsolete. The RPC, a solid-state relay with built-in wire/equipment protection, has been under development for a number of years. For some applications, primarily low-power dc circuits, the RPC may be acceptable in its present form. For higher power levels and for many ac power circuits, device limitations tend to restrict its use. Power dissipation in the RPC, resulting from device voltage drops, may require special mounting to cold plates, etc., to dissipate the heat. The ac power controller has a potential failure mode that can cause dc power to be applied to the load and, until this is resolved, it would be unacceptable for aircraft use (ref. 9).

Advanced crew controls and displays are needed to provide more systems information to the flight crew without increasing panel area and complexity. Electronic displays of a variety of information can supplement or replace conventional panel meters. Failure annunciation can be signaled by flashing an out-of-tolerance parameter on the display. Digital panel meters can replace conventional pointer types with greatly improved accuracy and readability. The gains to be expected from work in this area will be a reduction in flight cabin clutter and complexity with better control of airplane systems and less confusion. Flight safety will be enhanced as a result.

Advanced wiring technology.—The development and application of advanced wiring and installation techniques is a particularly fruitful area with respect to savings of weight, cost, and maintenance. Development activities in five categories are recommended: wiring system integration, aluminum wire development (both round and flat conductors), flat conductors for power feeders, fly-by-wire system wiring, and computer wire sizing.

Wiring system integration is aimed at "total" airplane wiring design, using integration/distribution/interconnection centers. These centers would allow maximum use of standard point-to-point bundles requiring a minimum amount of coding. Wire integration within the center provides maximum flexibility to effect changes and repairs. The potential cost-of-ownership reduction is predicted to exceed 50% of present total wiring costs.

Development of aluminum wire to replace copper will emphasize expanded use of smaller gages. Present practice limits the use of aluminum conductor sizes to AWG 8 and larger. The bulk of airplane wiring, however, is in sizes AGW 18 to 24, and the successful development of high-strength aluminum conductors (in both round and flat configurations) in the smaller sizes will permit large weight savings.

Flat conductor cable (FCC) offers significant benefits to the electrical distribution system. FCC exhibits better heat dissipation, and more current can be conducted for a given circular mil area. Some FCC three-phase configurations have much lower inductance than symmetrical groups of round wire. The lower reactance is directly convertible to weight savings. A potential application that may benefit significantly from the use of FCC is the high-frequency feeders for the VSCF (IEG) systems.

"Fly-by-wire" automatic flight control systems impose a level of criticality on airplane wiring that necessitates new designs, guidelines, and techniques. Certain sections of airplane wiring, such as those connected to frequently replaced units or located in unprotected "traffic" areas, receive heavy wear. The designer should have sufficient data to enable him to design bundles that provide adequate reliability and safety for the least installation and maintenance costs. In addition to employing proper bundle isolation techniques, techniques to improve wire harness strength, bundle stress relief, and termination techniques are required. Other techniques such as controlled wire lay, continuous jackets, metal braid, etc., can be used to improve bundle integrity.

Wire sizing by computer can avoid excessive conservatism, which frequently results in unnecessary wiring weight. Wire selection problems and engineering time can be reduced. Input data would include load requirements, wire type, bus assignment, separation, initial routing, voltage drop limits, locations of source and load, etc. Output would specify proper routing, bundle assignment, spare bundle capacity, voltage drop, and bus loading and balancing. Airplane wiring system integration would be enhanced.

Fail-operative power system.—The trend in airplane design is toward greater use of electrically powered flight-critical systems. Two examples are: autopilot systems with category III autoland capability, and stability augmentation systems (SAS). Dependence on these systems for safe flight and landing imposes an order of reliability far exceeding that on current airplanes. Typically, this reliability is achieved by increasing the redundancy of equipment performing like functions. The power sources for this equipment must be designed and arranged to preserve the required reliability, not detract from it. Parallel operation of generators is not desirable for the reason that a single fault affects the entire system until the fault is cleared. In some cases, this could take 4 to 6 sec and would impose undesirable requirements on some critical load equipment.

The principal tasks required to arrive at generating systems compatible with flight-critical load equipment are:

- Establish fail-operative user equipment requirements with more detail and accuracy.

- Conduct electrical system configuration studies aimed at simplifying fail-operative systems for two-, three-, and four-engine airplanes.

- Perform detailed design/development, evaluation, and comparison of the final configurations.

- Develop a fail-safe load transfer system. This will permit making more power sources available to selected loads while maintaining adequate electrical isolation between power sources.

- Develop a fail-safe APU power intertie system to allow use of a single APU generator as a backup source. This generator should be connected to any one of several independent system sections in flight and to all sections on the ground without violating fail-operational characteristics of the overall power system during flight.

- Develop new contactors (generator, bus tie, and transfer circuit breakers) in which the usual auxiliary contacts are replaced by triple-redundant, solid-state, magnetic or other contactless position-sensing devices. Also develop the external circuits required for the power system to interface with the new circuit breakers.

Generator/starter drive.—The GSD application is covered in part II.

Hydraulic Power Generation, Distribution, and Control

Potential Payoff

Significant payoff could be achieved by selected research and development activities in hydraulic systems. A potential reduction of up to 353 kg (780 lb) in system weight is expected. This savings is equivalent to a cycled reduction of TOGW of 0.7% or 1080 kg (2370 lb).

In addition, substantial benefits in terms of initial cost and maintenance savings are possible. Savings of $19 000 per airplane in initial costs and $17 000 per year per airplane in maintenance are expected. Reduced nacelle size, improved maintainability, and improved system reliability would be achieved.

State of Readiness

The status of the proposed research and development activities in hydraulic systems can be described in four basic categories.

Category A items are technology advances which have been developed through the laboratory phase, and in some cases have been incorporated in military or experimental airplanes. Generally, performance benefits have been identified. Work is now required to achieve reliability and long life in an economical fashion consistent with commercial objectives. Specific items are:

- 2.76×10^7 N/m^2 (4000 psi) operating pressure
- High-speed pumps
- Hydrofluidic systems
- Intermittent-duty power sources
- Fixed-body actuators with flexible rods for load attachment

Category B items are new technology items which as yet have not proven practical but indicate promise of performance and cost benefits. Such items include structure-integrated fluid distribution, flywheel energy storage, and permanent assembly of hydraulic components.

Category C items are development activities resulting from technology trends in systems which interface with hydraulic systems. Work is required to adapt the hydraulic system to this new interface so that the full benefits of the technology advance can be utilized. Proposed items are:

- Large integrated electric-motor-driven hydraulic pumps. The need for this item would be amplified with incorporation of an internal engine generator.

- Continuous-duty hydraulic-motor-driven generators of 5- to 10-kVA size.

- Integrated actuator packages.

Category D items are directed to solutions of problems inherent in present systems. Reduction of control valve erosion (fig. 54) and development of power transfer units fall into this class.

This development work is necessary to provide the most cost-effective hydraulic system. The deficiencies to be improved upon are primarily excessive weight, initial cost, and maintenance cost.

Recommended Action

Development effort is recommended in each of the above four areas. Priority should be given the electric-motor-driven pump, integrated actuator package, and the valve erosion problem. NASA, in conjunction with industry, should prepare work statements leading to development program contracts. A proposed plan for these programs (analysis, hardware development, and test) is shown in figure 55.

Cost and Schedules

The recommended research and development in the hydraulic systems area would require 6 calendar years for completion and an estimated total funding level of $12.8 million.

Discussion

Category A

- High-pressure hydraulic systems—The trend has been toward increased use of hydraulic power in secondary power systems. The increased power demands have mainly been

FIGURE 54.—EXAMPLE OF SEVERE CONTROL VALVE EROSION AFTER 3.6×10^5 SECONDS (100 HR) OF LABORATORY TESTING

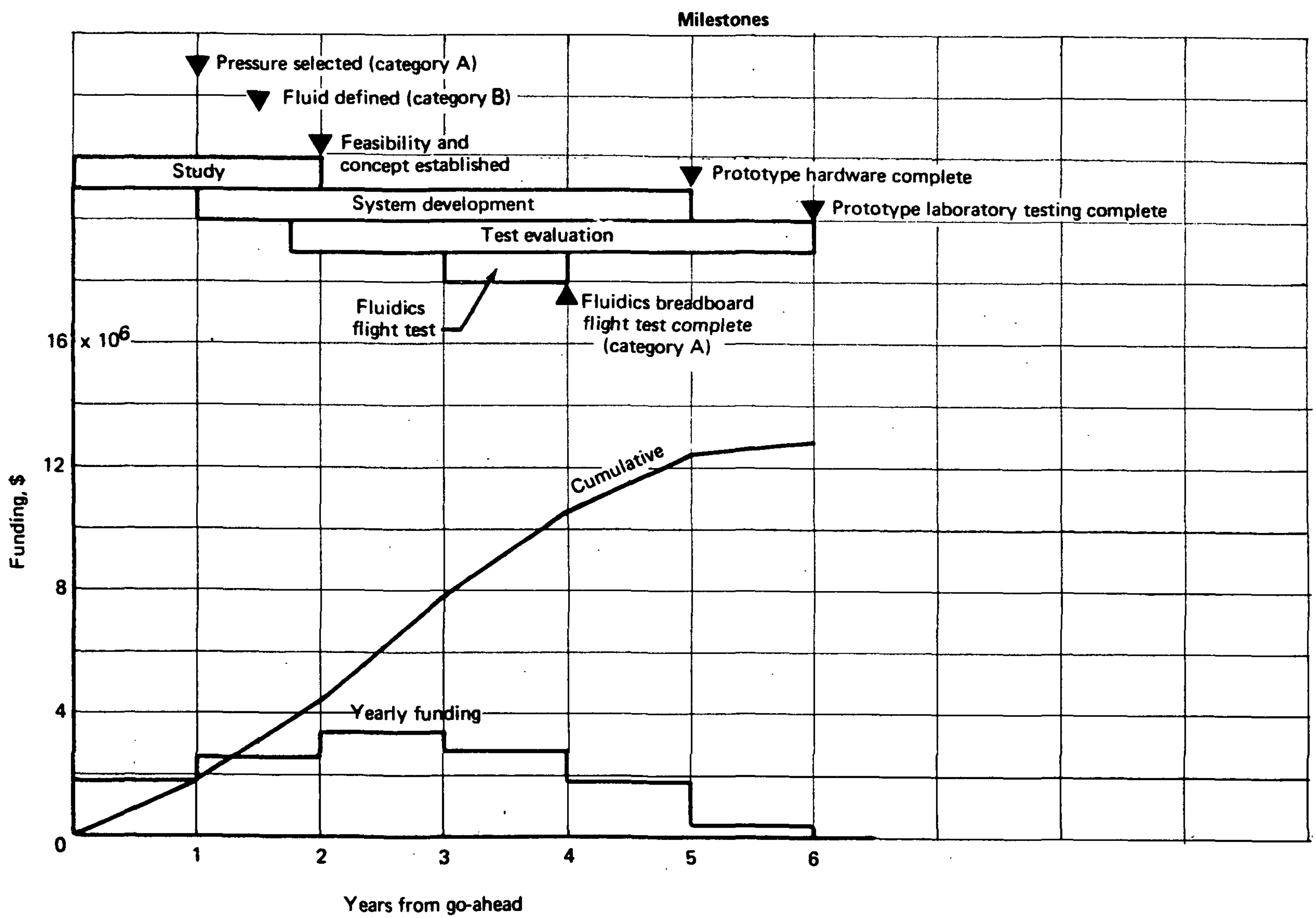


FIGURE 55.– HYDRAULIC POWER GENERATION, DISTRIBUTION, AND CONTROL PROGRAM PLAN

satisfied by increasing the pump flow capability, which results in larger envelopes and weights for pumps and distribution system tubing. Increased pressures would reduce pump size and distribution system weight. For example, on the advanced technology airplane a weight reduction of 67.5 kg (150 lb) was achieved by increasing pressure from 2.07×10^7 to 2.76×10^7 N/m^2 (3000 to 4000 psi). However, the optimum pressure level for minimum system weight is configuration dependent. Increases in system pressures will require hardware and seal developments. Special considerations must be given to flight control actuator dynamic stiffness as pressure is increased (and actuation area reduced).

- High-speed pumps—Future airplanes with higher cruise speeds will benefit from the development of smaller hydraulic pumps if the pumps continue to be engine mounted. Drag penalty at high subsonic Mach number associated with nacelle frontal area is the incentive to increase pump operating speed to reduce pump volume. Weight savings may be achieved for both the pump and gearbox by increased pump speed. Considerable developmental work will be required to achieve economical high-speed pumps without sacrificing present reliability and to obtain low noise level equipment.

- Hydrofluidic systems—Because the number of moving parts can be reduced, thus providing a potential increase in system reliability, fluidics appear to have many promising applications in hydraulic systems. A prototype artificial feel computer has been designed and tested. The hardware shows a considerable reduction in both weight and volume at a rate of one to four over a conventional feel computer. A landing gear sequencing circuit has been designed and brought to the breadboard stage. Analysis and testing was performed on a backup fluidic stability augmentation system (SAS) for the SST. Other applications which look promising but require additional feasibility studies are: active failure monitoring, detection, and switching, and engine inlet controls.

- Intermittent-duty power sources—A limited-life, lightweight, low-noise pump is needed to provide supplemental hydraulic power for a limited-duty-cycle operation such as flap or landing gear actuation. Typically the duty cycle for these utility functions is 10-15 sec twice per flight for landing gear actuation and less than 60 sec four times per flight for flap actuation. Such a unit would permit a reduction in the basic hydraulic system, which is normally sized by the utility loads.

- Fixed-body actuators—A considerable improvement in performance and reliability, together with reduced cost and weight, appears to be possible through the use of fixed-body actuators with flexure-loaded connecting rods to drive the control surface. One end of the actuator is permanently fixed to structure to obtain negligible angular deflection of the actuator body. Angular deflection is accomplished by the flexible rod. Preliminary work on the SST program indicated improved system performance resulting from higher stiffness and reduced backlash in the servo loop.

Category B

- Structure-integrated fluid distribution system—Portions of the hydraulic distribution system could be integrated in the basic airplane structure to perform the dual functions of providing a fluid path and adding structural stiffness. This would result in the potential for higher reliability, improved maintainability, and savings of weight, cost, and space. The structure-integrated distribution system would require installation of major portions of the distribution system during the early stages of airplane production. The fluid path could be included in extrusions, built-up tube/sheet metal combinations as part of stringers, and built-up structure panels. When tubing is a part of built-up structures, the tubes may be joined to these structures by bonding, brazing, or welding. Good joining techniques are required between major structural sections.

- Flywheels in secondary power systems—Flywheel energy storage systems have been considered for both utility and flight control functions. Laboratory hardware has been built and tested which shows the potential for weight reduction (ref. 10). Flywheels can ensure better utilization of the power sources if surplus power is stored during periods of low power demands. This may allow a reduction in the size of the power source and transmission lines, resulting in an overall weight savings. New flywheel designs using new materials allowing higher rotational speeds and correspondingly higher energy storage capacity per unit mass, combined with the development of high-speed hydraulic pumps and motors, appear potentially attractive and should be investigated.

- Permanent assembly of hydraulic components—The primary intent of this item is the permanent assembly of actuators, with end caps welded or brazed in place on the cylinder. This would eliminate the need for static seals and potential leakage paths, and thus improve reliability and maintainability. This concept is primarily applicable to simple actuators without valves. Locking actuators in the landing gear systems are typical examples. Emphasis should be placed on developing simple, low-cost components on which no maintenance or overhauls are planned.

Category C

- Large integrated electric-motor-driven hydraulic pumps—The need for these motor-pump combinations is contingent on the development of high-capacity engine-driven generators. Typically the total hydraulic demand in each hydraulic system would be satisfied by two pumps, each with capacity comparable to that of the largest pumps currently used in commercial airplanes. High-speed, integrated, direct drive motor-pump units are envisioned to develop lightweight hardware consistent with high reliability. The integration of motor and pump should decrease the number of rotating parts and bearings, which should ensure good reliability. Since these motor-pump units will be installed closer to the passenger cabin than current engine-driven pumps, low motor-pump noise must be among the primary design considerations.

- Continuous-duty hydraulic motor generator—There is a requirement for increased redundancy of electric power generation for flight-critical automatic guidance and control systems. Small (5-10 kVA) hydraulic-motor-driven electrical generators can satisfy this need and allow retention of the current operating flexibility of dispatch with one main electrical generator inoperative. Developmental program emphasis should be placed on high-reliability, lightweight, and low-noise units.

- Integrated actuator packages—The use of integrated actuator packages should receive serious consideration for future transports. Several factors will make them competitive with a central hydraulic distribution system. Optimum configurations will probably mix integrated packages with packages powered by a central system.

  - The improved ability to create high-fidelity simulations early in the airplane program will permit early establishment of aerodynamic surface rate requirements. Volume and weight of the integrated package can thus be minimized.

  - The current development work on servopumps for integrated packages will permit reduced heat rejection, thus reducing volume and weight for heat exchangers.

  - The evolution of fly-by-wire flight control systems will provide a more advantageous interface with the integrated packages.

  - The development of an IEG may provide a high-power, high-reliability source of electrical power for the integrated packages.

  The units currently in use are large, heavy, and difficult to package in the advanced technology transports.

Category D

- Valve erosion—Hydraulic control valve erosion is the cause of high maintenance and overhaul cost. The goal for this research is to reduce valve erosion to one-third of present levels. If this goal is attained, a savings of $17 000 per airplane per year can be realized. A major cause of the valve erosion on current commercial transport airplanes is believed to be the type of hydraulic fluid used. This research should concentrate on reducing valve erosion through:

- Development of a low-cost, low-erosion, fire-resistant hydraulic fluid
- Improved valve design
- Improved valve materials

- Power transfer units (PTU)—Bidirectional PTUs are coming into use on new airplanes to transfer power between hydraulic systems to improve flight safety (provide redundancy without mixing system fluid). Use of fully powered flight controls without manual reversion and large distances between pump and power sources (i.e., wing- and tail-mounted engines) have provided the impetus for PTU development.

  Improvements are required in PTUs in the areas of starting torque and operating stability. Future airplanes can be expected to use PTUs to reduce system weight and complexity. Improved PTUs will allow true load sharing between hydraulic systems, which in turn could reduce pump size (weight and envelope).

Pneumatic, Conditioning, and Protective Systems

Potential Payoff

Advanced technology in pneumatic, conditioning, and protective systems has the potential of reducing system weight by 364 kg (800 lb) and cruise SFC by 1.75%. These items are equivalent to a cycled takeoff gross weight reduction of 1.7% or 2640 kg (5800 lb). In addition, these improvements should reduce maintenance cost and airline warranty claims.

State of Readiness

Contractor research, engineering work for the SST, and industry effort in design of the 747, DC-10, and L-1011 airplane systems have exposed and identified development items having the potential of achieving the above payoff. Specific items are: expansion of cabin air recirculation, air source development, fuel autogenous ignition temperature definition, precooled fuel conditioning, cooling cycle evaluation, ice protection requirements, low-drag cooling system design, windshield heating system, and total energy management secondary power concept development. Some current subsonic jets use a limited amount of unfiltered cabin air recirculation. The SST program investigated the use of precooled fuel and sponsored preliminary development of shaft-driven boost compressors. A substantial amount of data is available on fuel autogenous ignition temperature in general, but data on the fuel vapor air duct interrelationship are lacking. Several approaches to overall systems energy management have been pursued by industry without a positive solution.

Recommended Action

It is recommended that a program be initiated and funded by NASA as follows:

Phase I.—Initiate analysis of the items that would most likely affect requirements of the engine (long lead item). These items would be: cabin air recirculation, fuel autogenous ignition temperature, precooled fuel, and cooling cycle evaluations. Results should include evaluation of existing data, establishment of design criteria, identification and analysis of promising concepts, and identification of hardware for test.

Phase II.—Fabricate, assemble, and conduct laboratory test of hardware identified in phase I as long-lead, high-payoff components.

Phase III.—Integrate components developed in phase II into an integrated system test rig. The system will also contain components not critical to engine development, but significant to overall state-of-the-art improvement. Conduct system tests to demonstrate performance and component compatibility. Significant airplane/system interfaces would also be evaluated.

Phase IV.—Fabricate and install the system in a prototype or flight test airplane. Conduct ground and flight tests to demonstrate performance and interface compatibility.

Cost and Schedules

A program plan is shown in figure 56. Each phase would take 1 year with some overlap, so that total program duration is about 3.5 years. The predicted total program cost is $5.3 million.

Discussion

Pneumatic power source development

- Engine bleed air—Engine bleed air is usually extracted from two available stages. The use of three or more stages to minimize engine bleed air penalty is impractical because of inherent engine design characteristics. This subject has been reviewed in some detail with engine manufacturers during previous in-house studies and is generally summarized below.

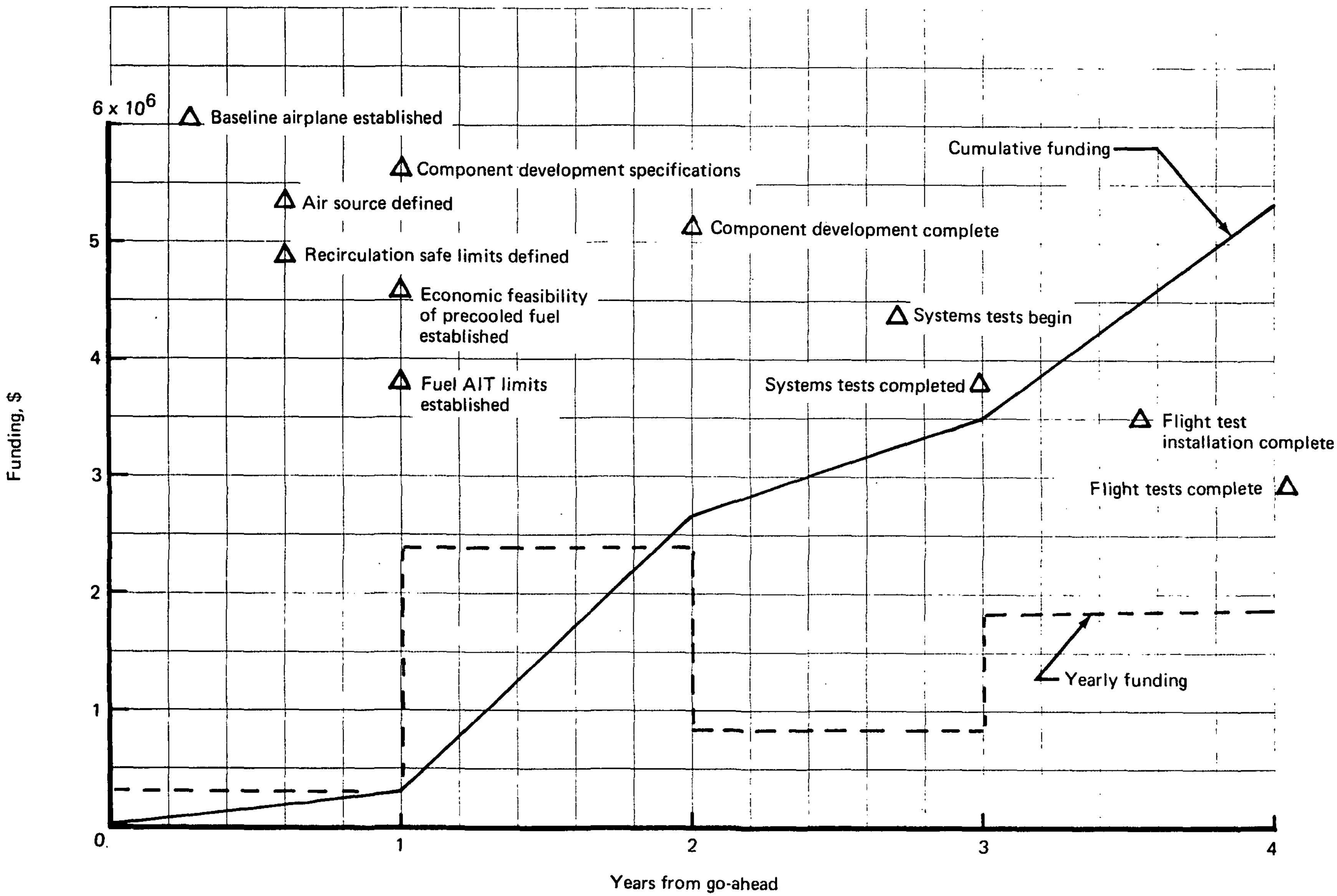


FIGURE 56.—PNEUMATIC, CONDITIONING, AND PROTECTIVE SYSTEMS PROGRAM PLAN

The basic engine is designed for no bleed. Industry practice is to design the compressor with a certain degree of margin. The engine is then uprated for operation with no bleed air extraction. Therefore, there would be no apparent engine weight savings if the engine were to be designed for engine bleed. To extract bleed air from an engine it is necessary to install manifolds and open passages to the specific compressor stages required. The basic engine is designed with a semblance of manifolding for both strength and potential surge bleeding. The weight required to add normal bleed provisions is therefore minimal (e.g., 2.25 to 4.5 kg (5 to 10 lb)).

Adjacent compressor stages are difficult to bleed because of the physical requirement for separate external manifolds. Engine length could be extended to provide the required space, but this would result in a significant weight penalty.

Some stages are essentially buried and would be extremely difficult to bleed. This would include stages adjacent to the last fan stage and those stages under the variable geometry (stator) controls. Any consideration of these stages would result in weight penalties.

Multistage bleeds (up to four stages) could be controlled by external valving and manifolding.

Studies conducted in-house showed essentially no weight differences for the number of stages selected because the variation in precooler size required for each configuration balanced the changes in valving and manifolding.

Integrating the bleed system within the engine would provide the optimum operating configuration. However, the engine suppliers indicated that the space requirements and sealing provisions between stages would add significantly to the engine costs and weight, making this approach impractical.

The inclusion of design provisions within the engine to limit bleed air to 505° K (450° F) were considered. The consensus of the engine suppliers was that this would cost considerably more than the installation of an external precooler and controls. No detailed cost estimate was made.

- Shaft power—The use of engine shaft power to drive a compressor that provides the pneumatic power source allows the deletion of the precooler and bleed air manifolding on the engine. The result is that the added gearbox and compressor components can be installed with essentially no weight penalty. The primary benefit is in reduced airplane operating penalty, because shaft power is less costly than bleed air. This overall weight reduction can be realized only if the compressor design allows the elimination of the precooler. To do this, either a controlled variable-speed drive or variable-geometry compressor would be required. Higher AIT limits (see below) might simplify compressor design. Development of a variable (two) speed drive was initiated for the SST. However, considerably more development for the drive and compressor controls is required to establish whether the operating penalty reduction could be realized. The addition of continuous-duty, high-speed rotating components may increase total maintenance costs.

- Autoignition temperature (AIT) of fuel—Tools to evaluate the AIT of fuel relative to the actual pneumatic system installation are required. For instance, a duct running through the engine strut near a fuel line constitutes a potential hazard if the air temperature in the duct exceeds the AIT of fuel. Present design requirements call for the air temperature to be limited to a safe level. Industry practice has been to use a conservative approach and over-precool air for the pneumatic systems. Past practice has been to limit duct air temperature to 505° K (450° F). This was based on laboratory type fuel properties tests, which are not representative of aircraft duct installations in potential fuel vapor compartments. Tests have been conducted recently by the U.S. Bureau of Mines for the Air Force, in which aircraft installation was roughly simulated and JP-4 fuel was used. These tests showed that temperatures of the order of 810° K (1000° F) are required for ignition, and that a slight amount of air ventilation around the duct helps to raise the ignition temperature. Similar tests have been run by the British Royal Aircraft Establishment, in which the fuel vapor compartment around the duct was spherical and the sphere temperature was independently controlled. Jet A kerosene fuel was used. The tests showed that a temperature of the order of 616°K (650° F) is required for ignition when the sphere temperature corresponds to expected aircraft fuel vapor compartment temperatures. The compartment wall temperature is a significant parameter.

  More representative aircraft installation tests with the proper fuel are required to establish more realistic design requirements.

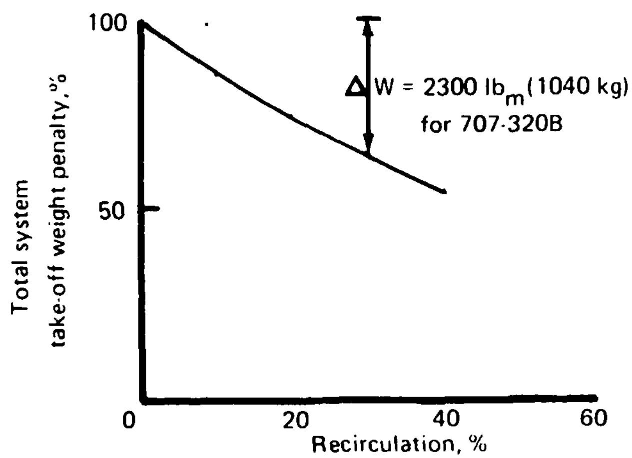
Conditioning systems

- Cabin air recirculation—This reduces the quantity of fresh airflow demanded. The aircraft industry accepted practice is to supply 9.44×10^{-3} m^3/s (20 cfm) of fresh air per passenger to the occupied cabin volume. This represents a significant SPS requirement and weight penalty when projected future airplane sizes are considered. Studies conducted on the model 707 showed that the airplane TOGW could be reduced by approximately 1043 kg (2300 lb) using 25% recirculated air (fig. 57). With the development of acceptable filtration and cabin air recirculation techniques, a spacecraft life support-type system

Objective : Define an optimum air recirculation system to improve overall airplane efficiency

Payoff: Equivalent to 1% TOGW reduction

Funding: $185,000



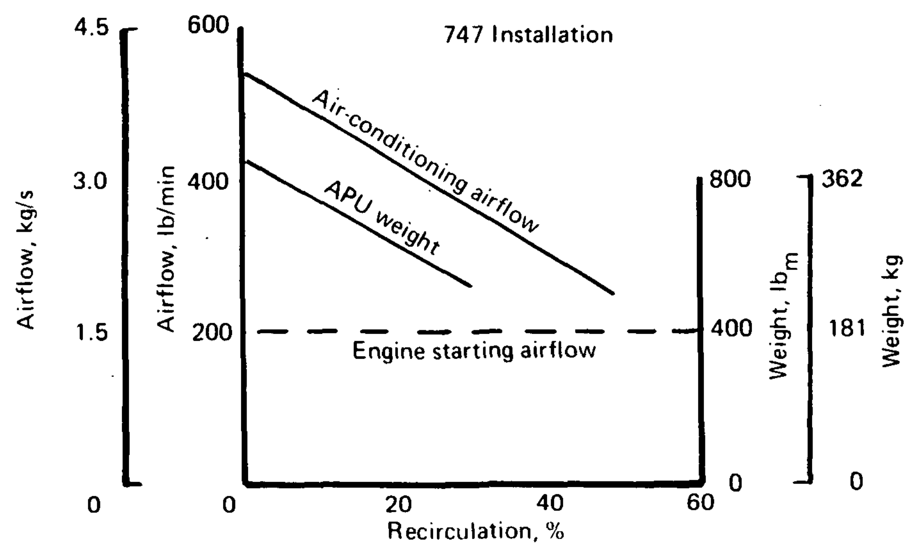


FIGURE 57.— CABIN AIR RECIRCULATION DEVELOPMENT

coupled with a cabin air recirculation system would also reduce the quantity of airflow demanded. This type of system is probably not practicable and would require considerable exploration and development to determine if it is at all economically feasible.

- Cooling cycle evaluation—The cooling cycle used to condition cabin supply air is directly affected by the SPS and the pneumatic air source. For instance, the use of an integrated engine generator and electrically driven cabin air compressor could make the selection of a vapor cycle cooling system mandatory. If the cabin air source were an engine-shaft-driven compressor, the air cycle cooling function could be integrated with the shaft-driven compressor (powered bootstrap) to eliminate the requirement for an additional rotating component. The main areas requiring development in the cooling cycle system are vapor cycle shaft seals and controls and the powered bootstrap cooling unit.

- Thermal management control of heat sinks—Thermal management of available heat sinks (engine fuel, ram air, engine fan air, etc.) is required to minimize airplane drag and reduce operating penalty. Some of the potential development areas are in fuel-to-air heat exchanger design and establishment of realistic fuel operational design requirements. The use of precooled fuel offers a significant reduction in operating penalty and weight. The attendant problems associated with providing precooled fuel and means of storing it, both on the ground and in flight, require exploration.

Protective systems

- Engine ice protection—The trend toward the use of engine inlet ring splitters to meet the operating noise objectives requires that alternate ice protection concepts be developed to minimize the power requirement and that engine manufacturers establish reasonable ice ingestion limits for the engine.

- Windshield heating—The trend toward larger and more stringent airplane windshield field-of-vision requirements indicates the need for the development of a three-phase electrical heating system or alternate forward viewing systems.

  Current practice is to use single-phase electrical heating. The large window area on advanced transports requires a heat input in excess of 1 kW. Single-phase loads of this magnitude can prevent proper distribution of loads between phases.

- Wing icing—The operational characteristics of commercial long-range jet transports are sufficiently different from those of piston engine airplanes to require a complete review of wing anti-icing requirements. An extensive review of probability of icing encounters, degree of icing, and effect on operation is recommended. The FAR should also be updated to include the latest available meteorological data.

Powered Wheel System

Potential Payoff

Development of a powered wheel system to provide ground maneuver capability without the use of main engines or tow tugs has several benefits (fig. 58). Ground air pollution would be reduced 65% to 80%. Ground noise could be reduced up to 7 dB (aft) and 38 dB (forward) when measured on "A" scale at 61 m (200 ft) radius. Elimination of the jet wake in the terminal area would be a very positive safety benefit. Preliminary study shows that reductions in ground operation costs (reduced ground equipment and taxi fuel) of the order of $30 000 per airplane per year could be expected to offset costs of a powered wheel system; however, some weight penalty would be expected.

State of Readiness

The conceptual studies indicate that the system is technically feasible. However, its practical application requires study to determine specific methods and a more precise understanding of its effect on airplane design and airline economics.

Recommended Action

A three-phase approach is recommended, with each succeeding phase dependent upon results of the completed phase. The first phase would a 1-year study and should include participation of two or three suppliers active in design of hydraulic, pneumatic, and electrical hardware, and involvement of one or more airlines in determination of economic potential.

Phase I results would be a definite decision as to the practicality of powered landing gear wheels and a recommendation as to the advisability of investigating nonairborne ground maneuver systems.

If phase I proves continuation of the powered wheel to be advisable, hardware development and demonstration would follow. The commercial jet transport, model 737, is a practical vehicle for use as a demonstrator. The torque and horsepower requirements are not extremely high, the landing gear is a relatively simple tricycle type, a suitable auxiliary power unit is available, and the airplane operating cost is low. All phases should be funded by the government, because there is little economic motivation for private industry funding.

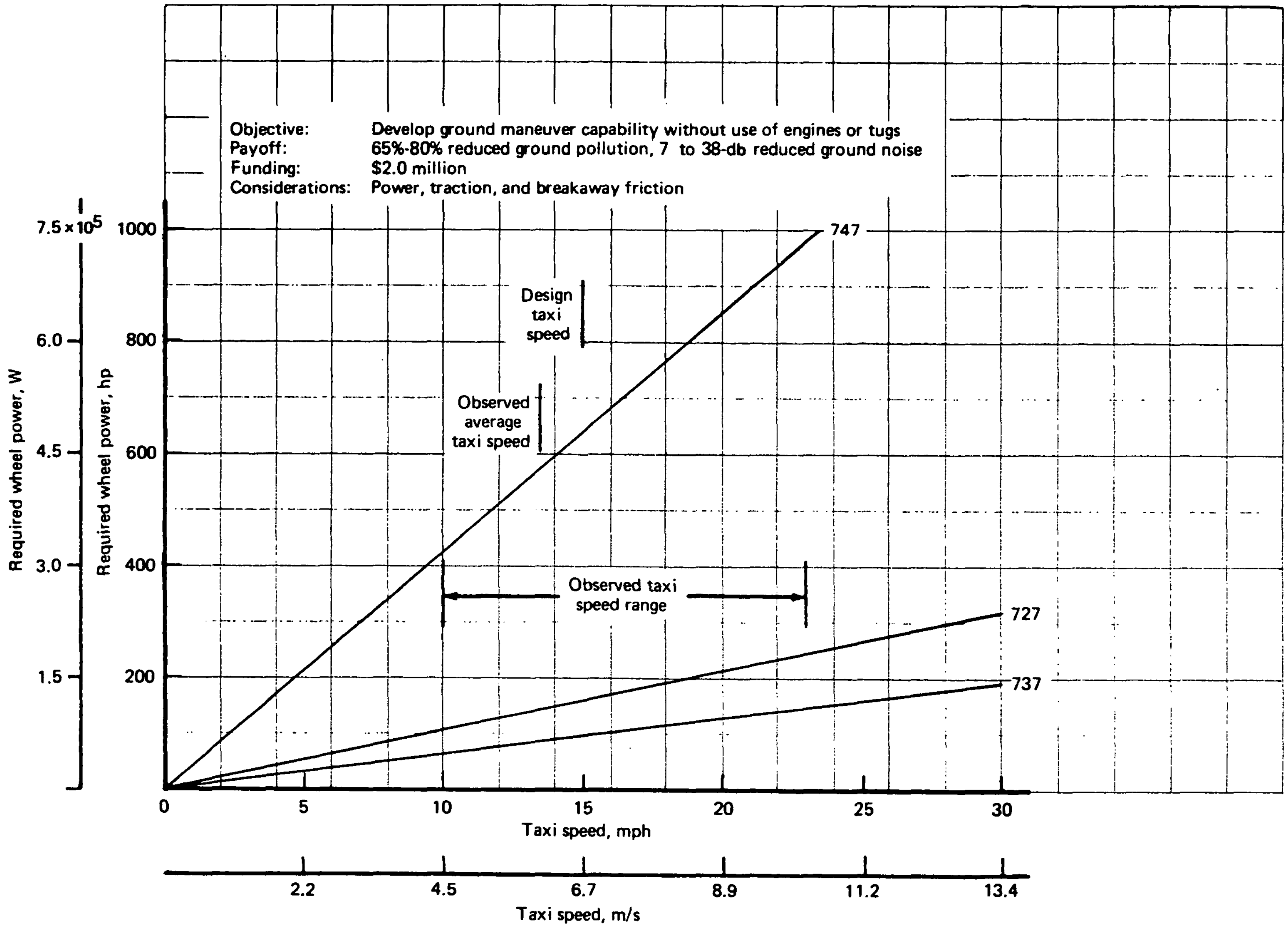


FIGURE 58.—WHEEL POWER REQUIREMENTS

Cost and Schedules

The initial study phase is estimated to cost approximately $200 000 over a 1-year period. The total program estimated cost is $1.4 to $2.6 million over a 3-year period. The plan is shown in figure 59.

Auxiliary Power Unit

Potential Payoff

The dedicated APU SPS configuration showed a cycled TOGW reduction of 0.52% or 845 kg (1860 lb). Further significant gains could be realized through a more in-depth integration of the APU with the airplane systems. This includes extension of the dedicated, isolated APU capability and also consideration of other integrated APU concepts. An extensive study is required to establish the quantitative gains.

State of Readiness

Current industry procedure has been to design the airplane systems to be independent of the APU and then to offer the APU as an optional item at a significant weight penalty to provide airplane ground self-sufficiency. In some instances the APU is used for limited operation in flight. APU installations require secondary power components such as generators and hydraulic pumps. These are in addition to the propulsion-engine-driven components. Pneumatic functions such as air conditioning and engine starting are accomplished inefficiently because of energy losses in air compression, transmission, and expansion.

The airline industry presently looks on the APU as a necessary operational item, and as such it should be integrated in a manner that would provide the optimum system installation. Very limited studies have been conducted on means of integrating the APU as an engine starter (jet fuel starter) or emergency power unit.

For the SPS for the B-1, means are being developed to provide both engine start provisions and electric and hydraulic power sources on a standby basis from either one or both APUs. Various helicopter systems have small standby APU/starter secondary power systems.

Recommended Action

It is recommended that studies be initiated to determine specifically how the APU might be integrated into the SPS to achieve maximum utilization with minimum weight and operating cost.

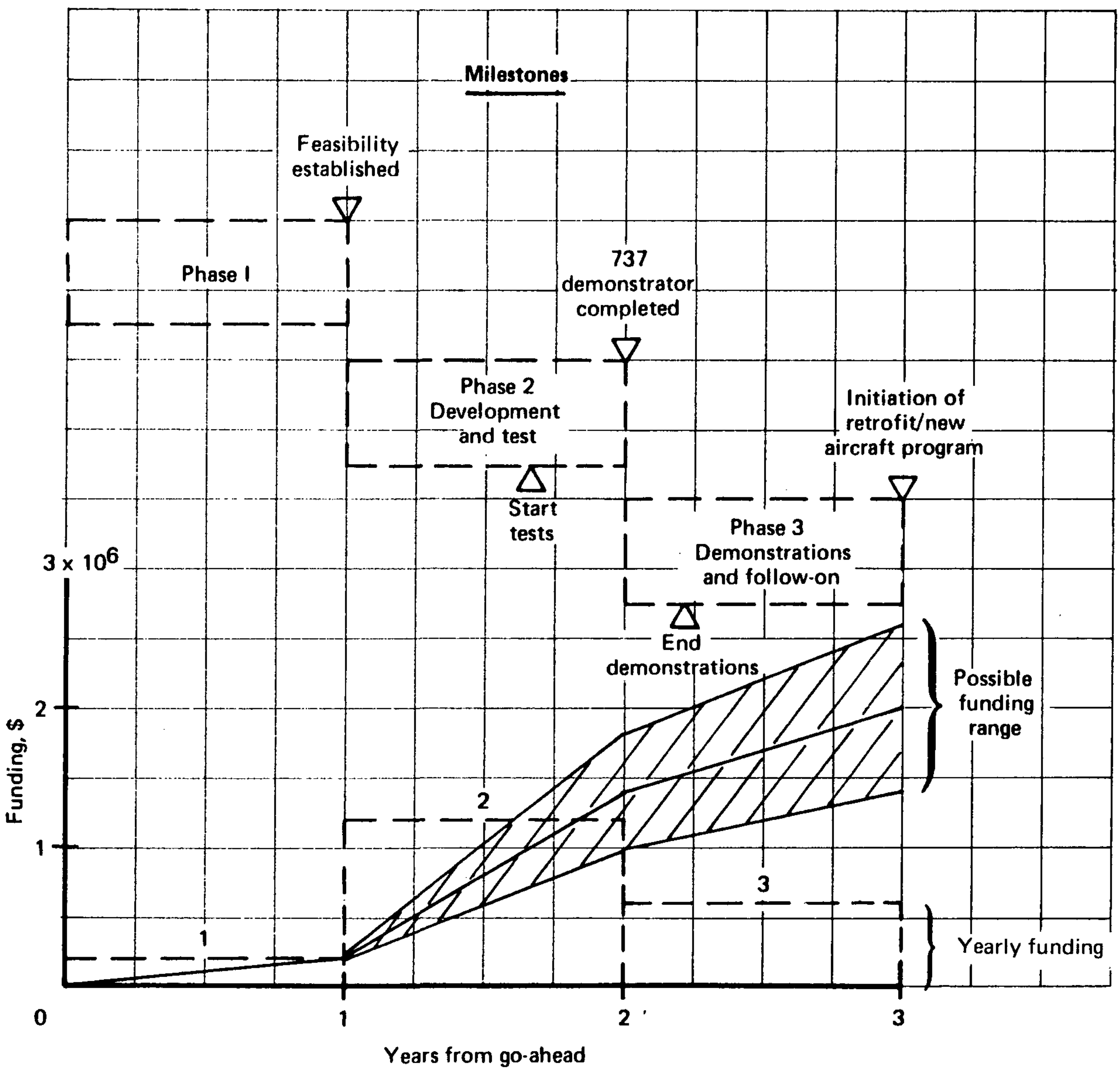


FIGURE 59.—POWERED WHEEL SYSTEM PROGRAM PLAN

These studies should be based on advanced state-of-the-art technology. Phase I would be a 1-year program with a cost of approximately $200 000. It will include analysis and preliminary design installation studies to compare system configurations on a weight, noise, airplane balance, and operating penalty basis. Hardware concepts requiring development and feasibility will be defined.

Phase II would be a 2-year program with a cost of $500 000 per year. It would include the release of problem statements to equipment suppliers for detail definition of integrated units and the design, fabrication, and test of appropriate hardware.

Cost and Schedules

The program plan is shown in figure 60 and will require 3 years with a total funding of $1.2 million.

Internal Engine Generator

Potential Payoff

The internal engine generator/starter (IEG) concept permits elimination or a reduction in size of the engine gearbox or engine-mounted accessories, resulting in engine frontal area reduction. Expected payoff per airplane for configuration described in part II is as follows:

• Drag reduction	2.3% (2.7% TOGW reduction or 4300 kg (9450 lb))
• Equipment weight change	32/55 kg (70/120 lb)
• Generator reliability– 1.8×10^8 to 3.6×10^8 sec (50 000 to 100 000 hr) MTBF	3:1 improvement
• Maintenance	Improved accessibility ("clean engine")

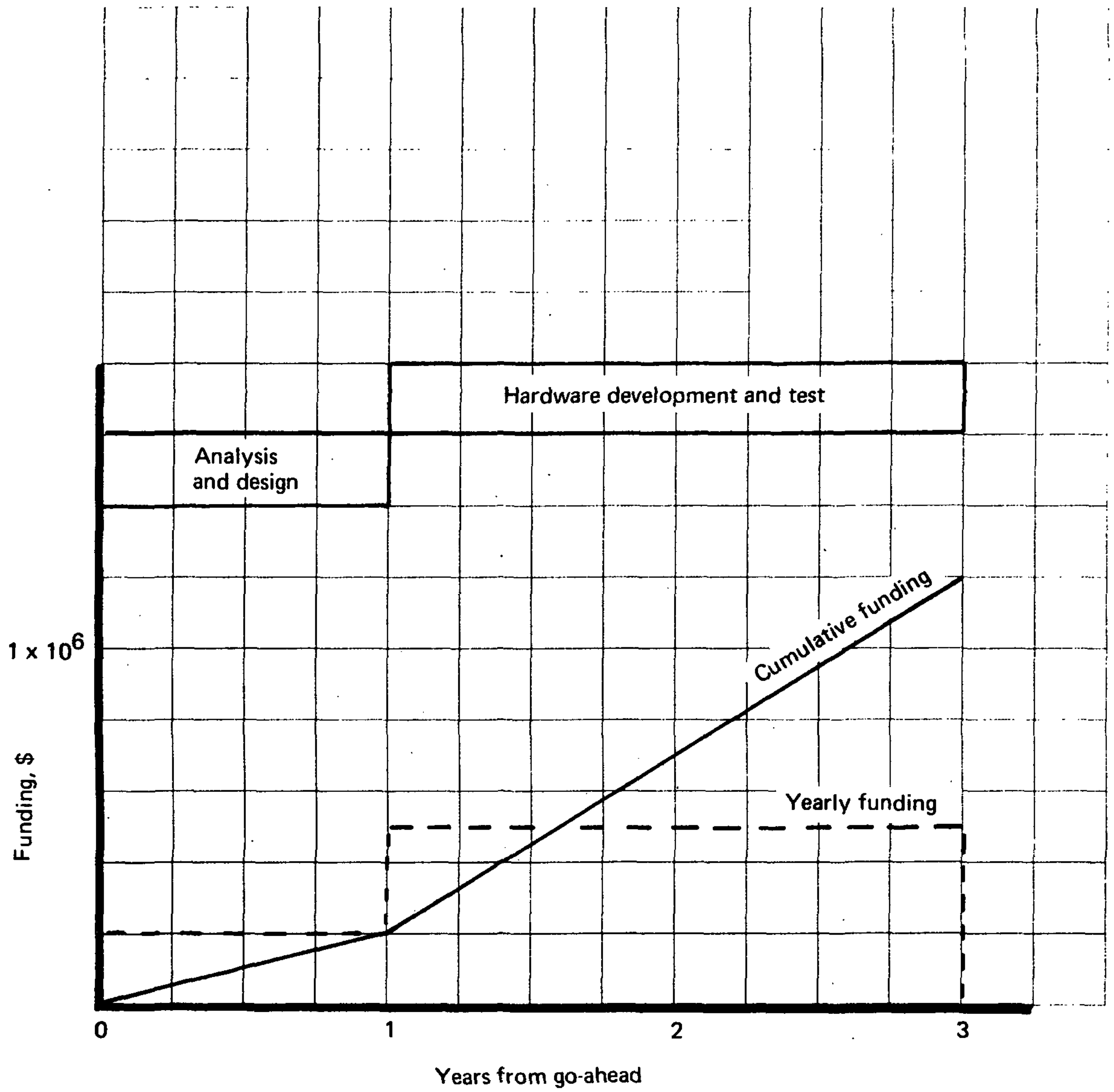


FIGURE 60.—APU INTEGRATION STUDY PROGRAM PLAN

State of Readiness

Preliminary studies have been conducted to determine concept feasibility. The results of part II of this study have confirmed early estimates and have provided added feasibility assurance. Potential drag reduction, impact of equipment weight on the airplane, and the electrical capacity required of a generator/starter developed as an integral part of the propulsion engine have been evaluated. The government and industry have been active in this preliminary work.

The need to reduce engine frontal area and attendant drag on the U.S. supersonic transport prompted a design which utilized a remotely located engine-shaft-driven accessory drive system mounted in the wing. A variable-speed, constant-frequency (VSCF) system was developed for this application. The IEG concept involves mounting the generator rotor and stator as an integral part of the engine and use of the VSCF concept to obtain constant-frequency electrical power and also to program voltage and frequency for engine starting. The studies indicate that the IEG could be mounted internally in the engine, provided it is integrated into the basic engine design.

Reliability is a major consideration, and a mean-time-between-failure rate of 1.8×10^8 to 3.6×10^8 sec (50 000 to 100 000 hr) should be achieved. Electrical and propulsion industry engineering representatives have expressed confidence in attaining this reliability goal.

Recommended Action

Figure 61 describes the recommended tasks and shows estimates of funding required to successfully complete an IEG development program. Phase I will complete a detailed feasibility analysis using data from prototype equipment tests. Demonstration of engine starting capability is one of the more important tasks for phase I. Airline support equipment compatibility will be determined. Concurrent development of both the cycloconverter and dc-link IEG concepts is recommended in the prototype phase, and the most promising will be selected for follow-on work. The analysis will cover the following technical risk areas:

- Effect of the IEG on new jet engines under consideration (mounting on high-pressure rotor shaft, number of bearings and bearing tolerances, engine assembly, oil system, etc.)
- Generator reliability
- Generator rotor stresses
- Generator air gap variation considerations and the effects of unbalanced magnetic forces on the rotor and engine shaft

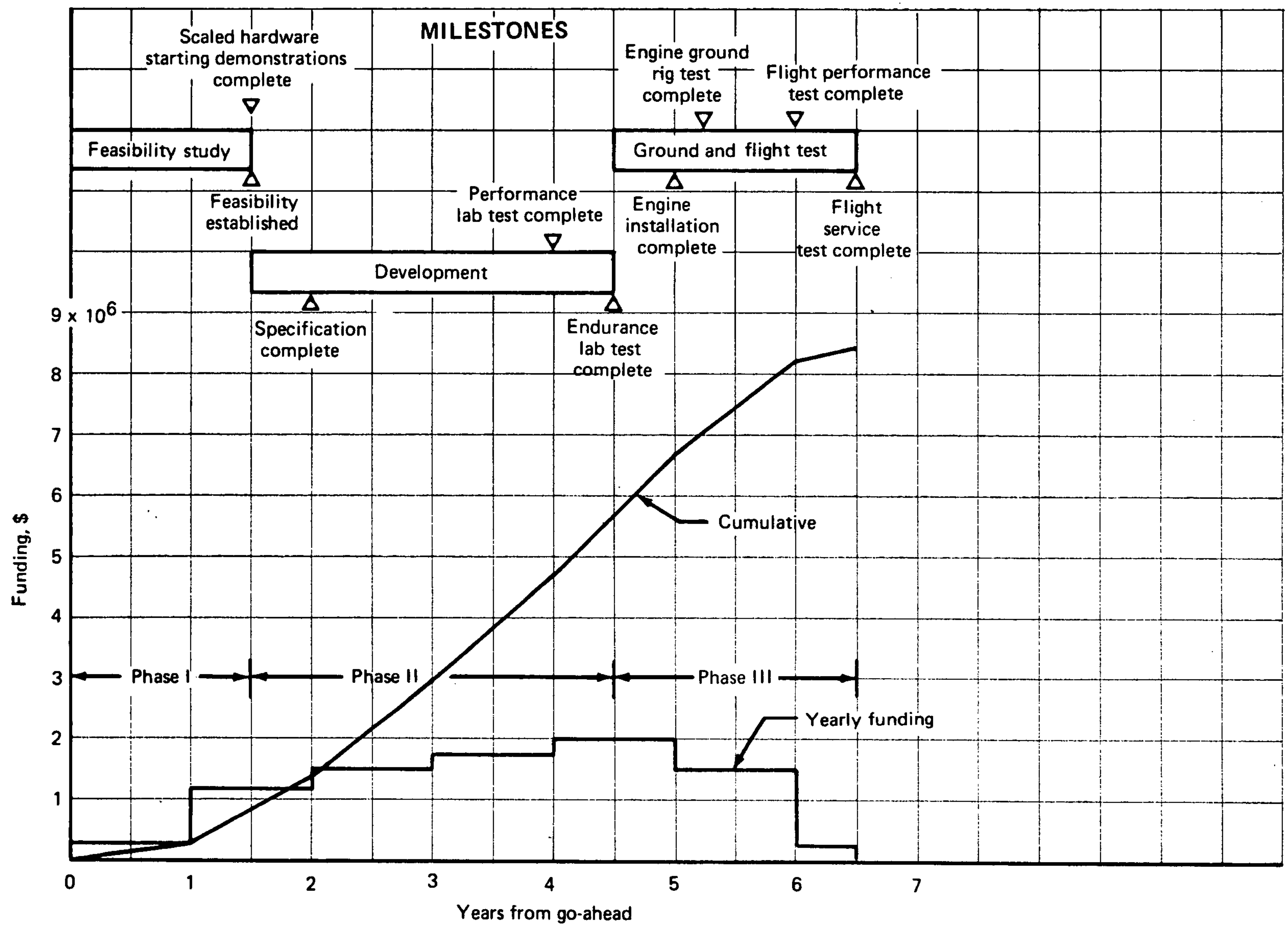


FIGURE 61. —INTERNAL ENGINE GENERATOR PROGRAM PLAN

- Generator feeder and control wire installation in the engine
- Generator compatibility with engine environment
- Generator voltage and corona
- Effects of stray magnetic flux
- Development of converter and controls
- Power quality and electromagnetic interference during engine starting

Phase II will establish generator/engine interface requirements, determine specifications for an IEG laboratory test system, and select suppliers to build full-scale IEG systems (135-150 kVA rating). Laboratory tests of these will be conducted, performance evaluated, and improvements incorporated.

Phase III tasks are to build flight-rated IEG systems and install the generators in new jet engines under development. Engine/IEG ground rig tests will be conducted. Engines incorporating the IEG systems will be installed in a test airplane and flight tests conducted.

Cost and Schedule

The development program outlined will require 6.5 years and a total funding of $8.5 million.

Accessory Gearbox Installation in Engine Fan Duct Bifurcation

Potential Payoff

The bifurcation mount SPS installation showed a cruise drag reduction of 1.7%. This is equivalent to a cycled TOGW reduction of 2% or 3100 kg (6800 lb).

State of Readiness

This concept uses proven hardware components. Industry practice is to install the engine and airplane accessories on a gearbox located either on the engine core or the fan case. In-house study has shown that engine frontal area can be reduced by installing the gearbox and accessories in the lower fan duct bifurcation.

Recommended Action

It is recommended that a program be conducted to establish packaging criteria and requirements to meet maintainability objectives. The program will consist of two phases.

Phase I will be a 1-year program with a cost of approximately $300 000 to perform analysis, prepare design layouts, coordinate with engine/airline/equipment suppliers, and recommend configurations for two engine sizes.

Phase II will be a 6-month program with a cost of approximately $200 000 to fabricate two engine mockups and confirm maintainability aspects of the installation.

Cost and Schedules

The program plan is shown in figure 62 and will require 1.5 years and a total funding of $500 000.

RECOMMENDATIONS

All of the research and development items presented are worthy of accomplishment. However, several factors (cost, limited interest, etc.) will affect the ability or desire to accomplish all items, and the need to prioritize is obvious. The order of preference, with reason as necessary, is as follows:

- Internal engine generator

  Reason: Preliminary study indicates good potential payoff. However, risk is high and substantial lead time is required to complete development. Industry will probably defer IEG approach to one with less risk unless government support is available.

- Vapor cycle cooling, electric motor/hydraulic pump (30-gpm rating), and integrated flight control actuator packages

  Reason: Any or all of these items would be used in conjunction with an IEG. Development may lag IEG work but should generally continue in parallel.

- Hydraulic control valve erosion reduction

- Cabin air recirculation

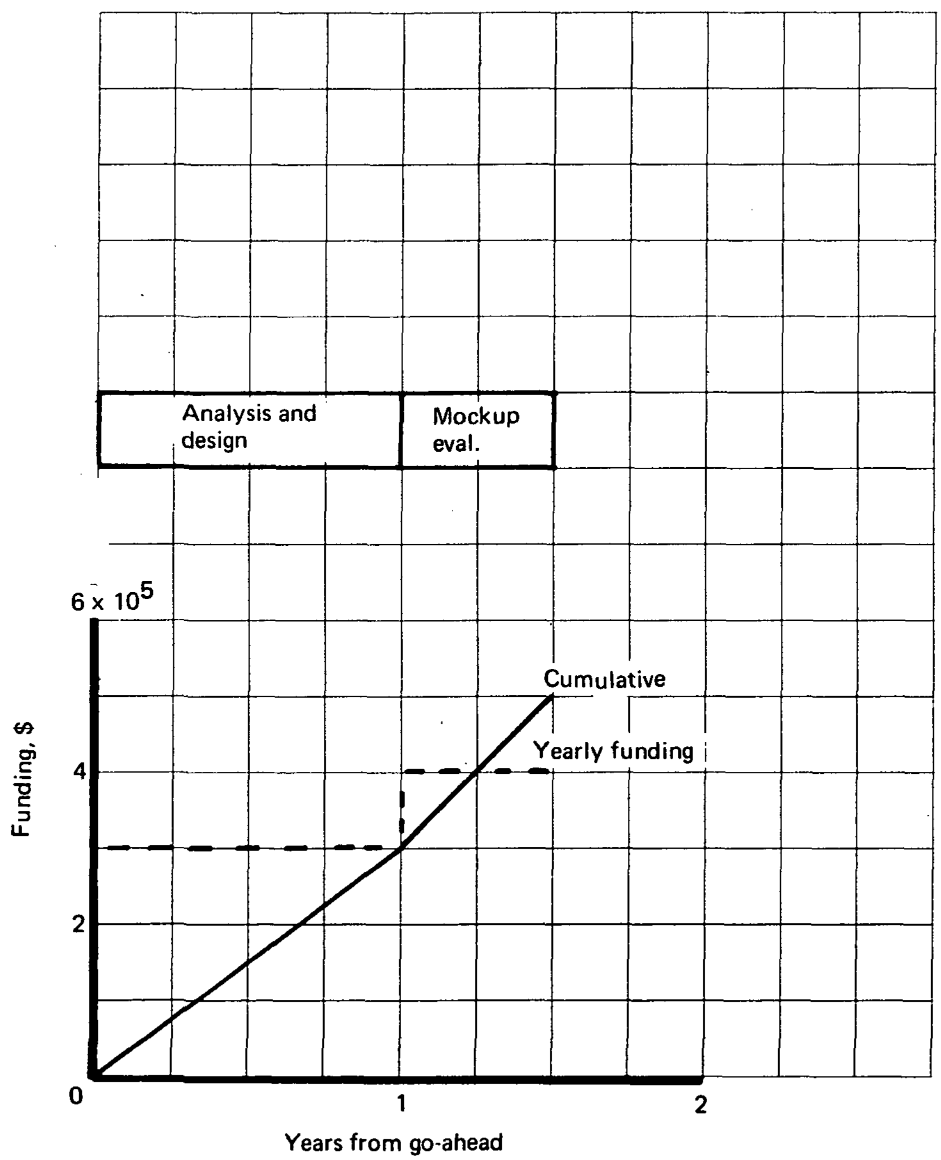


FIGURE 62.—ACCESSORY GEARBOX INSTALLATION IN ENGINE FAN DUCT BIFURCATION

- Electric wiring
- Other

Reason: In general, the development of these items will provide the building blocks to an optimized integrated secondary power system. Inasmuch as each new airplane must be evaluated on an individual basis, it is imperative that substantial building block data be available prior to the preliminary design of the airplane.

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