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**Scientific and Technical
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Summary

A study has indicated that a proposed new approach to secondary power on transport aircraft can result in a reliable, lightweight, and efficient system that can reduce aircraft empty weight by 10 percent and fuel consumption by 9 percent. The approach involves a totally new concept for aircraft power systems that is derived largely from space power technology.

A 200-passenger, twin-engine transport was selected as the baseline aircraft. A 20-kHz, 440-V, sine-wave power distribution system concept was selected to provide all secondary power on the aircraft. Lower frequency power would be synthesized from the 20 kHz to meet the load requirements. An electrically powered flight control system was integrated with the power system. The other aircraft systems were replaced or modified to make them compatible with the new power system. The system weight reductions were calculated, and the aircraft was resized to take advantage of these weight reductions.

A total of 2950 kg (6500 lb) was removed from the aircraft systems. When the aircraft was resized for lower system weight and no engine bleed, the total weight reduction was more than 7700 kg (17 000 lb). The fuel reduction for the design mission was approximately 2270 kg (5000 lb), or 9 percent of mission fuel.

Introduction

Present transport aircraft subsystems evolved during the years when fuel was relatively inexpensive. As a result their design was not optimized for minimum fuel consumption. A transport aircraft normally uses three types of secondary power: hydraulic, pneumatic, and electric. Each of these power systems must be sized to handle its maximum load with some margin for safety. Many loads, such as flaps and landing gear, are intermittent and usually occur only at the beginning and end of the flight. Anti-icing is hardly ever used on large transport aircraft since they normally fly above icing conditions. As a result, both the hydraulic and pneumatic power systems are underutilized during most of the flight. Using a single type of power for all aircraft secondary functions would allow load sharing. With load sharing the duty cycle of the secondary power system could be optimized over the entire flight. This would result in a smaller capacity power system and a significant weight reduction.

Current transport aircraft engines have low to moderate bypass ratios: a large percentage of the mass airflow passes through the engine core. The amount of bleed air used to power the anti-icing and environmental control systems is a fairly small percentage of the total core flow. However, the newer, more efficient engines being developed for future transport aircraft have a much higher bypass ratio. This will greatly reduce, or eliminate, the engine bleed air available for anti-icing systems and air-cycle environmental control systems.

Because avionics, lighting, and the galley require electric power, it is the only type of power that can supply all of the aircraft loads. Therefore electromechanical actuators must be substituted for hydraulic ones and environmental control and icing protection must be powered electrically instead of pneumatically.

To determine the effect of an advanced power system on a transport aircraft, it was first necessary to select a baseline aircraft. Then an advanced power system concept was identified that would satisfy all of the requirements of the aircraft. Next an electrically powered flight control system was integrated with the secondary power system. Then the ice protection system, the environmental control system, and other miscellaneous systems were modified to be compatible with the new power system. The system weight reductions were calculated, and the baseline aircraft was resized to take full advantage of the weight reductions.

This report documents the conceptual design of the advanced all-electric secondary power system and the resulting weight and fuel reductions made possible by that design.

The Boeing Commercial Aircraft Company was very helpful in providing information on the aircraft and its subsystems used as the baseline in this study. Without their generous assistance the value of this study would have been substantially reduced.

Study Approach

The basic intent of this study was to evaluate the weight and fuel savings associated with a transport aircraft configuration in which all secondary power is supplied electrically. The first step in the study was to define a baseline aircraft and engine. This baseline design was then modified by replacing the baseline secondary power system with an advanced electric power generation and distribution system. For the purposes of this study the baseline secondary power system was defined as the generation and distribution equipment for the hydraulic and electric subsystems and the distribution equipment for the pneumatic subsystem. The advanced electric system was designed to support all of the functions previously supplied by the baseline secondary power system.

With all secondary power being furnished by an electric power system, other aircraft systems had to be modified to accept electric power. The flight control system required the most substantial modifications. The hydraulic flight control actuators, their associated servovalves, and the mechanical cables and pulleys (for control input signals) were eliminated from the baseline design. These components were replaced with electromechanical actuators, their associated load receivers, and a digital fly-by-wire control system. The actuators were replaced on a one-for-one basis to keep the same level of redundancy as the original flight control system. Also, the electromechanical actuators were connected to the same mechanical structure as were the baseline actuators. No attempt was made to optimize the aerodynamic design of the aircraft that might result from a modified electromechanical actuator installation.

The baseline air-cycle environmental control system (ECS) was the primary consumer of pneumatic power. It was replaced with an electric-motor-driven vapor-cycle system for temperature control and an engine-gearbox-driven compressor for cabin pressurization. The other pneumatic power user, the baseline anti-icing system, was replaced with an electroimpulse deicing system (refs. 1 and 2).

Besides the major systems a number of other aircraft systems were changed or eliminated to accommodate the new electric secondary power system. The auxiliary power unit (APU) was removed since most of its functions could be performed by ground utility power. The air starter was also eliminated since the starting function in the new power system design is provided by the power system generators operating as motors. The landing gear, brakes, and thrust-reverser actuators were all changed from hydraulic to electric actuation.

All of the electric loads in the baseline aircraft, such as avionics, lighting, and small motors, were left unchanged even though the advanced power system could provide weight reductions in such areas as avionics power supplies. These reductions were considered too fine a detail for the level of this study. The only digital data-bus systems used in the advanced design were for the flight controls and the power system control. The remainder of the existing information transmission systems were left intact in order to limit the study benefits to those associated with the electric power system.

The total weight savings from these modifications to the baseline aircraft systems was determined. The airplane was then resized, for the same mission and payload, to take advantage of the reduced systems weight and the removal of the customer bleed from the engine. Both the baseline and modified aircraft were flown over the design mission to determine fuel usage. The weights and fuel consumption for this resized "all electric" aircraft were then compared with those for the baseline aircraft.

Baseline Aircraft

Aircraft Selection

The baseline aircraft selected for this study was a 200-passenger, twin-engine transport, very similar to the Boeing 767. This aircraft was selected for four basic reasons: passenger size, number of engines, technology level, and information availability.

The 200-passenger size was selected to fill a gap left by previous studies (refs. 3 and 4), which investigated large and small aircraft but did not study medium-size aircraft. Those studies indicated significant fuel savings for large (350 and 500 passenger) aircraft using all-electric secondary power. However, for small (30 and 50 passenger) commuter aircraft, the fuel savings were negligible. It was not apparent from these studies what the fuel saving would be for an intermediate-size aircraft, such as the 200-passenger transport selected for this study.

Because of the rising popularity of twin-engine transports they are becoming responsible for a larger portion of aircraft fuel consumption. This makes the twin-engine configuration very important in a study concerned with reducing fuel consumption. Previous studies were of three-engine configurations.

Since this study addresses a 1990's type of aircraft, it would be desirable to have a baseline with 1990's technology throughout. However, just exactly what the various technologies will be is open to speculation. Also the weight and performance characteristics of the various secondary systems on the aircraft would not be defined. Therefore, to have a baseline with accurate information available for the various systems, it was decided to select the latest technology aircraft presently in production. The Boeing 767 was judged to meet this criterion.

Once the baseline aircraft was selected, consideration was given to the engine. The performance of the engine and its sensitivity to customer bleed and horsepower extraction would significantly affect the study results. Therefore it was highly desirable to have an engine representative of 1990's technology. The NASA/General Electric energy efficient engine (E^3) (ref. 5) is the most advanced-technology, fuel-efficient transport engine for which engine test data are available. As a result it was selected as the engine for the baseline aircraft.

Aircraft Configuration

The baseline aircraft (fig. 1) is a 200-passenger, twin-engine, wide-body design. The GASP computer code (ref. 6) was used for weight sizing and performance and mission analysis modeling of all designs for this study. The GASP code uses an integrated approach that ensures that the results contain design interactions. For example, a change in wing loading affects wing area, tail size, lift, drag, propulsion system size, cruise altitude, structural weight, range, and other parameters. Any net effect may be small, but nevertheless it is determined numerically regardless of its magnitude. The baseline design was first configured on GASP and sized to "match" the Boeing 767. Then the model was modified by using scaled NASA/GE E^3 engines to replace the current 767 engines. The unscaled E^3 engine has the following sea-level static (SLS) characteristics:

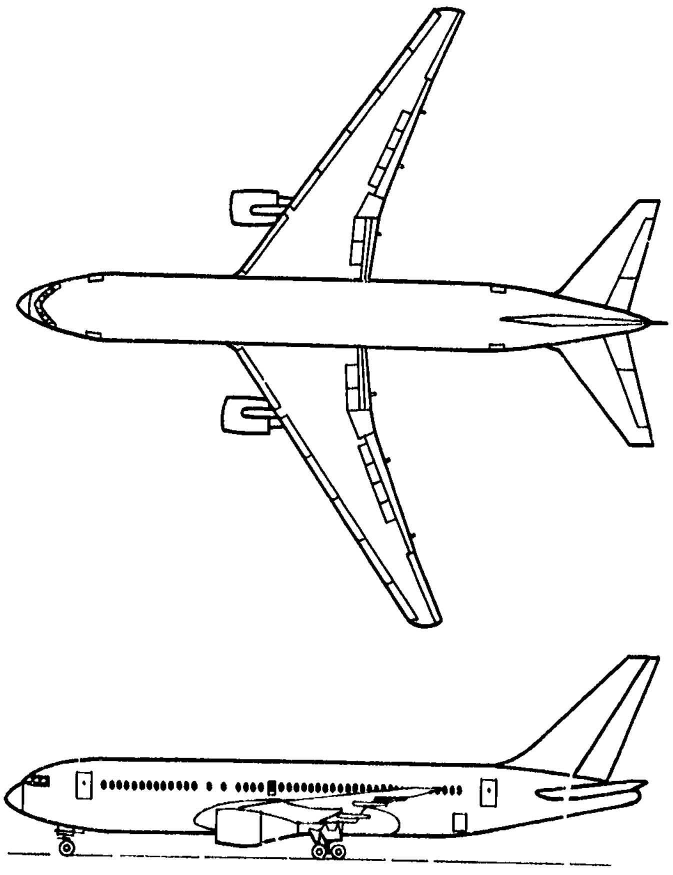

Figure 1.—Baseline aircraft.

Net thrust, installed, kN (lb)	155.9 (35 040)
Bypass ratio	7.3
Overall pressure ratio	29
Total engine airflow, kg/s (lb/s)	567 (1250)
Thrust specific fuel consumption (TSFC), installed, k/N hr (lb/lbf hr)	0.0306 (0.30)

These installed engine data were computed by assuming a 11.34-kW (152-hp) extraction to drive the electric generators, hydraulic pumps, and engine accessories and a 0.68-kg/s (1.5-lb/s) midstage compressor bleed to drive the ECS.

The baseline aircraft design was then resized to match the performance of the new E^3 engines with an aircraft capable of 3593-km (1940-n mi) range with a 17 578-kg (38 760-lb) payload. The baseline engine size was scaled to allow the aircraft to perform the design mission (fig. 2). The design mission of 3593 km (1940 n mi) was flown at 10 668 m (35 000 ft) at Mach 0.8. Reserve fuel included a 1-hr extended cruise and a 370-km (200-n mi) alternate-field distance. The engines were sized to meet the following requirements:

Takeoff distance over a 10.7-m (35-ft) obstacle, m (ft)	1829 (6000)
First-segment one-engine-out climb gradient, percent	>0
Second-segment one-engine-out climb gradient, percent	>2.4
Engine-out service ceiling, m (ft)	>6096 (20 000)

All engine-out sizing points were assumed to require an additional 1.8-kg/s (4-lb/s) bleed from the remaining engine for anti-icing and ECS operations. Airport elevation was assumed to be 1524 m (5000 ft)

The final baseline aircraft geometry is described in table I and a weight breakdown is given in table II. The design mission is summarized in table III.

Secondary Power Systems and Auxiliary Power Unit

The secondary power systems on the baseline aircraft include the hydraulic, electric, and pneumatic systems. The APU is intended primarily for ground operations, but it can provide emergency secondary power if necessary.

Hydraulic.—The hydraulic generation and distribution system (fig. 3) is a three-channel system using a combination of engine-driven pumps (EDP), electric-motor-driven pumps (EMP), and an air-turbine-driven pump (ATDP). There is also a ram air turbine (RAT) pump for emergency conditions. Using three independent channels and a variety of pumps makes catastrophic failure highly improbable. The maximum output of the total system (not including the RAT) is 176 kW (236 hp). The generation and distribution equipment, including accumulators but not including the APU or any air

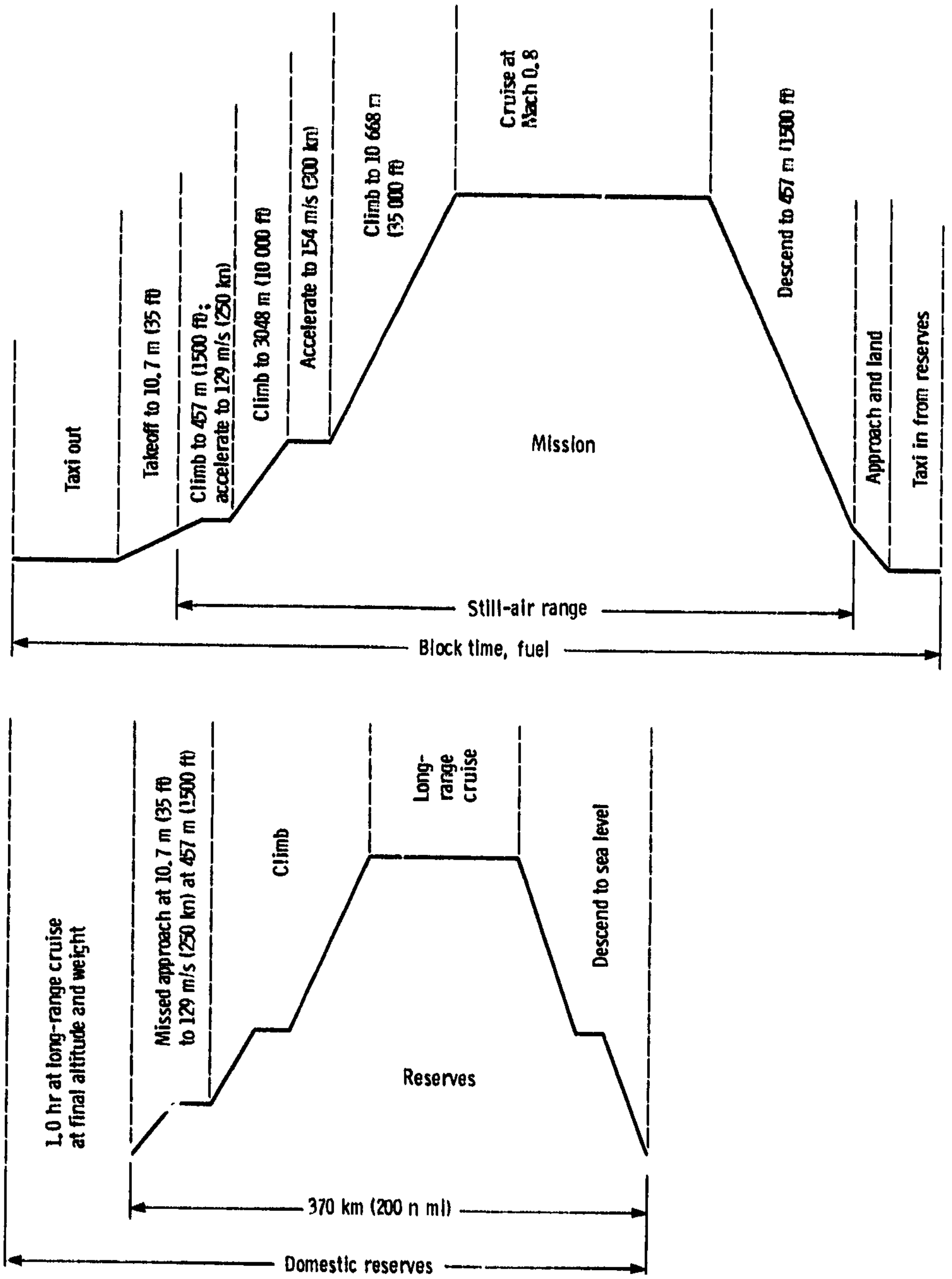

Figure 2.—Mission profile.

TABLE I.—DESCRIPTION OF BASELINE AIRCRAFT

Takeoff gross weight, kg (lb)	120 601 (265 933)
Sea-level static thrust per engine, kN (lb)	201.6 (45 319)
Wing area, m^2 (ft^2)	270.6 (2912.6)
Wing span, m (ft)	46.8 (153.7)
Wing 1/4-chord sweep, deg	0.55 (31.5)
Wing aspect ratio	8.11
Wing taper ratio	0.267
Wing thickness ratio, root	0.151
Wing thickness ratio, tip	0.103
Wing mean geometric chord, m (ft)	6.4 (21.1)
Horizontal tail area, m^2 (ft^2)	53.8 (578.8)
Horizontal tail volume coefficient	0.898
Horizontal tail aspect ratio	4.00
Vertical tail area, m^2 (ft^2)	64 (688.8)
Vertical tail volume coefficient	0.100
Vertical tail aspect ratio	0.67
Body length, m (ft)	46.3 (152)
Body width, m (ft)	5 (16.4)
Seating abreast	7
Number of aisles	2

TABLE II.—WEIGHT BREAKDOWN FOR BASELINE AIRCRAFT

	Weight	
	kg	lb
Propulsion group:		
Primary engines	8 788	19 378
Engine installation	2 095	4 620
Fuel system	539	1 188
Structures group:		
Wing	15 402	33 963
Horizontal tail	1 468	3 237
Vertical tail	2 082	4 590
Fuselage	15 579	34 352
Landing gear	6 995	15 425
Primary engine section	2 671	5 890
Flight controls	1 874	4 132
Fixed useful load	8 029	17 705
Fixed equipment:		
Auxiliary power unit	676	1 490
Instruments	472	1 040
Hydraulics	1 295	2 855
Electric	739	1 630
Avionics	721	1 590
Furnishings	7 914	17 450
Air-conditioning	975	2 150
Anti-icing	186	410
Pneumatics	354	780
Airframe lights	113	250
Operating-empty weight	78 966	174 125
Payload	17 578	38 760
Fuel	24 057	53 048
Gross weight	120 601	265 933

TABLE III.—DESIGN MISSION

Mission segment	Segment fuel usage		Segment range		Segment time, min
	kg	lb	km	n mi	
Taxi	346	763	0	0	14
Takeoff	153	338	1.9	1	1
Climb/acceleration	2 359	5 202	209	113	17
Cruise	13 598	29 985	3122	1686	219
Descent/land	943	2 080	259	140	20
Total	17 400	38 368	3593	1940	271

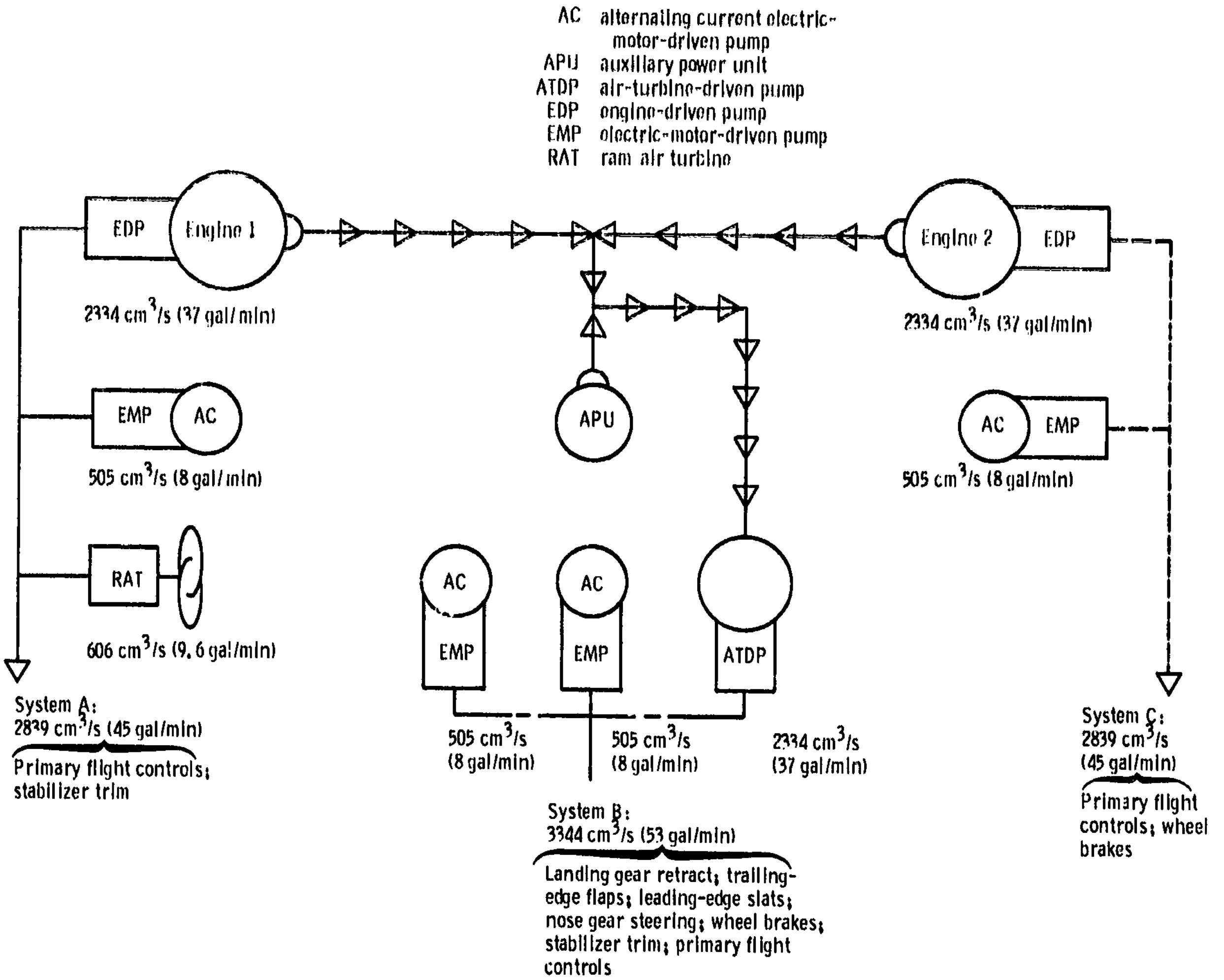

Figure 3.—Baseline hydraulic generation and distribution system.

ducting, weighs 1295 kg (2855 lb). The utilization factor of this system at cruise is 0.19 (cruise power divided by rated power). The utilization is low for most of the flight profile because the major power demand on this system is the raising and lowering of the landing gear, flaps, and slats.

Electric.—The electric generation and distribution system (fig. 4) is basically a two-channel system with a generator mounted on each engine. A battery serves as a third power source as well as emergency power. A third generator is mounted on the APU for ground electric power and for flight backup if one of the two main generators fails. The combined rated output of the two main generators is 180 kW (240 hp). The generation and distribution system weighs 739 kg (1630 lb) not including the APU, its generator, or any APU-associated equipment.

Pneumatic.—The pneumatic distribution system consists primarily of piping for the cowl and wing anti-icing supply, the ECS air supply, the air starter supply, and the APU source lines. The system weighs 354 kg (780 lb). The pneumatic system is not normally rated with a horsepower output. However, converting the bleed air, for both ECS and anti-icing, to an equivalent horsepower for an idle descent condition resulted in a rating of 290 kW (389 hp). This was assumed to be the maximum horsepower load for the bleed system.

The total secondary power system is the sum of these three systems. Therefore the secondary power system weighs 2388 kg (5265 lb), its power output is 646 kW (865 hp), and its utilization factor at cruise is 0.42.

Auxiliary power unit.—The APU can provide both compressed air and electric power. The air is used for engine starting and ground ECS operation. The air can also be used in emergency situations to power the ATDP and thus supply hydraulic power. The electric power is used for ground operations and as an in-flight backup should one of the two primary generators fail. The APU is rated at 448 kW (600 hp) for ground operation and can provide the full 90 kVA of electric power at cruise altitude. The APU total installed weight, including a dedicated battery for APU startup, is 676 kg (1490 lb).

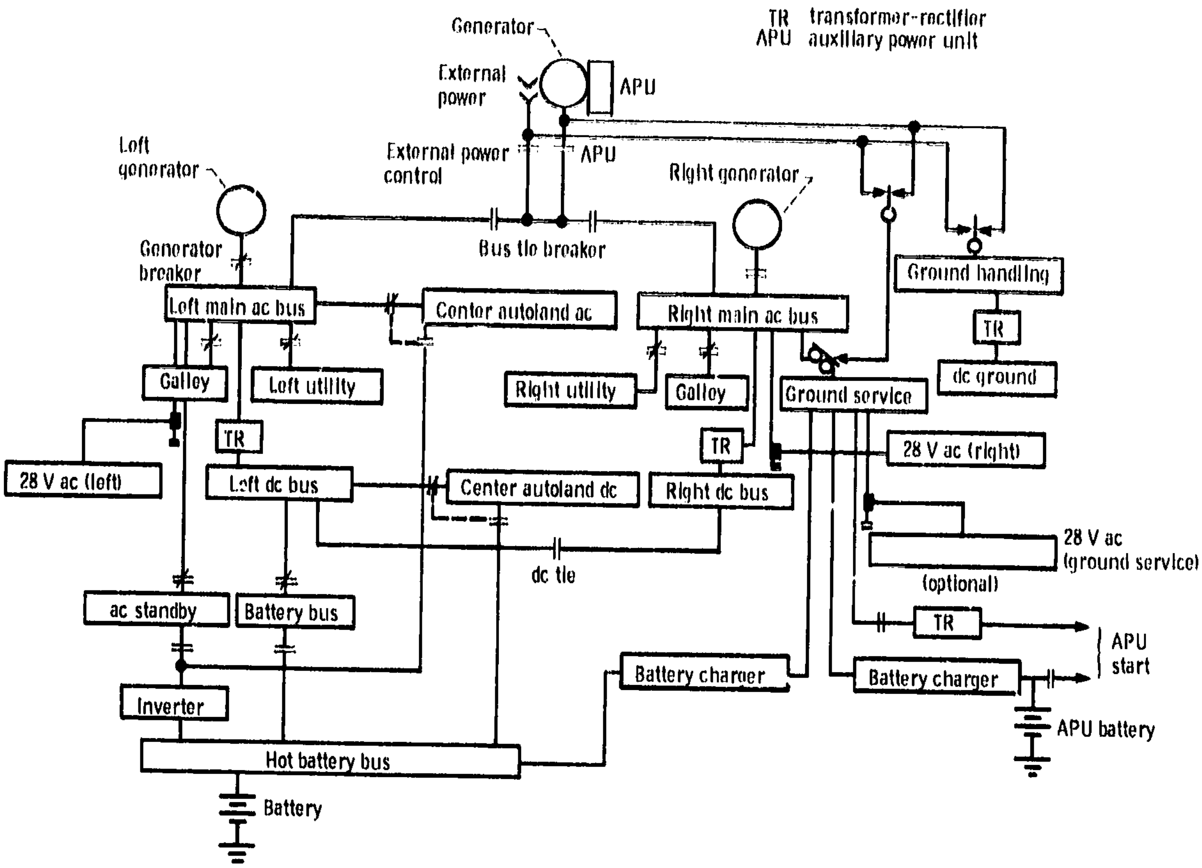

Figure 4.—Baseline electric generation and distribution system.

Flight Control System

The baseline aircraft flight control system (fig. 5) consists of primary and secondary groups. The primary flight control group includes the elevator, stabilizer, aileron, and rudder systems, which provide continuous control of the pitch, roll, and yaw axes. The secondary group includes the flaps, leading-edge slats, and spoilers, which improve performance at a particular flight condition by modifying the basic aerodynamic configuration of the aircraft. The flight control system is a mechanical, hydraulic, and electric system. The primary system consists of cables mechanically linked to the cockpit that allow control of the variable-authority hydraulic actuators located at the control surface. The secondary system consists of both mechanical/hydraulic and electric/hydraulic controls.

The hydraulic flight control system (fig. 6) contains three separate hydraulic systems that supply the power to move the flight control surfaces. Redundant actuators at a flight control surface are powered independently by the hydraulic systems in order to protect against failure. The yaw damping, rollout guidance, feel, trim, and autopilot control functions also are redundant and independently powered by the three hydraulic systems.

The primary flight control system can be divided as follows:

(1) Longitudinal control system—Pitch of the baseline aircraft is controlled by elevators and a variable-incidence horizontal stabilizer.

(2) Lateral control system—Lateral control is provided by two ailerons and six spoilers on each wing.

(3) Directional control system—Directional control in the yaw axis of the airplane is provided by a single span rudder.

The secondary flight control system has the following high-lift devices on each wing:

(1) A double-slotted inboard flap

(2) A single-slotted outboard flap

(3) Five outboard and one inboard leading-edge slats

Each control system is linked to the pilot's control either mechanically by the cable system or, in the case of the spoilers, electrically. The fully powered flight control system requires the integration of the following functions in each control system:

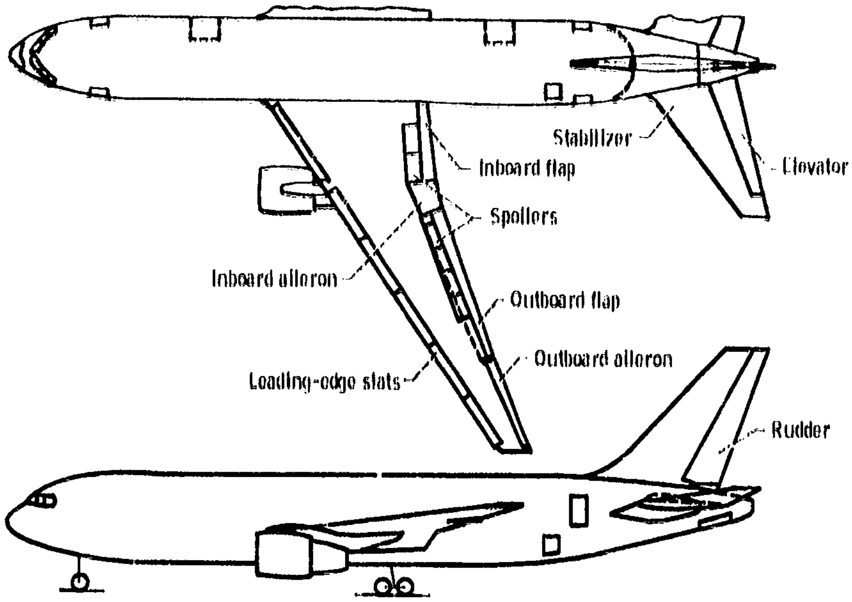

Figure 5.—Baseline flight control surfaces.

(1) Autopilot
(2) Feel
(3) Trim centering
(4) Position information
(5) Controlled surface displacement
(6) Stall warning

Some of these functions are obtained by using mechanical linkages and controlled servoactuators linked to the flight computers. Each subsystem is described in detail and in schematic diagrams in the appendix.

Environmental Control System

The environmental control system uses two air-cycle machines powered by engine bleed. Air is bled from the engine compressor either at an intermediate stage or at compressor discharge. The bleed air passes through a fan air precooler to reduce its temperature and is then fed to the air-conditioning pack. The air is further cooled by ram air in a heat exchanger and then fed to the air-cycle machine. The conditioned air is mixed equally with recirculated cabin air and is then ducted to the flight deck and cabin. Under normal conditions, with both air-conditioning packs operating, the flow rate is approximately 0.28 m^3/min (10 ft^3/min) per passenger. If one pack should fail, the remaining pack can be operated at 165 percent of normal flow. The cabin pressure is regulated by controlling the air outflow from the cabin through an outflow valve.

Anti-Icing System

The same engine bleed air supply that powers the ECS also provides hot air for engine cowl and wing leading-edge anti-icing. After passing through the precooler the bleed air goes to a pressure-regulating valve and is then delivered to a spray pipe in the engine cowl. The wing leading-edge slats outboard of the engine are anti-iced in the same fashion through a separate pressure-regulating valve. Cross ducting allows one engine to anti-ice both sides of the aircraft if necessary.

Miscellaneous

The engines are started with an air turbine starter mounted on the accessory gearbox. The air source can be from a ground cart, the APU, or the other engine. The landing gear actuators, nose wheel steering, brake actuators, and engine thrust reversers are all hydraulically powered.

Advanced Systems to Replace Baseline Systems

The primary system of interest in this study is the advanced electric power system that replaces the baseline hydraulic, electric, and pneumatic systems. When the type of secondary power is changed as in this study, all systems that draw energy from the secondary power system must be made compatible with the new power system. For example, the flight control system, which previously used hydraulic actuators, now must be converted to electric actuators. The baseline anti-icing system, which previously used engine bleed, now must be converted to some type of electric system. The environmental control system, which also previously used engine bleed, now must be replaced with either an electrically powered system or a combination electrically

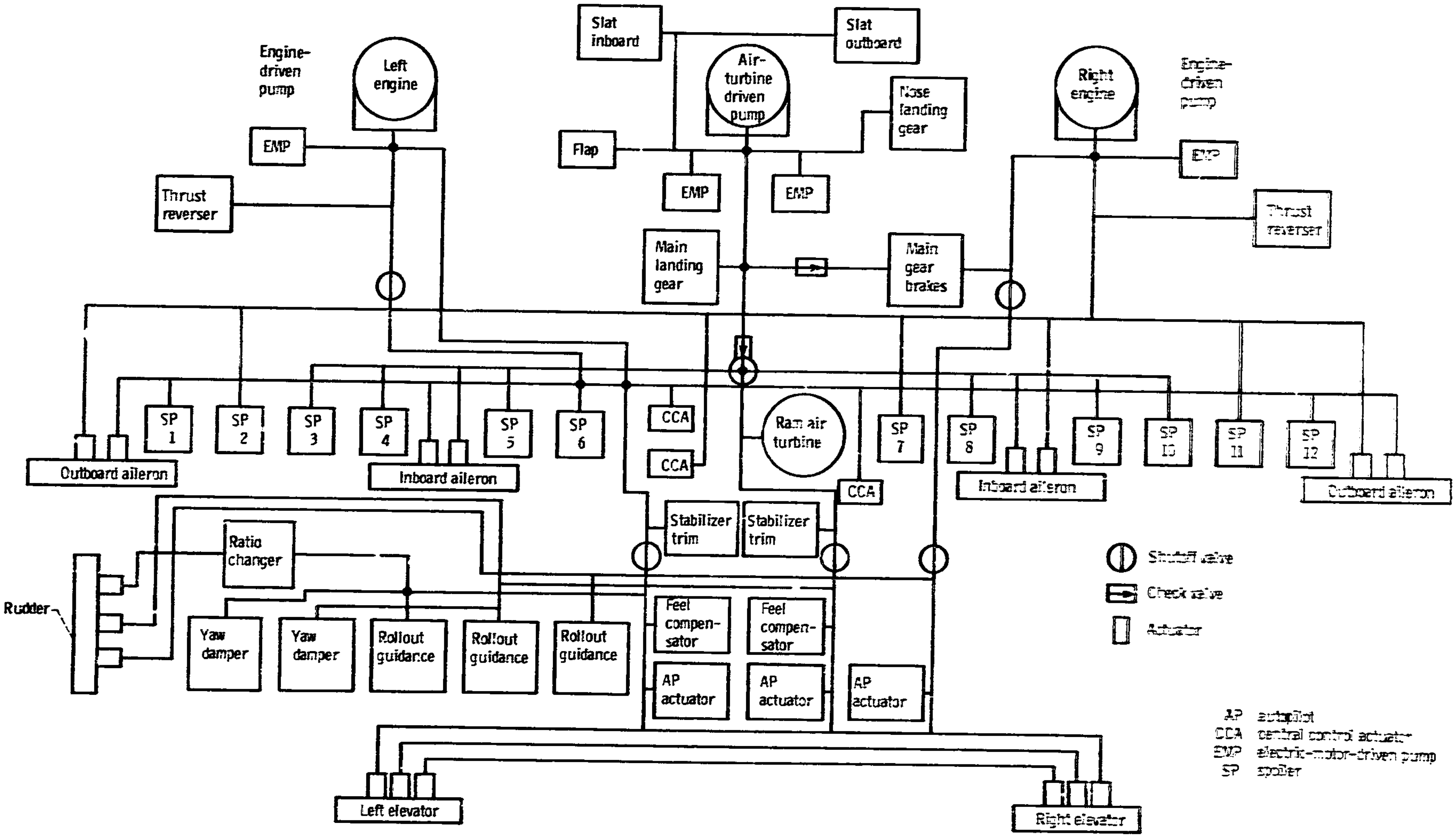

Figure 6.—Baseline hydraulic flight control system.

and directly driven system. A number of other items also must be addressed such as the APU, landing gear, steering, brakes, and thrust reverser. The following sections describe these replacement systems and subsystems.

Advanced Power System

When evaluating a replacement system to provide secondary power for aircraft, one of the first conclusions reached is that a single type of power is most desirable. This allows load sharing between all of the power sources and results in a smaller capacity power system. When considering the type of power to be used, it is important to note that the power requirements for the avionics, lights, fans, etc., can only be met by electric power. As a result, electric power is the only type that can potentially accommodate all of the secondary loads on the aircraft.

Many types of electric power system can provide secondary power. Existing-technology systems considered included 400-Hz generators with integrated constant-speed drives, cycloconverter systems, 270-V dc distribution systems, and dc link systems. When evaluating a power system for an all-electric aircraft, the effect on other systems and the overall effect on the airplane must also be considered. This includes such factors as the types of controllers required for control surface actuators, the size and weight of the circuit breakers, and the system failure modes.

A preliminary estimate was made of the effect on aircraft fuel consumption for the existing-technology power systems considered. The saving was 2 to 3 percent over the present-technology baseline. However, the saving for a high-frequency ac distribution system based on a bidirectional resonant power converter was estimated at 8.4 percent. Since this was a substantial gain over the other systems, this system was selected for the rest of the study.

Resonant power conversion.—The primary technology used in the high-frequency ac power distribution system is resonant power conversion. Converters of this type have been built and tested for space applications (ref. 7). Although the space applications used dc sources and different loads than an aircraft, the basic technology has been tested.

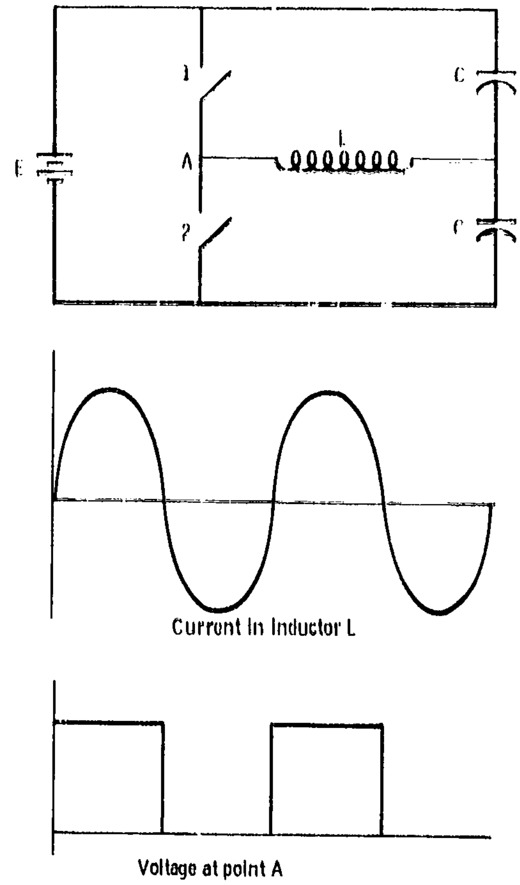

Figure 7.—Basic concept of resonant power converter.

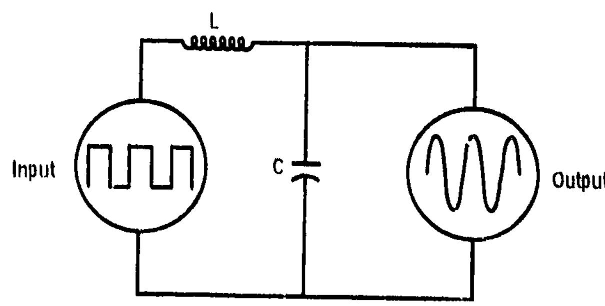

Figure 8.—Resonant power converter equivalent circuit.

In the basic concept of the resonant power converter (fig. 7) switches 1 and 2 are alternately switched in such a manner as to present the series inductor-capacitor (LC) circuit with square-wave voltage. The LC circuitry, performing the function of a low-pass filter, allows essentially only the fundamental (sinusoidal) current to flow in the series circuit. This characteristic is more easily seen by inspecting the equivalent circuit (fig. 8). In this configuration the load is placed across the capacitor and thus provides a low-impedance sinusoidal voltage source. Returning to figure 7, since the current in the inductor is sinusoidal, the switch can be opened as the current passes through zero. Zero-cross switching yields an advantage that cannot be overemphasized: the absence of energy loss during power switch turnoff is a major advantage of resonant power conversion (ref. 8). Not only will silicon-controlled rectifier (SCR) switches self-commutate, but also there is no frequency-proportional converter power loss at turnoff. As a result power devices may be safely and efficiently operated at power and frequency levels unobtainable by other conversion techniques.

Lower frequency waveforms can be synthesized from this high-frequency carrier as required to satisfy load requirements. A basic circuitry connection to allow this is shown in figure 9. In this circuit, switch pairs 1,1′ and 2,2′ are operated in such a manner as to perform synchronous rectification of the 20-kHz source and thus synthesize a lower frequency output. In this respect the circuit operates somewhat as a conventional cycloconverter. Proper sequencing of the switch pairs will also allow reverse power flow by chopping a lower frequency (including dc) into the 20-kHz source.

The inherent symmetry of the high-frequency inversion system is illustrated in the bidirectional implementation shown in figure 10. In this configuration port A can act as a source and port B as a load, or vice versa. Also both ports can act as sources with other loads connected across the high-frequency link. The circuitry illustrated in figure 10 is for a dc or single-phase ac source, but it can be expanded for multiphase sources. This configuration illustrates the interface between multiple power sources, power generation and storage, and power sources and system loads (ref. 9).

Rationale for final system selection.—The electric distribution system is a single-phase, sinusoidal alternating-current, 440-V, 20-kHz power system. This particular configuration was selected for the following reasons: A single-phase system minimizes the switching and termination complexity involved in a fully redundant multiphase, multisource power distribution system. The normally recognized advantages of three-phase power distribution are lower conductor weight and constant power delivery. However these advantages are not pronounced in this application. The relatively high operating voltage and the distributed power system limit the total-installed-power conductor weight of the aircraft to only a few hundred pounds. Therefore the potential for weight reduction is small and could easily be offset by the added complexity in switching and terminations. The 20-kHz power system has a power frequency of 40 kHz (both the positive and negative half-cycles deliver power). Although this is not truly "constant power," 40 kHz is well beyond the mechanical frequency response of any contemplated load and will result in virtually constant electric power delivery.

There are many advantages to ac sinusoidal power distribution. Of primary importance is the ease of current limiting and fault protection. As the ac source always returns to zero, switching devices do not have to interrupt large current overloads. Another major advantage is that voltage levels can be passively transformed without waveform distortion. Finally the single carrier frequency presents minimal problems in regard to electromagnetic interference since it can be easily filtered or shielded.

For any level of electric power distribution there exists an optimum voltage level depending on the associated design constraints. In general, however, power should be transmitted at the highest practical voltage level to minimize line losses. A voltage of 440 V ac was selected as providing good transmission efficiency, requiring only normal insulation thickness, and allowing reasonable conductor spacing. This voltage is below corona initiation potentials at altitudes of less than 15 240 m (50 000 ft).

The high-frequency power distribution, in this case 20 kHz, has numerous advantages. A high operating frequency minimizes the weight of magnetic devices such

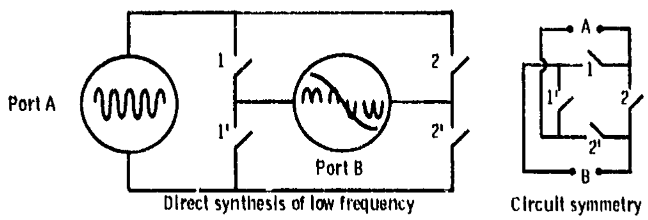

Figure 9.—Basic circuitry for waveform synthesis.

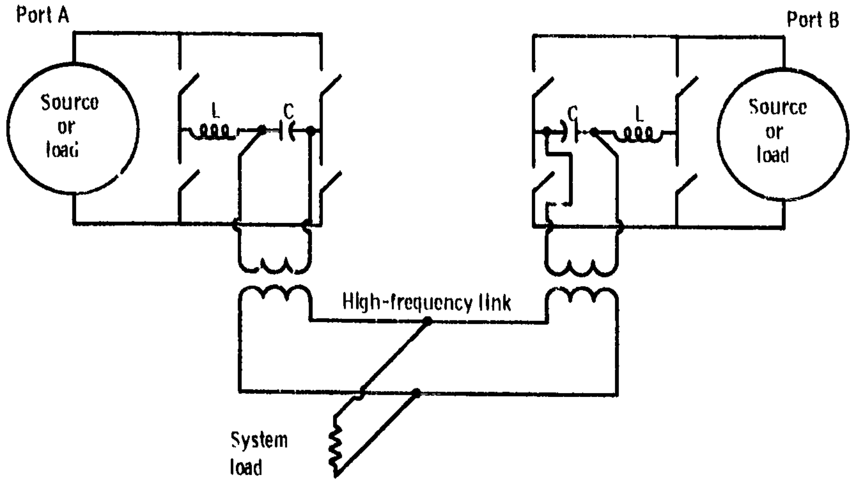

Figure 10.—Bidirectional high-frequency power interface.

as transformers. The fundamental operating frequency, and in particular the power frequency, are above the audio range. This will provide a quiet power system. The energy available per cycle is inversely proportional to the operating frequency, and this minimizes the damage occurring during electrical faults. With the interrupt occurring within one half-cycle, the total energy available in the power system is only about 10 J. The low energy content also improves personnel safety over conventional power systems. Accidental encounters with this system, while quite painful, would most probably not be fatal. A 400-Hz power system would have 50 times more available energy per cycle than an equivalent 20-kHz system. Finally a high-frequency power system allows lower frequency waveforms to be synthesized with a minimum of distortion. Lower distortion waveforms result in more efficient operation of electrical devices. Waveform synthesis allows "load tailoring" of power with relatively simple circuitry, which in turn easily implements such concepts as variable speed control of ac motors or true harmonic (jerk free) actuator motion.

System description.---Using resonant power conversion as the core technology, a power system was configured (fig. 11). The system has four motor/generators, two mounted on the accessory gearbox of each engine. These are 90-kVA, three-phase, 24 000-rpm, 1200-Hz induction machines. The outputs of these generators are fed to three-phase bidirectional, resonant converters. Each phase of these converters has the configuration shown in figure 12. The outputs of the three phases are connected in series and tied to the distribution bus. Although the generator speeds for the two engines are different, the outputs of the converters are synchronized so that all power sources can be paralleled.

The power distributed is a 440-V, 20-kHz, single-phase sine wave. There are dual-redundant buses, each capable of carrying the entire load. All of the sources and all of the loads have access to either bus, and under normal conditions, the two buses are operated in parallel.

Emergency power is provided by a ram air turbine (RAT) and two batteries. The RAT drives a 12-kVA generator, which feeds power through a resonant converter to the distribution bus. The two nickel cadmium batteries are each rated at 1.5 kW-hr. The batteries also provide uninterruptable power for the computer bus in case one or both of the main buses have a power interruption.

The loads are fed from the main distribution bus through isolators and controllers (fig. 13) to protect the bus from load faults. The amount of power conditioning varies considerably depending on the load (fig. 14). (The bus isolators and controllers have been deleted in figure 14 for simplicity.) The galley, which may use induction heating, can use the 20-kHz power directly. The ECS drive motor requires a load receiver for starting and for varying motor speed. The flight control actuators require a separate load receiver for each actuator. Figure 15 is a simplified schematic of the load receiver used to drive any three-phase motor (ECS or actuator) from the single-phase distribution bus. The main and utility bus loads that require lower voltages are supplied through step-down transformers. Those loads requiring a frequency change are supplied by a standard-configuration load receiver.

System operation.—Under typical conditions the induction generators provide 440-V, 1200-Hz, three-phase power to the converters. The voltage and frequency vary as the engine speed varies, but the voltage is regulated by controlling the slip frequency. Since this type of regulation responds slowly, rapid regulation will be provided by controlling the relationship between the three converter phases. Figure 16 illustrates two phasor diagrams that have the same bus voltage. However, the phase voltages (A, B, and C) are approximately 25 percent larger in the left diagram than in the right diagram. The bus voltages are made equal by adjusting the phase angles, with the higher phase voltages having smaller angles and vice versa. It is anticipated that this type of regulation can be made to respond within one cycle of the converter frequency, or approximately 50 μs. The frequency of the converter is controlled by its resonant circuit and is therefore independent of the input frequency. The result is a constant-frequency (20 kHz), constant-voltage (440 V), single-phase output to the distribution bus. Since the power to the loads is synthesized by a load receiver, some latitude in the bus voltage and frequency can be allowed. Under normal operation all four converter outputs and both distribution buses are paralleled.

In most instances 20-kHz and 440-V power cannot be used directly by the loads. Load receivers are used to synthesize the output required by the load. Even though the bus is single phase, a three-phase motor can be driven from the receiver by using a simple set of switches (fig. 15). The power flow from the main distribution bus is managed in half-cycle increments by the load receiver switches. The pulses generate a sinusoidal energy pattern when the proper switching sequence is used. The pattern (fig. 17) represents a 21:1 frequency ratio and the maximum output voltage. All of the available pulses are being used; and with two phases of the motor connected to the bus at any time, the maximum number of pulses per phase is two-thirds of the total. To maintain a constant voltage-to-frequency ratio into the motor, the pulse pattern is varied by inserting blank spaces into the pattern. This is done by leaving all load receiver switches off for one half-cycle. A pulse pattern with a 42:1

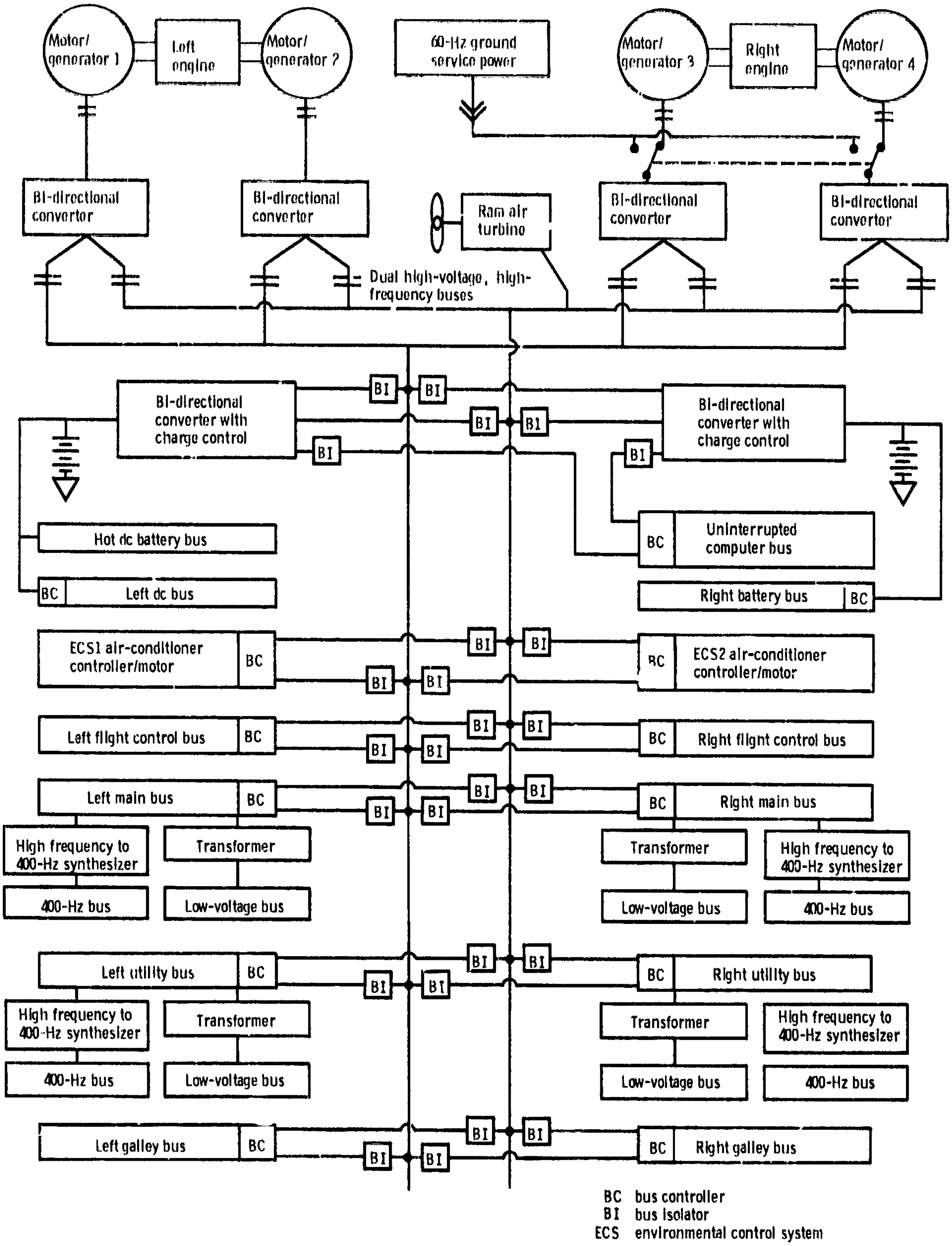

Figure 11.—Advanced power distribution system.

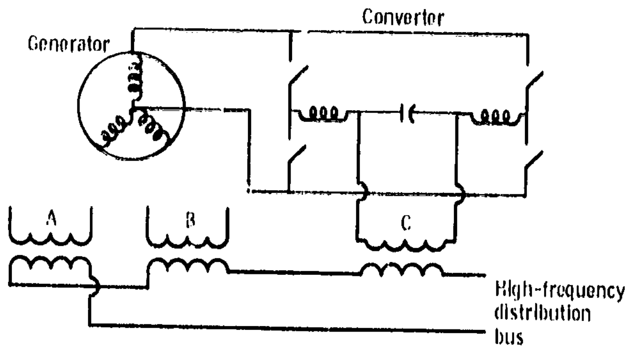

Figure 12.—Generator-converter interconnection for advanced system.

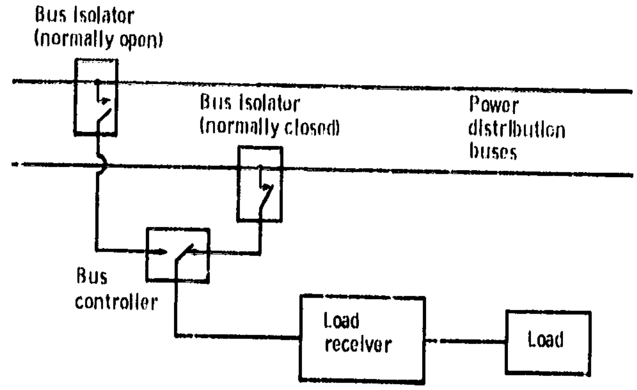

Figure 13.—Load power bus interconnection for advanced system.

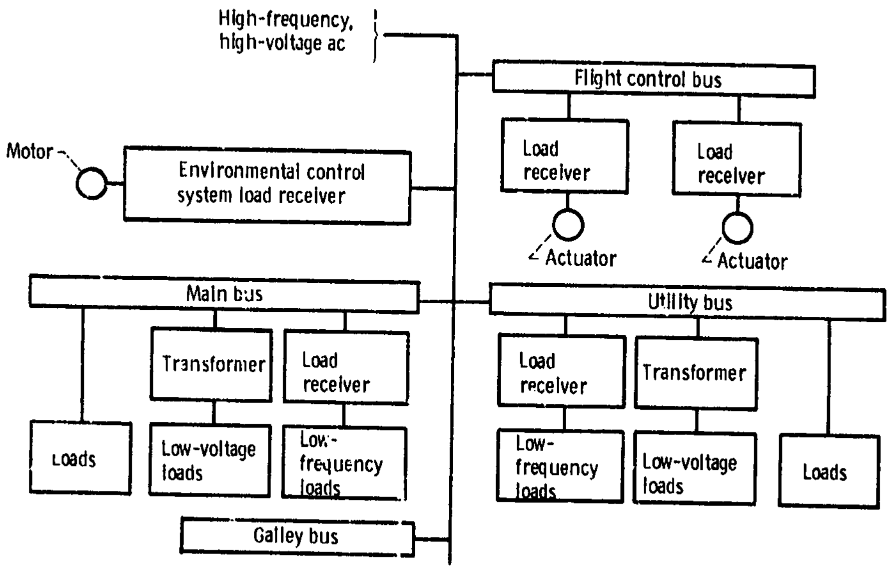

Figure 14.—Load power conditioning for advanced system.

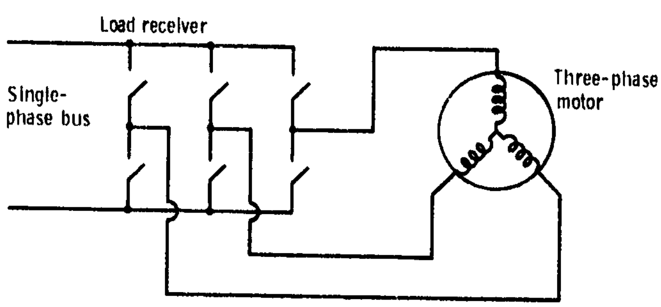

Figure 15.—Load receiver for three-phase load for advanced system.

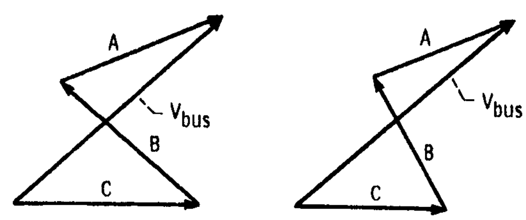

Figure 16.—Regulation of bus voltage by controlling phase relationship.

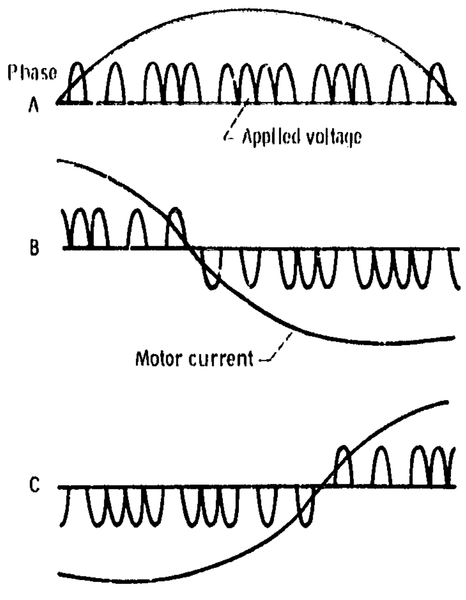

Figure 17.—Synthesis of three-phase, low-frequency output—21:1 frequency ratio; maximum output voltage.

frequency ratio is shown in figure 18. This is the same pattern used previously, but with all switches off every other half-cycle. The resultant voltage and frequency have both been halved, maintaining the desired ratio. For variable motor loads the voltage and frequency can be varied to meet the load requirements all the way from dc for holding actuator motors in a fixed position to approximately 1000 Hz for providing fast slew rates. Higher frequencies can be obtained by selecting other pulse patterns.

The overall management of the power system is accomplished through a digital data and control system. The various power converters, load receivers, bus controllers, bus isolators, and power contactors are all tied into this digital system. These circuit elements control load shedding, bus switching, and fault clearing. Individual loads are shed by commanding the load receiver to turn off for that particular load. For major load shedding the main power converters are turned off and entire buses (such as the utility bus or galley bus) are disconnected by either the bus controller or the bus isolator and then the main converters are turned back on. It is estimated that this can be done in under 10 ms. Fault clearing is similar. A fault can be cleared at three points between it and the main bus: the load receiver, the bus controller, and the bus isolator. A fault on the main distribution bus is cleared by isolating that bus and transferring all desired and essential loads to the remaining bus.

The bus isolators (fig. 13) are single-pole, single-throw metallic contacts. They prevent any direct connection between the distribution buses and provide a positive off-condition. Although solid-state remote power controllers have been developed at NASA Lewis and elsewhere, there are certain inherent limitations to their universal application. First, they are not circuit breakers in the usual sense because their interrupt ratings are not appreciably higher than their carry ratings. This first characteristic requires careful analysis in their application. Second, their most common failure mode is a short of the series semiconductor element. At least, such a short could mean a runaway load. To prevent such problems, the proposed system uses metallic contact bus isolators. Since such switches have only to open, no

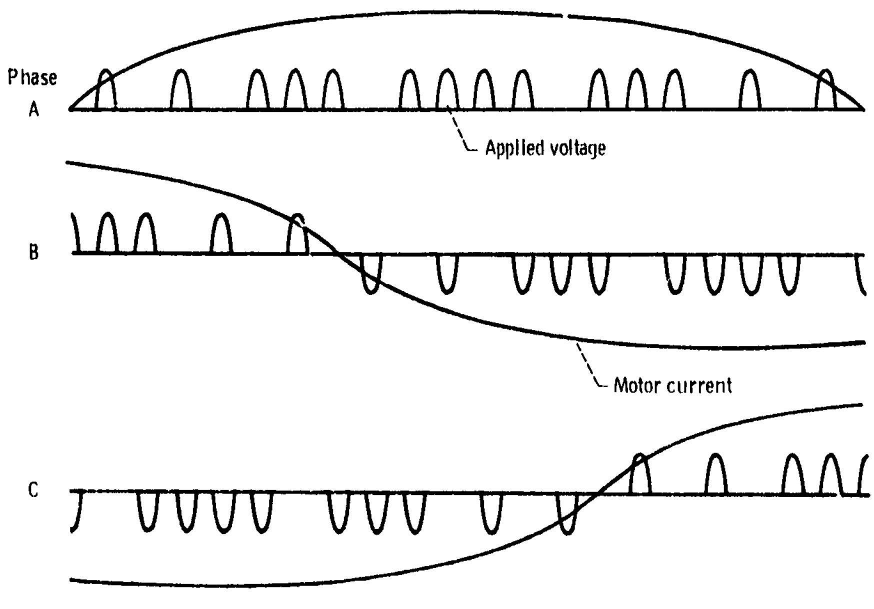

Figure 18.—Synthesis of three-phase, low-frequency output—42:1 frequency ratio; one-half maximum output voltage.

bounce time is involved. Also, as the contacts do not interrupt current, they are relatively light. These characteristics result in a rapid relay response time.

The motor/generators and main power converters are used to start the engines. For a typical ground start, power is fed from a ground source such as a three-phase, 440-V, 60-Hz utility line. The converters associated with the right engine convert this power to single phase, 440-V, 20 kHz. This power is fed over the main transmission b to the converters for the left engine. Since the converters are bidirectional, they can control and convert power flow in either direction. These converters provide controlled-current power with a constant voltage-to-frequency ratio to the two machines, now acting as motors. The motors thus provide a constant torque to the engine to bring it up to starting speed. Although both motors would normally be used for starting, one motor has adquate power to start the engine, but with an associated increase in starting time. Also, other loads can be fed from the 20-kHz bus during the starting operation. Since the power to the starters is current controlled, there should be no significant perturbations on the bus. Once the left engine has been started, the power from its two machines, now acting as generators, is fed in the opposite direction to start the right engine.

System sizing.—There were two criteria for sizing the power system. First, all essential loads would be provided by a single generator. This allows normal flight operation even if one engine and one generator on the opposite engine should fail. Second, all loads would be provided by three generators. This allows aircraft dispatch with one generator failed. The steady-state loads for six phases of the flight profile (table IV) were divided into three major categories: essential, utility, and galley. Essential loads include all those loads required for normal flight operation (but not convenience loads such as galley and ECS loads). The main and flight control buses (fig. 11) are a part of the essential loads. The major utility load is the environmental control system. Utility buses (fig. 11) are also included in this category. The galley category covers just the galley loads.

Presenting the same load information graphically (fig. 19) shows that all loads can be accommodated with three generators (one generator failed). With only two generators (one engine out) the essential and utility loads can be accommodated at cruise. Even with only one generator functioning, all essential loads can be powered, allowing normal flight control.

The major transient loads (fig. 20) occur during takeoff and approach. To maintain the load within the normal ratings of three generators requires load shedding. For takeoff the galley load was shed to accommodate the gear retraction load. However, the galley would not need to be on during this phase of the flight. With the failure of one engine on takeoff the two remaining generators would provide sufficient power for the essential loads and full-speed retraction of gear, flaps, and slats.

The emergency backup power system was sized to provide power to flight-critical loads if all generator power were lost (both engines out). The one additional flight-critical load that is now powered electrically is the flight control actuation. To provide continuous power for the flight controls, a ram air turbine (RAT) drives a 12-kVA generator. This is essentially the same power rating as for the hydraulic pump on the baseline-aircraft RAT that powered the baseline flight control actuators. Short-term excursions above this power level were

TABLE IV.--MAXIMUM EXPECTED STEADY-STATE LOADS FOR ALL-ELECTRIC AIRCRAFT

[Environment control system at hot-day, full-load conditions; galley at full power throughout flight.]

Load category	Flight phase											
	Loading		Engine start[a]		Taxi		Takeoff and climb		Cruise		Descent, hold, and land	
	kVA	Power factor	kVA	Power factor	kVA	Power factor	kVA	Power factor	kVA	Power factor	kVA	Power factor
Essential (includes main bus and flight controls)	29.7	0.92	38.1	0.91	56.5	0.91	80.3	0.91	80.3	0.91	74.9	0.91
Utility (includes environmental control system)	133.5	.85	----	-----	129.1	.85	130.4	.85	87.2	.85	87.8	.85
Galley	37.0	1.0	----	-----	50.8	.99	47.8	.99	57.8	.99	13.2	.95
Total	195.7	.91	38.1	.91	233.3	.91	254.7	.91	222.4	.92	175.3	.89

[a]Engine start loads are those in addition to the starter load.

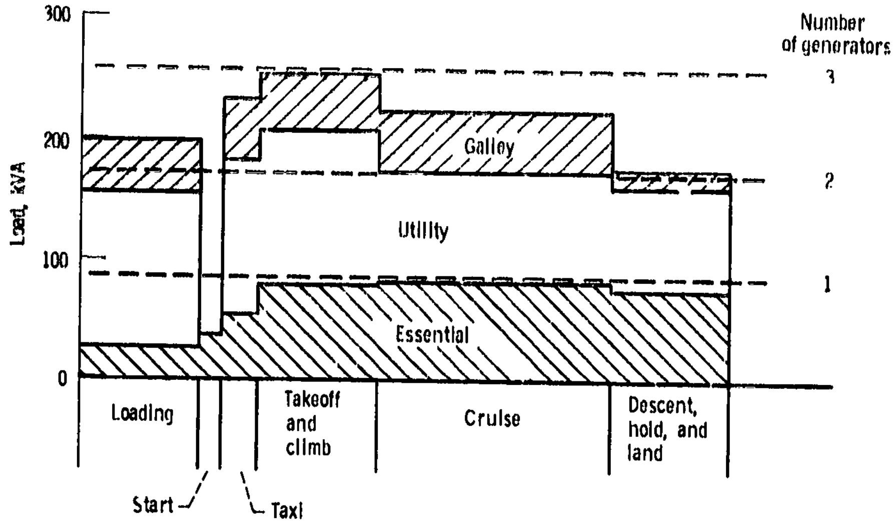

Figure 19.—All-electric aircraft load profile.

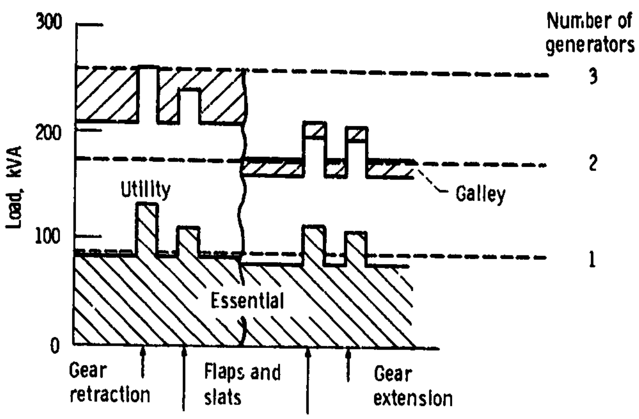

Figure 20.—All-electric aircraft takeoff and approach load profile.

provided for by accumulators in the baseline-aircraft hydraulic system. In the electrical system batteries are used to accommodate these short-term excursions.

In addition to the RAT two battery systems are provided. Each of these batteries is approximately equivalent to the battery on the baseline aircraft. The battery power drain under emergency conditions is essentially the same as that for the baseline aircraft with the exception of control system surges. Therefore battery power is expected to be available under emergency conditions approximately twice as long as for the baseline aircraft.

The electric generation and distribution equipment for the advanced power system weighs 1039 kg (2290 lb). This includes the generators, main converters, distribution buses, bus isolators, batteries, battery converters, ram air turbine, power management system, and some miscellaneous load receivers.

Advanced Aircraft Flight Control System

With an all-electric secondary power system on the aircraft the flight control system must be modified to be compatible with this new power source. Electromechanical actuators (EMA) were selected as being the most direct and simplest method of powering the flight control surfaces. With EMA's the only logical control signal system would be an electric system. A digital databus system was selected as being the most flexible, most noise-immune, and lightest way to provide the control signals to the EMA's.

Several ground rules were established to define the scope of the flight control system effort. First, the number and location of the flight control surfaces would remain the same as on the baseline aircraft. This was done to minimize the differences between the aircraft except for those directly associated with the new power system. Second, the redundancy levels for all flight control elements would be the same or greater than in the baseline system. This was done in an effort to make the advanced flight control system at least as reliable as the baseline, although a reliability analysis was not performed. Third, the baseline hydraulic actuators would be replaced with EMA's in the same location and with the same attachment points. This was done to minimize differences between the baseline and advanced aircraft and also to limit the scope of this effort. Although EMA-driven control surfaces would very possibly be configured differently than those with hydraulic actuators, that was judged to be beyond the main intent of this study.

The flight control system defined in this study is simply a conceptual layout of a system. No system parameters, such as data rates, were defined. The state of the technology was assumed to be that which would be available in the same time period as the power system technology, or about the mid-1990's.

General system description.—The electric power distribution for the flight control system is shown schematically in figure 21. Seven separate power sources can supply energy to the flight control system: four engine-mounted generators (two on each engine), one ram air turbine (RAT), and two separate battery systems. The RAT is sized to provide sufficient power for normal flight control loads, with the batteries supplying additional power for short-term peak loads. Thus the flight control system is fully operative even with both engines out.

All of the primary flight control surfaces (aileron, rudder, and elevator) have multiple actuators. This redundancy protects against failures. The actuators are fed from different power buses, as indicated by the solid line connection (fig. 21), so that all primary surfaces have power in the event of a bus failure. The dashed lines represent potential connections so that an actuator can be switched from one power bus to the other to maintain all actuators powered. The spoiler panels each have only a single actuator. Redundancy is provided by multiple panels. The spoiler actuators are connected to the bus in a symmetrical pattern on each wing. This will prevent asymmetrical forces on the aircraft during speed braking should a bus failure occur.

The signals for the flight control system are distributed over a digital data-bus system (fig. 22). The redundant data buses carry identical command information for protection against bus failure. The flight computer and the remote terminals (RT) have access to either data bus, but the RT's are connected to only one at a time. The computer and RT's gain access to the data bus on a priority basis. Command information has the highest priority and routine sensing or status information, the lowest priority.

The control commands originate at the flight controls with inputs from the pilot or autopilot. Sensors feed the information to the input RT's, where it is placed onto the cockpit data bus. The information is then available to all RT's and computers connected to that data bus. The status-display RT's use the information to update their display. The flight computer uses the information to calculate the necessary flight control commands. These commands are fed from the flight computer to the flight control data bus. The command information then becomes available to all RT's connected to that data bus.

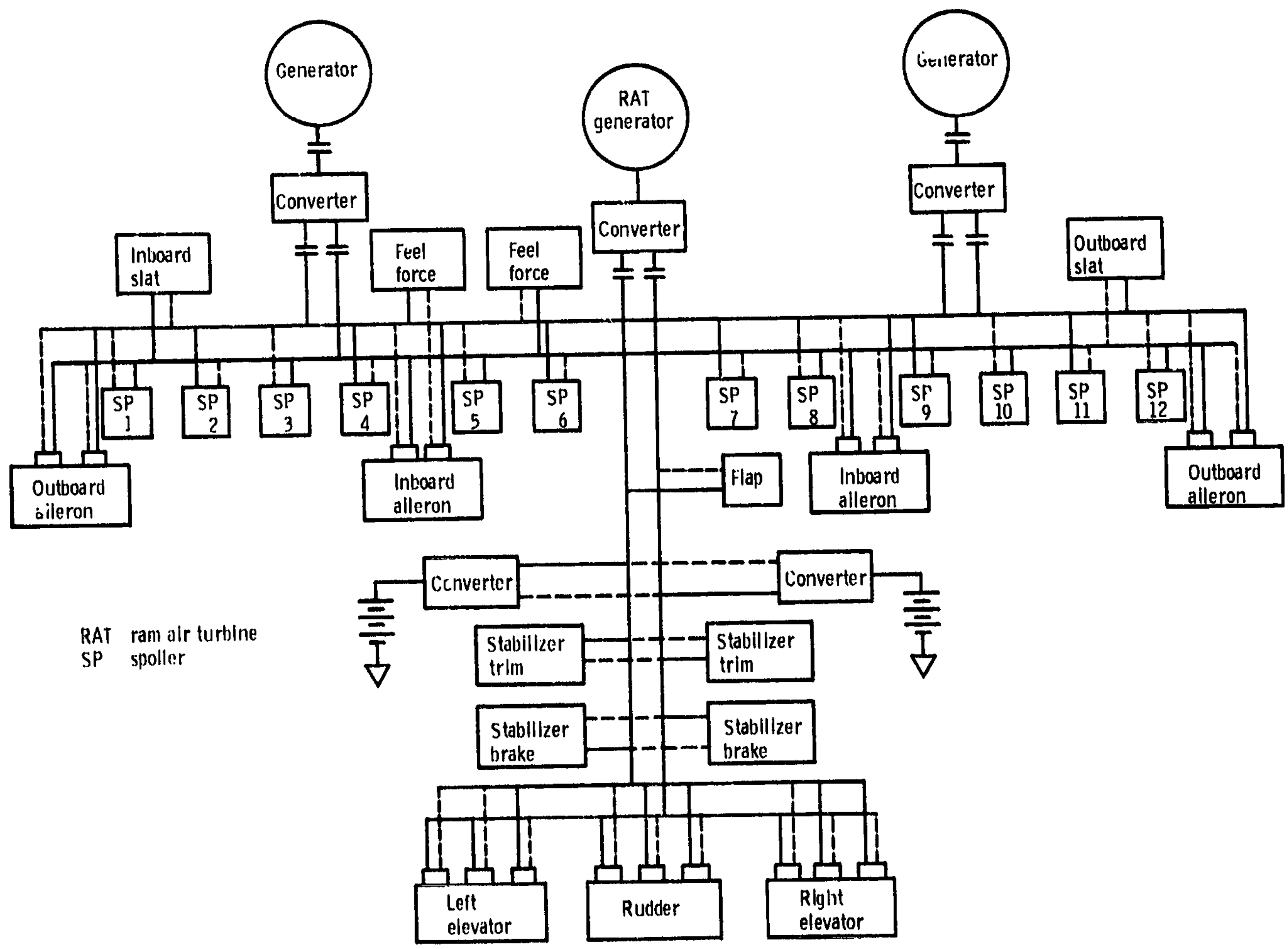

Figure 21.—Electrical schematic of all-electric aircraft flight control system.

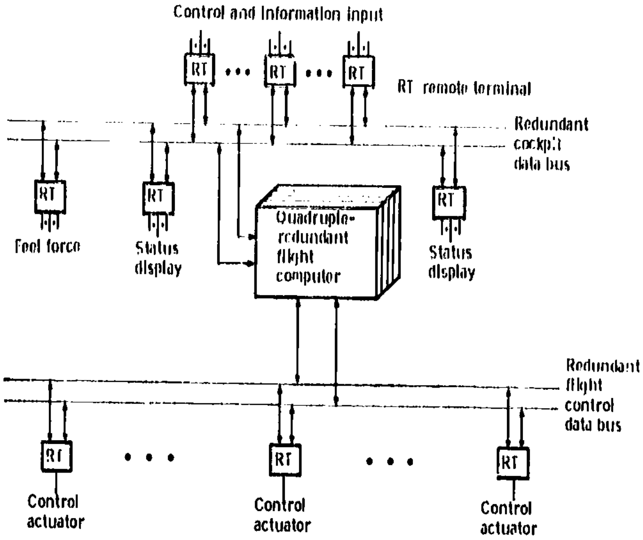

Figure 22.—Digital flight control data system.

RT's associated with commanded actuators accept the information and operate their actuators in accordance with the commands.

The function of the remote terminal (fig. 23) is to provide the data processing and control required by the actuator. The bus transfer unit selects one of the two redundant buses. Redundant actuators operate from different buses so that both will not be affected by a bus interruption. The RT processes the command data and provides the required output signals to operate the actuator. The RT also routes information from the actuators to the data bus so that the flight computer obtains the necessary information from the actuators.

A remote terminal interfaces with an actuator through a load receiver (fig. 24). The processed command signals are fed to a load receiver, which controls the power flow to the actuator.

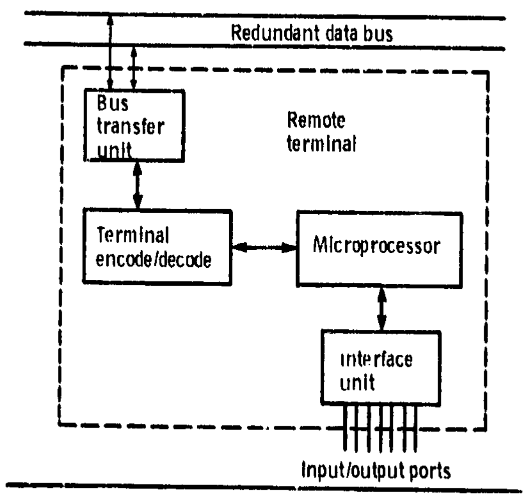

Figure 23.—Block diagram of remote terminal.

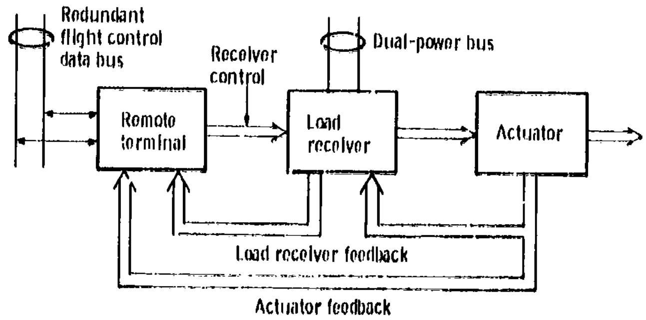

Figure 24.—Block diagram of actuator control.

Detailed description.—The flight control system has four parts: longitudinal, lateral, directional, and high-lift control.

Longitudinal control system: As with the baseline aircraft pitch is controlled by elevators and a variable-incidence horizontal stabilizer (figs. 25 and 26). The right and left elevators are each driven by three electromechanical actuators that receive their inputs from the flight control computer via the redundant digital bus. The pilot's control column interfaces, through position sensors, with the flight control computers. The actuators are distributed on the left or right power bus but are capable of switching to either bus if a fault or failure occurs in its primary bus. The fully powered control system will require artificial "feel" forces to be provided at the pilot's control column. These forces are provided by electromechanical actuators that connect to the pilot's control column via mechanical linkages.

Longitudinal trim is provided by varying the angle of incidence of the stabilizer. The stabilizer is trimmed in manual or automatic mode through a dual-load-path ballscrew actuator powered by two electric motors via a differential gearbox.

Lateral control system: Lateral control, as with the baseline system, is provided by two ailerons and six spoilers on each wing (fig. 27). The pilot's control wheel position is now input to the flight control computer, via the cockpit data bus, which interfaces with the flight control bus.

The advanced system has the same number of actuators and redundancy as the baseline system. There are, however, significant differences between the systems. The lateral central control actuators, which were used to boost the control wheel forces and to input the autopilot control, have been eliminated. The flight control digital data bus interfaces directly with the flight control computer, and all aileron and spoiler autopilot functions are implemented in software in the flight control computer.

The artificial "feel" forces for the lateral control system are obtained by using controlled torque motors connected to the pilot's control wheel with cable linkages.

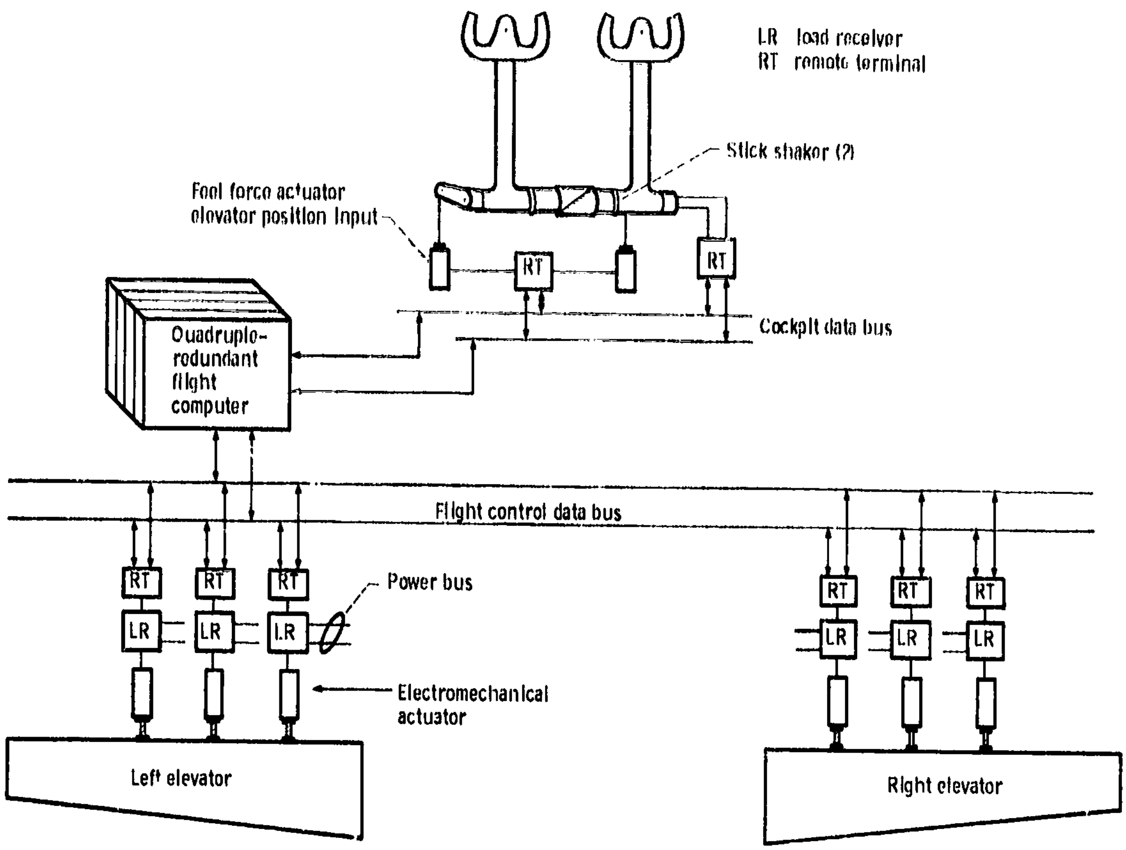

Figure 25.—Schematic of elevator control system.

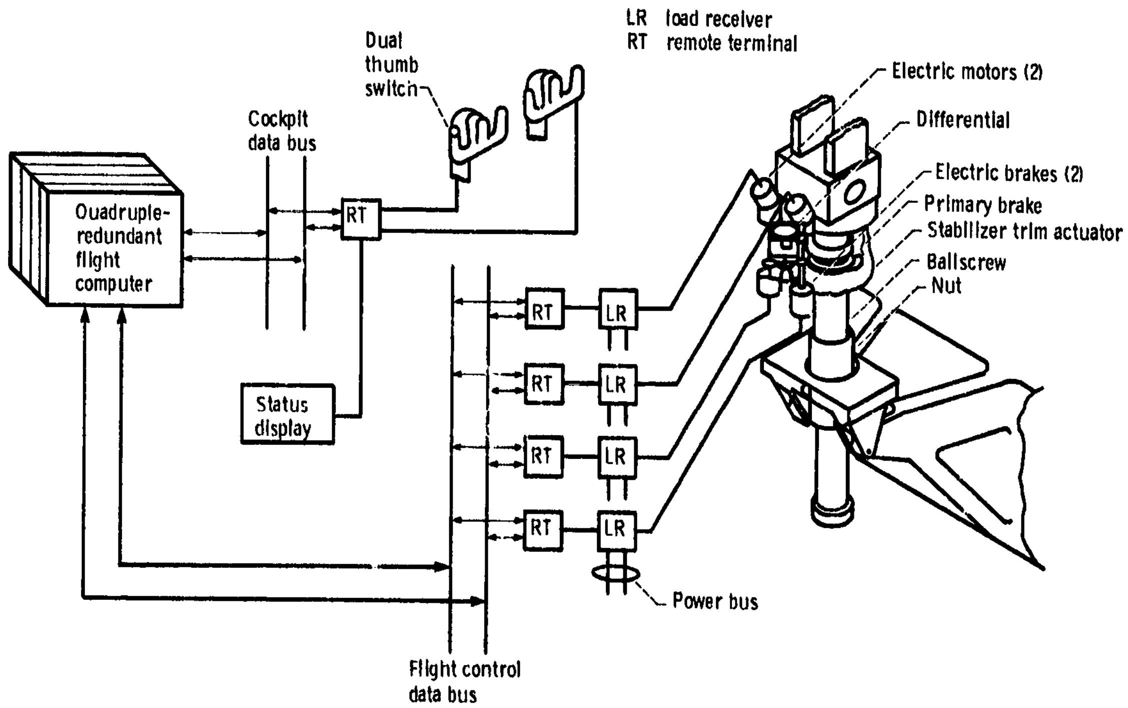

Figure 26.—Schematic of stabilizer trim control system.

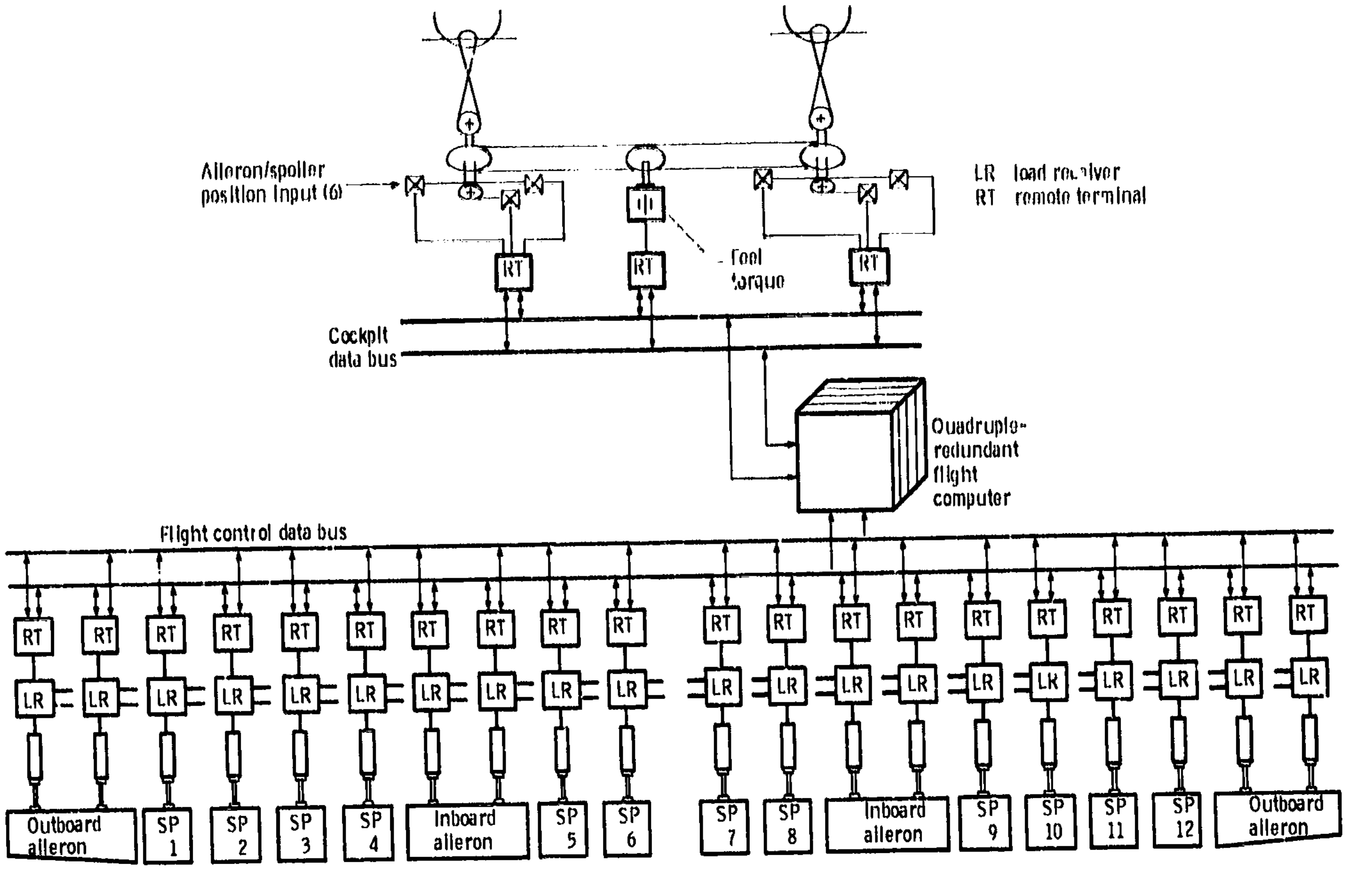

Figure 27.—Schematic of lateral control system.

The torque motors are controlled via the cockpit data bus.

Directional control system: Directional control of the advanced system, as for the baseline system, is provided by the rudder control system (fig. 28). The rudder control system is operated by displacing the pilot's pedals, which are linked to transducers. Pedal motion is transmitted via the cockpit data bus to the flight control computer, which links to the rudder actuators via the flight control data bus.

The rudder "feel" system is a spring-loaded cam and follower system that is powered and controlled electrically and interfaced with the flight control computers via the cockpit data bus.

Rudder trim is done with the rudder trim knob, which inputs trim information to the flight control computer. The yaw damper and autopilot functions, which were obtained by using mechanical hardware in the baseline system, are now implemented in software. The software algorithms for these functions are done redundantly in the flight computer.

High-lift control system: The high-lift system structure and function remain as in the baseline aircraft. The hydraulic power drive units and associated gearboxes have been replaced by electric motor drive units and gearboxes (fig. 29). The electric drive units and RT's interface with the flight computer via the digital flight control data bus. The flap/slat control lever in the cockpit inputs the set point to the flight control computer. Flap/slat position information and system status are now communicated over the digital data bus to the flight computer.

The electric drive motors for the advanced high-lift system will be capable of accessing either the left or right power bus. This allows complete system operation if either power bus fails.

The advanced aircraft flight control system equipment weighs 363 kg (800 lb). This includes the electromechanical actuators, the load receivers, the remote terminals, the digital data buses for cockpit and flight control, and the flight deck controls.

Deicing System

The baseline aircraft uses a bleed air anti-icing system on the engine inlets and the wing leading edge outboard of the engines. The replacement system will be limited to those same areas.

Many methods can be used to provide ice protection on aircraft surfaces. Only two were given serious consid-

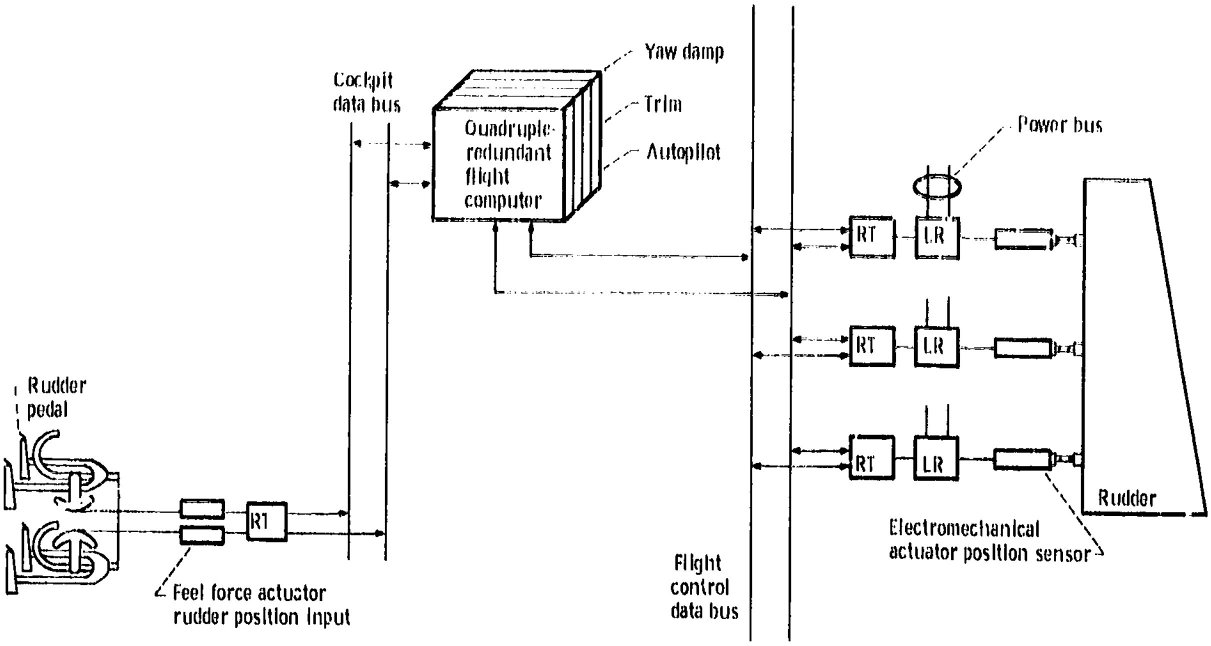

Figure 28.—Schematic of rudder control system.

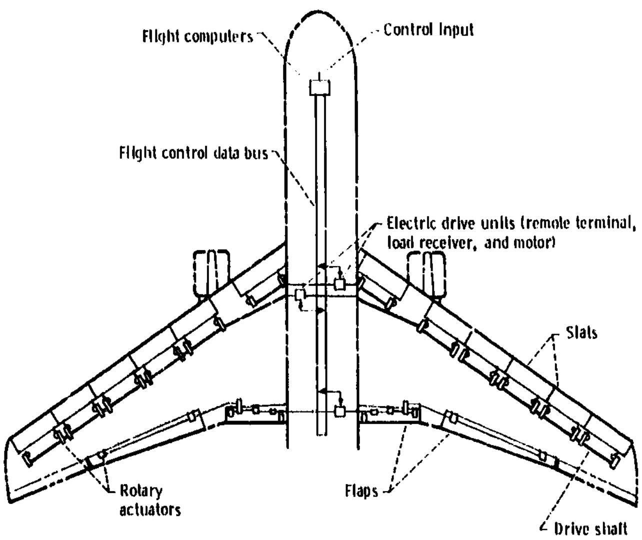

Figure 29.—High-lift control system.

eration in this study: electrothermal anti-icing, and electroimpulse deicing (EIDI). Electrothermal anti-icing can provide protection at about one-tenth of the power consumption used by a bleed air system. It is very compatible with an all-electric aircraft, particularly one with a high-frequency distribution bus, because induction heating is easily accomplished. Electroimpulse deicing (ref. 2) is even more energy efficient, with its power consumption less than one-tenth of that for electrothermal. It is also very lightweight. As a result, it was selected as the ice protection system for this study.

Electroimpulse deicing (ref. 1) operates on the principle of suddenly, and elastically, deforming the metal skin of the airfoil, causing the ice to fracture and subsequently be ejected from the surface. Electrostatic energy is discharged into electromagnetic coils mounted close to the aircraft skin. Electric energy is supplied at a relatively low power level to a capacitor bank from a convenient power source and allowed to build up a substantial charge. Upon discharging, the surge of electric energy from the capacitor creates a steep wave front in the coil, and this results in the aircraft skin moving away (repelling) rapidly within its elastic limit. Since the ice deposited on the airfoil is brittle, the sudden movement of the aircraft skin causes the ice to fracture explosively, and it is thereby ejected from the surface in that area. In some instances a first impulse only fractures the ice deposit, which remains intact over the airfoil leading-edge surface. A subsequent impulse, however, successfully removes the ice formations in the area. A schematic of the aircraft's portside wing (fig. 30) illustrates the EIDI system configuration. The power source and controlling unit will be located in the fuselage; the capacitor bank will be located in the wing about in the middle of the overall length of the five slats outboard of the engine. A silicon-controlled rectifier will be used in conjunction with each coil.

The five slats (fig. 31) are the same in length, about 3.84 m (12.6 ft), but the leading-edge contour varies. EIDI tests conducted at NASA Lewis (ref. 2) show that a coil spacing of 0.61 m (24 in) or less is adequate for

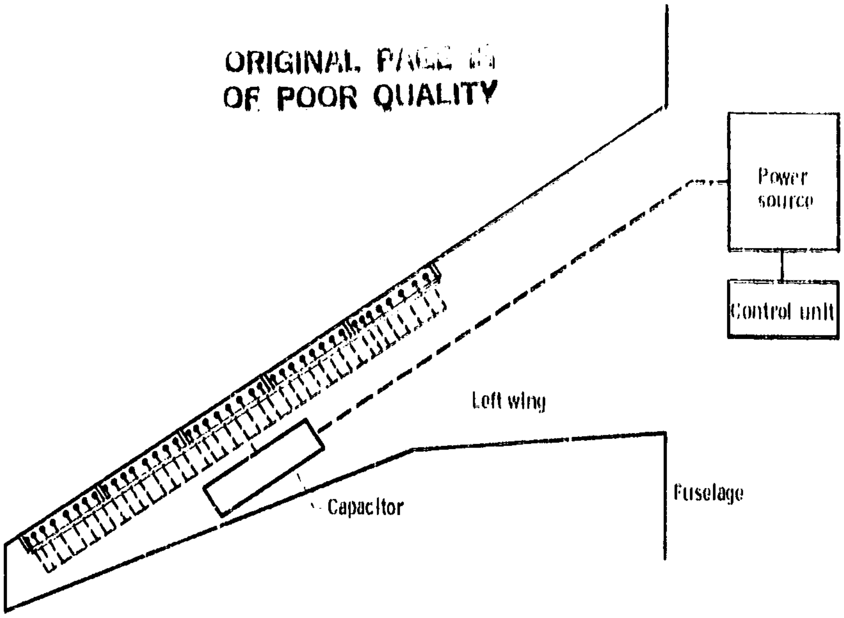

Figure 30.—Schematic of electroimpulse deicing system.

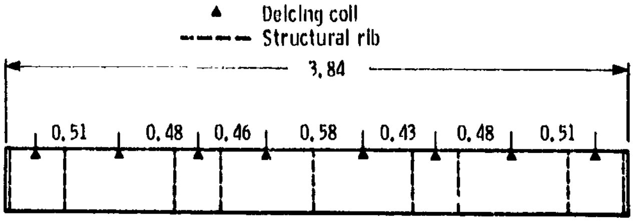

Figure 31.—Typical wing slat showing deicing coil locations. Center slat depicted; typical of all slats (5 per wing). (Dimensions are in meters.)

deicing even under severe icing conditions. The tests also concluded that one coil located right at the wing slat leading edge was just as effective as a pair of coils located just a short distance away on upper and lower slat surfaces. The maximum distance between coils is about 0.58 m (23 in) and there are eight coils for each slat (fig. 31). The other wing's deicing system will incorporate identical components. Each wing requires 40 coils.

The installation of a coil in the leading edge of a wing slat is very simple (fig. 32). In designing a new aircraft the support beam would be made an integral part of the structural design to minimize any weight penalty imposed by the EIDI system. In any case, the support system must withstand the forces generated by the electromagnetic coils without any significant deformation for the EIDI system to work properly.

The engine-nacelle configuration evolving from the NASA/GE Energy Efficient Engine program (ref. 3) was selected for use in this study of the all-electric aircraft. Although some slight sizing adjustments may be needed to obtain the correct engine thrust requirements, they will be insignificant insofar as these deicing weight estimates are concerned.

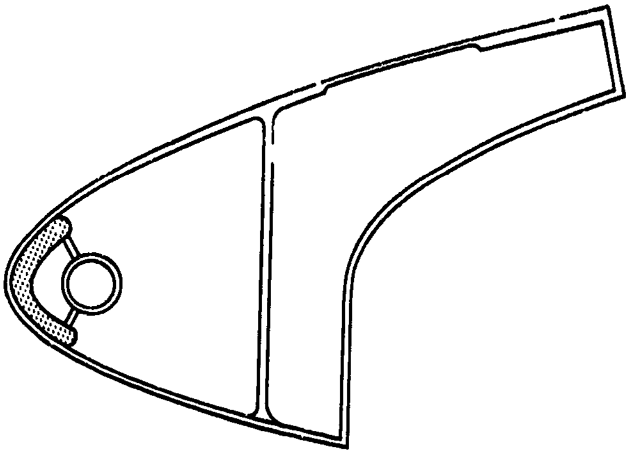

Figure 32.—Typical installation of electroimpulse deicing coil.

The mean radius at the inlet to the nacelle, 1.07 m (42.14 in), translates to a circumferential dimension of 6.73 m (265 in), or about 6.71 m (22 ft). To locate an induction coil at intervals of about 0.46 m (18 in), 15 coils will be required for each inlet. As with the wing EIDI system, it was assumed that a capacitor bank will be located near the coils—in this case, within the nacelle leading-edge area. The power system and control unit will be located in the fuselage, but only one such combination will be required to service the EIDI units for both inlets. Otherwise, the EIDI systems (wing and nacelle) are schematically similar.

The estimated weight for the electroimpulse deicing system for both wings and both engine inlets is 95.2 kg (210 lb). This includes coils, mounting brackets, SCR's, capacitors, power supplies, control units, and wiring.

The system power consumption will be approximately 1 kW if the aircraft is deiced every 90 s.

Environmental Control System

When converting the environmental control system from a bleed-air-powered system to an electrically powered system, any approach that can be driven by an electric motor is a possible candidate. Only two approaches were considered in this study: air cycle and vapor cycle. The air-cycle system, because of its relatively lower efficiency, requires a larger drive motor. This larger electrical load, in turn, requires larger generators and converters, which adversely affects the weight of the entire power system. As a result the more efficient vapor-cycle system was selected for this study.

Since high-pressure bleed air is no longer being used, some other method must be selected for cabin pressurization. Two possible choices were evaluated: an electrically driven, variable-speed compressor and a directly driven (from the accessory gearbox) compressor with variable inlet guide vanes. Driving the compressor electrically would require an increase in the size of the power system. Also, converting the power to electrical and then back to mechanical adds inefficiencies to the system. As a result it was decided to drive the cabin pressurization compressor directly from the engine accessory gearbox.

The replacement ECS has two engine-driven compressors (one on each engine) to provide pressurized air to the cabin. The core speed for the E^3 engine at flight idle is 80 percent of rated. Because of the small speed variation during flight, a directly driven compressor with no speed changer should be quite satisfactory. The airflow is regulated by the inlet guide vanes of the directly driven compressor. The aircraft pressure is regulated by outflow valves, the same as used on the baseline. Each compressor is adequate to make up for aircraft leakage so that cabin pressurization can be maintained in the event of a failure of one compressor or engine.

The layout of the ECS is as shown in figure 33. The air from the compressor passes through a precooler to reduce the temperature before it enters the system. Because the precooler is needed primarily at the higher altitudes and is not needed on the ground, no precooler fan is provided for ground operation. The air to the cabin is 50 percent recirculated, the same as on the baseline. The cabin ducting, recirculation fans, valves, controls, etc., are all retained from the original system.

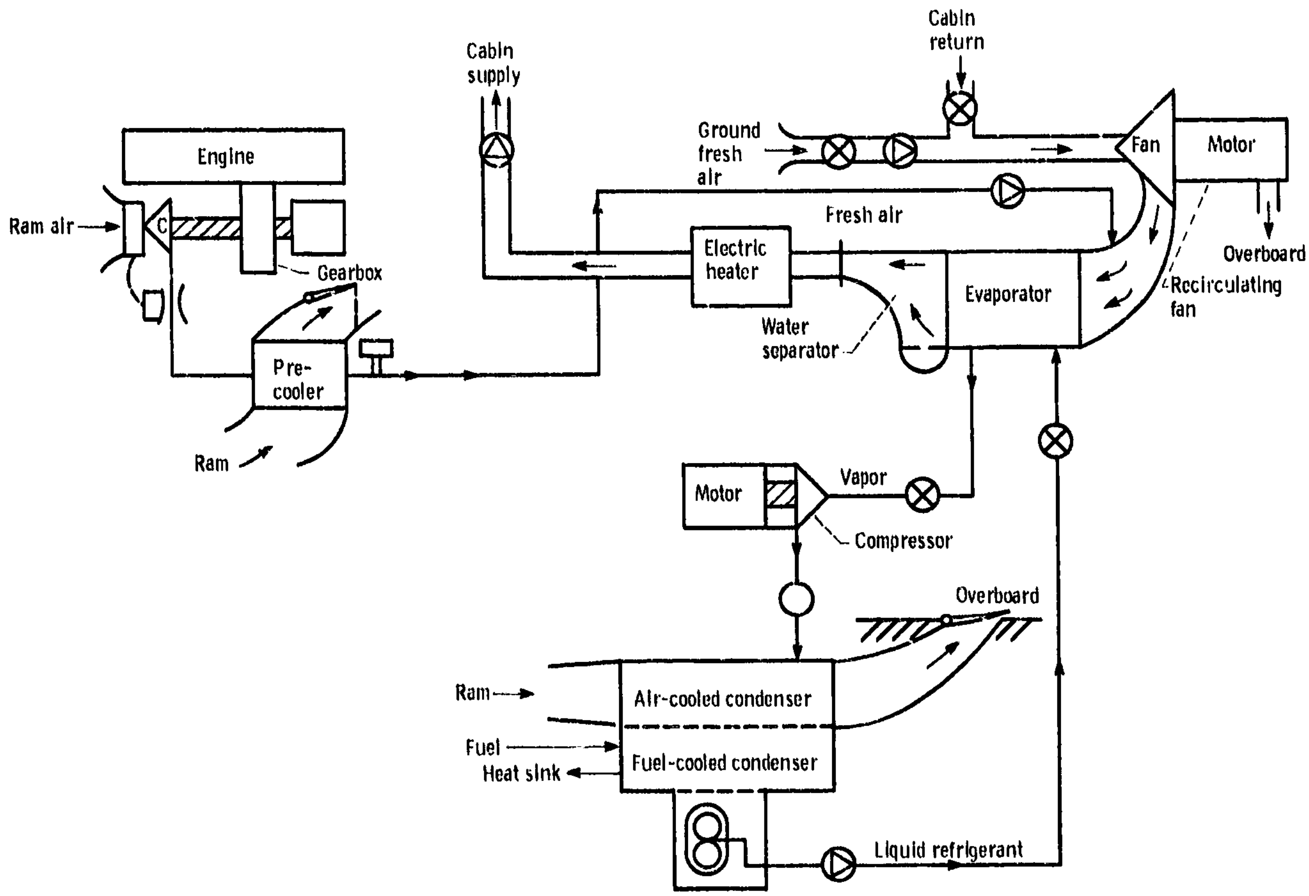

Figure 33.—Schematic of vapor-cycle environmental control system.

Cooling is provided by an electric-motor-driven Freon compressor. The motor can be operated over a wide speed range to accommodate the varying load. A dual heat exchanger is used to condense the vapor. On the ground, fuel is used as the heat sink. The long ground cooling periods are normally at the start of the mission when the tanks contain the maximum amount of fuel. At the end of the mission the remaining fuel is normally cool and the required cooling time is short. During the flight portion of the mission ram air is used as the heat sink.

Hot-day, full-load, sea-level conditions require a 44.8-kW (60-hp) drive motor on the Freon compressor. At the cruise altitude of 10 668 m (35 000 ft) the motor requirement drops to 27.2 kW (36.5 hp). Should one unit fail, operation of the remaining unit at 44.8 kW (60 hp) provides the same 165 percent of rated as the original baseline system.

The portion of the ECS that was replaced weighs approximately 453.5 kg (1000 lb). This covers both systems and includes the engine-driven compressors, precoolers, refrigeration units, drive motors, load receivers, and temperature controls.

Other System Changes

An auxiliary power unit is included in current aircraft to make them independent of ground carts for electric power when the main engines are not running and for pneumatic power to drive the ECS and to start the engines. The APU is also used as a backup electrical generator, enabling the aircraft to be dispatched with an unserviceable engine generator, and as a backup to the electric, pneumatic, and (through the ATDP) hydraulic secondary power systems in flight.

With the advanced all-electric system ground requirements can be met at most airports by using conventional electric utility power. The advanced power system has four primary sources of power: two generators on each engine. As one generator is sufficient for all essential loads, this system, even without an APU, has more inherent reliability than the baseline secondary power system. Therefore an APU was not used in this all-electric airplane configuration. For aircraft use in remote locations an APU can be included in the system. The weight penalty would be less than the baseline APU weight because the compressor to produce pneumatic power would not be required.

The baseline engines are started with air turbines mounted on the engine accessory gearbox. The engine starts are now provided by the electric generators operating as motors. Since they are no longer needed, the air starters were removed.

The actuators for landing gear retraction, nose wheel steering, and thrust reversers were changed from hydraulic to electric. On the basis of the technology of an all-electric braking system (ref. 10) being developed and tested by Goodyear Aerospace Corp., the braking system was also changed from hydraulic to electric.

Description and Performance of All-Electric Aircraft

The all-electric aircraft was designed for the same payload and mission requirements as was the baseline aircraft. However, the baseline hydraulic, pneumatic, and electric systems were replaced with the previously discussed all-electric system. Switching to the all-electric system eliminated all of the customer bleed flow from the engines. However, the horsepower extraction from the high-pressure spool of the engine was increased from 111.9 kW (150 hp) to 194 kW (260 hp) to account for the higher electric power requirements. These changes resulted in a 1.3 percent increase in sea-level static thrust at takeoff power and a 1.5 percent decrease in thrust specific fuel consumption (TSFC) at maximum cruise power.

The baseline flight control system was replaced with a digital fly-by-wire/electric power-by-wire system. The static stability requirement for the advanced all-electric aircraft was assumed to be the same as that for the baseline aircraft. Therefore, although it would be possible to incorporate a stability augmentation system in the advanced flight control system, no such system was included in this study.

As discussed in the previous section, an APU was not included in the design. The hydraulic actuators in the landing gear, brakes, and thrust reversers were replaced with electric actuators. Deicing was accomplished with an advanced electroimpulse system, and the ECS was an electrically driven vapor-cycle system. Since the weight of the starter-generator was accounted for in the electric system, the starter weight was eliminated from the engine installation weight.

Net changes in the system weights before any resizing of the aircraft from those for the baseline airplane are shown in table V. After applying the delta system weight of table V and modifying the engine performance to account for the elimination of bleed and the increase in horsepower extraction, the baseline-size aircraft and engine were allowed to scale such that they once again met the baseline mission and payload. The resized geometry for the all-electric aircraft is given in table VI; and a weight breakdown, in table VII. Note that all of the fixed-equipment weights were assumed to be invariant with aircraft weight. Therefore applying the net change listed in table V to any particular baseline fixed-equipment weight will result in the advanced system weight after the aircraft is resized. However, the weights of many of the other aircraft components were assumed

TABLE V.—NET CHANGES IN SYSTEM WEIGHT FOR ALL-ELECTRIC AIRCRAFT

System	Net weight change	
	kg	lb
Fixed equipment:		
Hydraulic	−1295	−2855
Pneumatic	−354	−780
Electric	+299	+659
Air-conditioning	+47	+104
Anti-icing/deicing	−92	−203
Auxiliary power unit	−676	−1490
Miscellaneous	+23	+50
Flight controls	−695	−1532
Landing gear, steering, and brakes	−131	−288
Starter	−77	−170
Total	−2950	−6505

TABLE VI.—DESCRIPTION OF ALL ELECTRIC AIRCRAFT

Takeoff gross weight, kg (lb)	110 663 (244 019)
Sea-level static thrust per engine, kN (lb)	175.9 (39 542)
Wing area, m^2 (ft^2)	248.3 (2673)
Wing span, m (ft^2)	44.8 (147)
Wing 1/4-chord sweep, rad (deg)	0.55 (31.5)
Wing aspect ratio	8.11
Wing taper ratio	0.267
Wing thickness ratio, root	0.151
Wing thickness ratio, tip	0.103
Wing mean geometric chord, m (ft)	6.2 (20.2)
Horizontal tail area, m^2 (ft^2)	49.0 (527)
Horizontal tail volume coefficient	0.913
Horizontal tail aspect ratio	4.00
Vertical tail area, m^2 (ft^2)	52.4 (564)
Vertical tail volume coefficient	0.095
Vertical tail aspect ratio	0.67
Body length, m (ft)	46.3 (152)
Body width, m (ft)	5.0 (16.4)
Seating abreast	7
Number of aisles	2

TABLE VII.—WEIGHT BREAKDOWN FOR ALL-ELECTRIC AIRCRAFT

	Weight	
	kg	lb
Propulsion group:		
Primary engines	7 591	16 738
Engine installation	1 828	4 031
Fuel system	488	1 077
Structures group:		
Wing	13 899	30 648
Horizontal tail	1 350	2 977
Vertical tail	1 778	3 920
Fuselage	15 226	33 574
Landing gear	6 288	13 865
Primary engine section	2 361	5 206
Flight controls	1 072	2 364
Fixed equipment:		
Instruments	472	1 040
Electric	1 038	2 289
Avionics	721	1 590
Furnishings	7 914	17 450
Air-conditioning	1 022	2 254
Deicing	94	207
Miscellaneous	23	50
Airframe lights	113	250
Fixed useful load	7 997	17 635
Operating-empty weight	71 274	157 165
Payload	17 578	38 760
Fuel	21 811	48 094
Gross weight	110 663	244 019

TABLE VIII.—KEY MISSION PARAMETERS

	Aircraft	
	Baseline	All-electric
Ratio of takeoff thrust to gross weight, N/kg (lbf/lb)	3.343 (0.341)	3.179 (0.324)
Second-segment climb gradient,[a] percent	2.4	2.4
Takeoff distance (to 10.7 m; 35 ft), m (ft)	1425 (4674)	1442 (4730)
Time to climb (to 10 668 m; 35 000 ft), min	17	19
Fuel to climb (to 10 668 m; 35 000 ft), kg (lb)	2512 (5540)	2447 (5369)
Cruise thrust specific fuel consumption (average), kg/N hr (lb/lbf hr)	0.0579 (0.568)	0.0566 (0.555)

[a]One engine inoperative.

to scale as the aircraft was resized. Thus for the flight control and landing gear systems there is a weight saving due to the scaling in addition to the net weight change noted in table V. Key mission parameters for the baseline and the all-electric aircraft are given in table VIII. Note that engines for both aircraft are sized to meet the 2.4 percent second-segment climb gradient with one engine out in icing conditions. However, because of the lower power requirement for electroimpulse deicing, the engine required to meet this climb gradient is substantially smaller. This results in a lower ratio of takeoff thrust to gross weight and a slightly longer takeoff distance and time to climb for the all-electric aircraft. However, these values are well within the desired limits. The fuel to climb was reduced substantially as a result of the lower aircraft weight and the improved TSFC obtained by eliminating bleed. The average cruise TSFC decreased 2.3 percent, greater than the 1.5 percent decrease that resulted from eliminating customer bleed. The additional TSFC improvement resulted from the lower ratio of takeoff thrust to gross weight for the all-electric aircraft. Consequently the all-electric aircraft cruises at a higher power setting, resulting in a higher efficiency.

Summary of Benefits

The weights of the various aircraft systems, for both the baseline and advanced aircraft, are shown in figure 34. The baseline power system, which includes the hydraulic and electric generation and distribution equipment and the pneumatic distribution equipment, weighs 2388 kg (5265 lb) and has a utilization factor at cruise of 0.42. The advanced power system weighs only 1039 kg (2290 lb) and has a cruise utilization factor of 0.62. This system change results in a weight reduction of 1349 kg (2975 lb). One of the reasons for this weight reduction is that using a single type of power allows load sharing throughout the flight profile and this results in a smaller capacity system. However, the major reason that the replacement power system is so much lighter has to do with the characteristics of that system. The high-frequency power distribution bus results in small energy storage components and very simple, lightweight load receivers. The failure protection scheme allows the use of a distributed bus without the penalty of heavy solid-state circuit breakers.

The portion of the baseline flight control system that was removed from the aircraft weighs 1057 kg (2330 lb). This includes the hydraulic actuators, the servovalves, and the mechanical control linkages. The advanced system that replaced the baseline system weighs only 363 kg (800 lb) and includes electromechanical actuators, load receivers, redundant digital data buses, and remote terminals. The net weight reduction for this change was 694 kg (1530 lb). The electromechanical actuators can perform the same function as hydraulic actuators with lower weight and without the sizing restrictions encountered with hydraulic actuators. The use of a digital data bus to provide control and sensor information is a much lighter weight approach than cables, pulleys, bellcranks, etc. Also, with the use of a digital fly-by-wire system, many things previously accomplished with hardware can be done with software. This includes such features as control surface damping and trim.

The replacement ECS at 460 kg (1014 lb) actually weighs more than the original system at 413 kg (910 lb). However, the basic benefit associated with the ECS is elimination of customer bleed from the engine. Generally bleed is withdrawn from the engine compressor at only two fixed ports, one for low and one for high engine power settings. The compressor stage for the high-power port is selected such that it will provide adequate pressure for all but low power conditions. As a result the pressure is normally higher than needed. Also the air is of a higher temperature than desired. All of this excess energy is wasted and contributes to higher than necessary fuel consumption. The new system avoids this waste.

The baseline bleed air anti-icing system, which consists mainly of pipes and valves, weighs 186 kg (410 lb). The

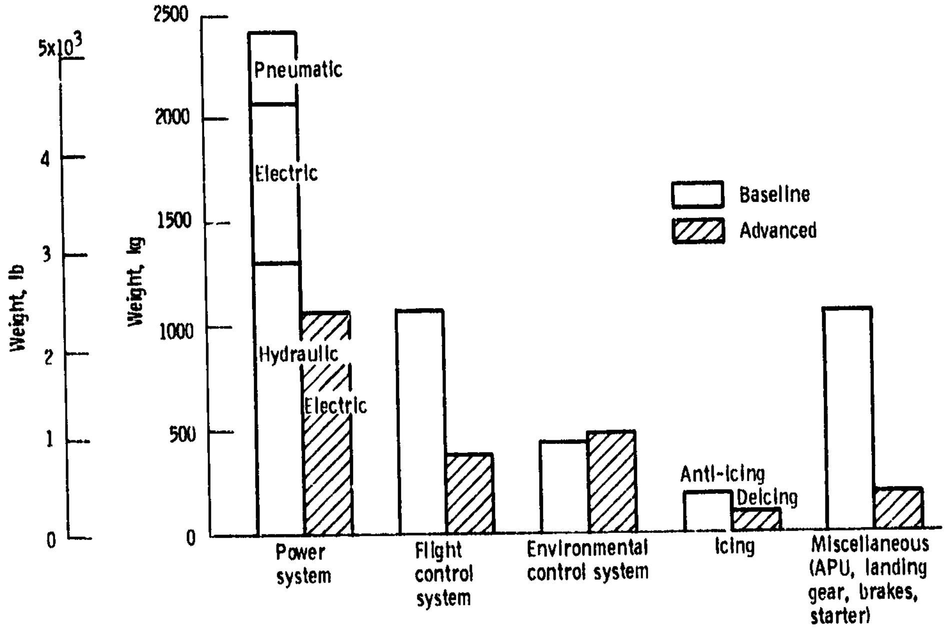

Figure 34.—Weights of aircraft systems.

electroimpulse deicing system that replaced it weighs 94 kg (207 lb). This weight difference is not very significant. Also the fuel penalty for bleeding the engines during anti-icing tends to be insignificant because bleeding is done so seldom. However, the engine must be sized to be able to provide that bleed when needed, even with one engine inoperative. As a result, the engines and the aircraft are larger and heavier than if bleed for anti-icing did not have to be provided. The new deicing system reduces fuel consumption since the engines and aircraft can be smaller.

Additional weight was saved in a number of other areas. The removal of the APU eliminated 676 kg (1490 lb). (If an APU were included in the advanced system, it would add about 455 kg (1000 lb).) The air starters (77 kg; 170 lb) were also removed. The landing gear, steering, brakes, and thrust-reverser actuation were changed from hydraulic to electric. This saved another 131 kg (288 lb).

The total weight reduction in the aircraft systems was 2950 kg (6505 lb). This weight reduction along with the bleed elimination was factored into aircraft size for the same mission (table IX). The engine thrust reduction comes about because of the aircraft weight reduction and also because the all-electric version does not have to provide icing bleed during single-engine climbout. Note that the percentage of reduction in fuel consumption does not vary appreciably with mission length. (If an APU were included in the all-electric aircraft, the empty weight would increase by about 910 kg (2000 lb) and the fuel saving would decrease by about 0.6 percent.)

The reductions in weight and fuel consumption are summarized in figure 35. Block A represents the 2950-kg (6505-lb) reduction in the aircraft secondary systems directly attributed to advanced secondary power system application. When that is factored into the entire aircraft, its structure and propulsion system weight are reduced by an additional 2948 kg (6500 lb) (block B). The reduction in fuel consumption associated with this weight saving amounts to approximately 4.5 percent of baseline (block D). Another weight saving of approximately 1995 kg (4000 lb) is associated with the propulsion and structure reduction for not having to bleed the engines (block C). The fuel saving associated with this weight reduction and the elimination of customer bleed from the engine is also approximately 4.5 percent of baseline (block E). This yields a total gross weight saving of 10 140 kg (22 360 lb).

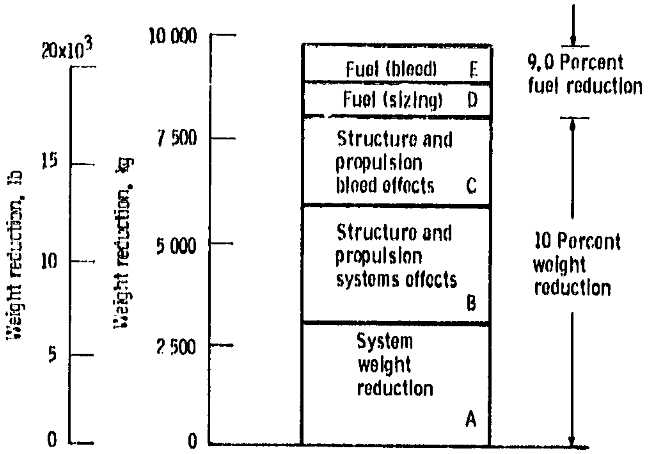

Figure 35.—Reduction in weight and fuel consumption from baseline, for design mission.

In conclusion, the total benefit for a 200-passenger, twin-engine transport using energy-efficient engines is a 9 percent reduction in fuel consumption and a 10 percent reduction in operating-empty weight.

Conclusions

This study showed that substantial benefits, in the form of reductions in aircraft weight and fuel consumption, are possible through the integration of an advanced electric secondary power system into a transport aircraft. The particular benefits achieved, 10 percent reduction in aircraft empty weight and 9 percent reduction in mission fuel, are valid for the aircraft selected for this study. Any medium or large transport aircraft using the same type of power system should achieve similar benefits. The benefits also depend on the type of engine selected, with higher bypass ratio engines providing higher benefits because they are more sensitive to bleed. The engine selected for this study, while an engine of the future, is sufficiently near term that it is expected to be in service at the time the power system technology would be available. Therefore the benefits

TABLE IX.—AIRCRAFT WEIGHT

	Baseline		All-electric		Change, percent
	kg	lb	kg	lb	
Empty weight	78 966	174 125	71 274	157 165	−9.74
Engine thrust	20 552	45 319	17 932	39 542	−12.75
Total fuel	24 057	53 048	21 811	48 094	−9.34
Fuel consumption:					
3590-km (1940-n mi) mission	17 400	38 368	15 859	34 971	−8.85
740-km (400-n mi) mission	4 949	10 912	4 516	9 958	−8.74

resulting from the engine bleed sensitivity are realistic. With the engine trends in transport aircraft moving toward even higher bypass ratios, the benefits associated with eliminating customer bleed will increase beyond the values achieved in this study.

The benefits achieved in this study occurred in several systems within the aircraft. The most substantial weight reduction (1349 kg; 2975 lb) was in the secondary power system. The baseline power system was composed of three separate systems: hydraulic, pneumatic, and electric. Each system powered its own loads. Their low utilization resulted in a fairly heavy secondary power system. However, the replacement system powered all the loads, and this allowed load sharing and better system utilization. Also the technology used in the replacement system provided a very lightweight system. The net result was a substantially lighter secondary power system.

Another system showing a substantial weight reduction (694 kg; 1530 lb) was the flight control system. This saving was obtained when the hydraulic actuators and servovalves were replaced with electromechanical actuators and load receivers. The benefit is larger than might normally be expected because of the lightweight load receivers resulting from the type of electric power system used. This is an instance where the secondary power system has a beneficial effect on another aircraft system.

Eliminating engine bleed for powering the environmental control system (ECS) provided the largest single fuel saving of any change made in this study. The ECS is run by an electric-motor-driven vapor-cycle system weighing only slightly more than the baseline system. As a result the fuel saving was not even partially offset by a weight increase. A major factor allowing this lightweight ECS is the lightweight load receiver used to control the electric drive motor. This is another instance where the secondary power system technology has a beneficial effect on another system.

The electric secondary power system must provide the proper interface for motor loads. Loads that were previously hydraulic and pneumatic will be driven almost exclusively with electric motors. The actuators in the flight control system are driven by many small motors. These motors must operate over a wide speed range in order to respond to the flight control commands. The synthesized output of the load receiver provides the frequency variability necessary to meet this requirement. The ECS vapor-cycle compressor will be driven by a large motor, almost equal in size to the power system generators. These motors must be soft started to prevent large variations of the power bus voltage. The constant voltage-to-frequency ratio output of the load receiver provides this soft-start capability. Therefore the secondary power system selected in this study does provide the proper interface for motor loads.

The secondary power system characteristics are the key to the benefits obtained in this study. The basic characteristics of the system are high-frequency (20 kHz) sinusoidal power distribution with controlled energy flow. The high frequency results in lightweight magnetics and other energy storage components. The sinusoidal waveshape is well ordered and easily controlled with a simple control system. Because of its single frequency electromagnetic interference protection is simple. Controlled energy flow allows the use of lightweight isolation and reconfiguration switches. All of these features result in a lightweight secondary power system. The high-frequency waveform can easily synthesize lower frequencies when a lightweight load receiver is used. This provides the range of frequencies required by the flight control system actuators. The result is a lightweight flight control system. The synthesized outputs can provide a constant voltage-to-frequency ratio for motor drives. This allows the controlled starting of the engines and the ECS drive motors. The result is an efficient system for bleed elimination.

To determine the effect of an electric secondary power system on an aircraft, it must be integrated into the aircraft and all those systems with which it interfaces. The power system can have a significant effect on other systems in the aircraft, such as the flight control and environmental control systems. All of these systems in turn affect the aircraft and engines, as can be seen by the final results of this study.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, January 25, 1985

Appendix—Baseline-Aircraft Flight Control

Longitudinal Control System

Single-span elevators (fig. 36) and a variable-incidence horizontal stabilizer (fig. 37) control the pitch of the baseline aircraft. Three hydraulic actuators drive the right and left elevators. The actuators receive their inputs from either the autopilot or the control columns via the cable system. All three hydraulic systems power each control surface to ensure continued operation of the elevator control system when two of the three hydraulic systems are inoperative.

The fully powered control system requires artificial "feel" forces at the pilot/copilot's control column. A combination of mechanical and hydraulic springs contained in the feel unit provides these forces. The feel computer programs hydraulic pressure for the feel unit actuators as a function of measured impact pressure and stabilizer position. Switching on the autopilot hydraulically actuates an autopilot-engage locking mechanism. The locking mechanism allows the autopilot actuator to drive the input to the surface actuators and to backdrive the control column.

The horizontal trim control system varies the angle of incidence of the stabilizer to provide longitudinal trim. The stabilizer is trimmed in manual or automatic mode, by a dual-load-path ballscrew actuator powered by two hydraulic motors via a differential gearbox. When trim is not commanded, stabilizer position is maintained by a primary thrust brake (a self-actuating mechanical brake that maintains stabilizer position in the absence of hydraulic pressure) and two secondary hydraulic brakes. The total stabilizer travel is minimized by using programmed elevator deflection with stabilizer angle.

In the manual mode both hydraulic modules and the associated hydraulic motor and brake respond to trim commands. In the autopilot mode trim commands are only transmitted to one of the two control modules when the elevator command exceeds ±0.007 rad (0.4 deg) from neutral for more than 5 s. In the autoland mode the remaining module is in a standby state; in the event the

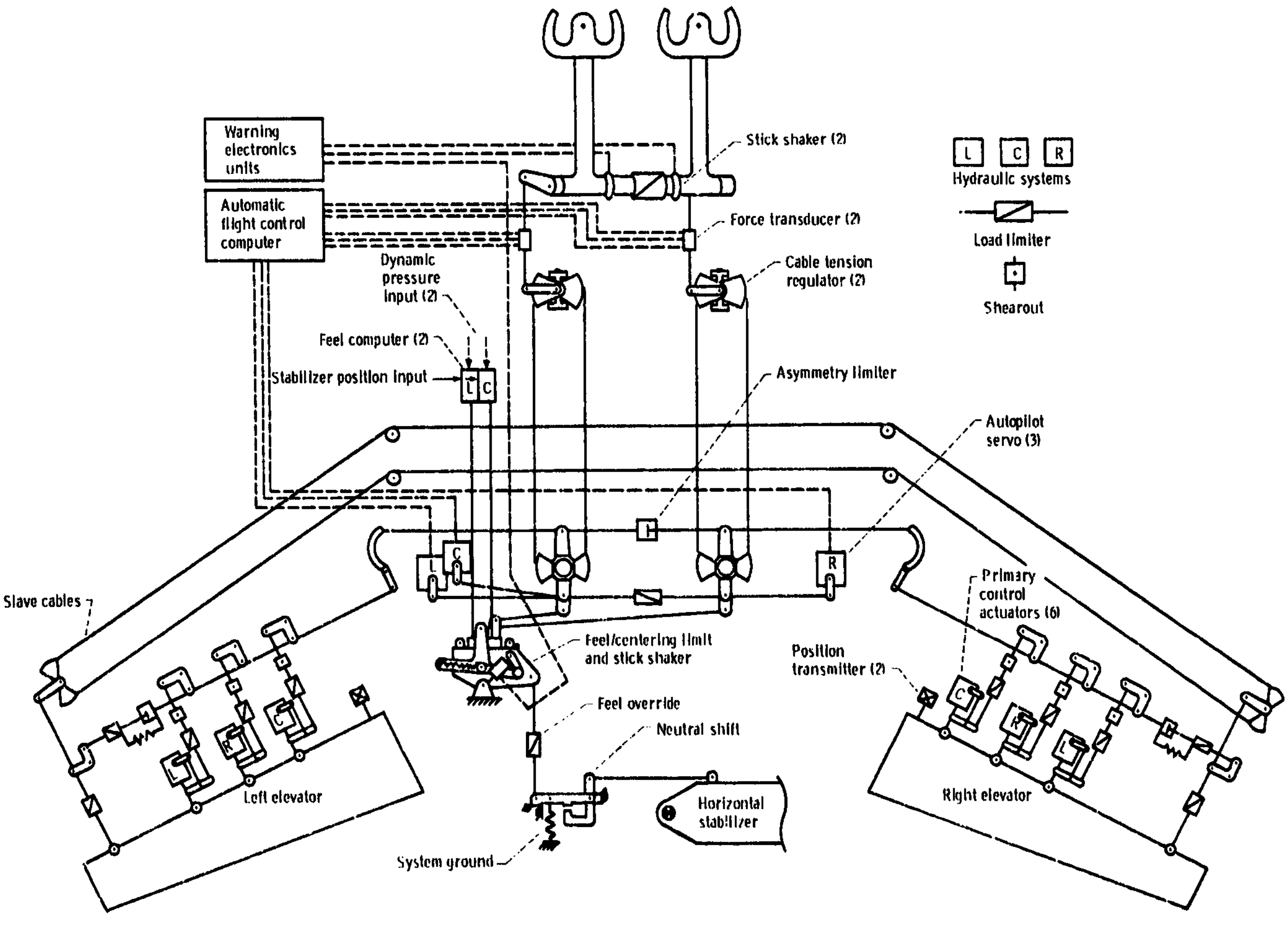

Figure 36.—Schematic of baseline elevator control.

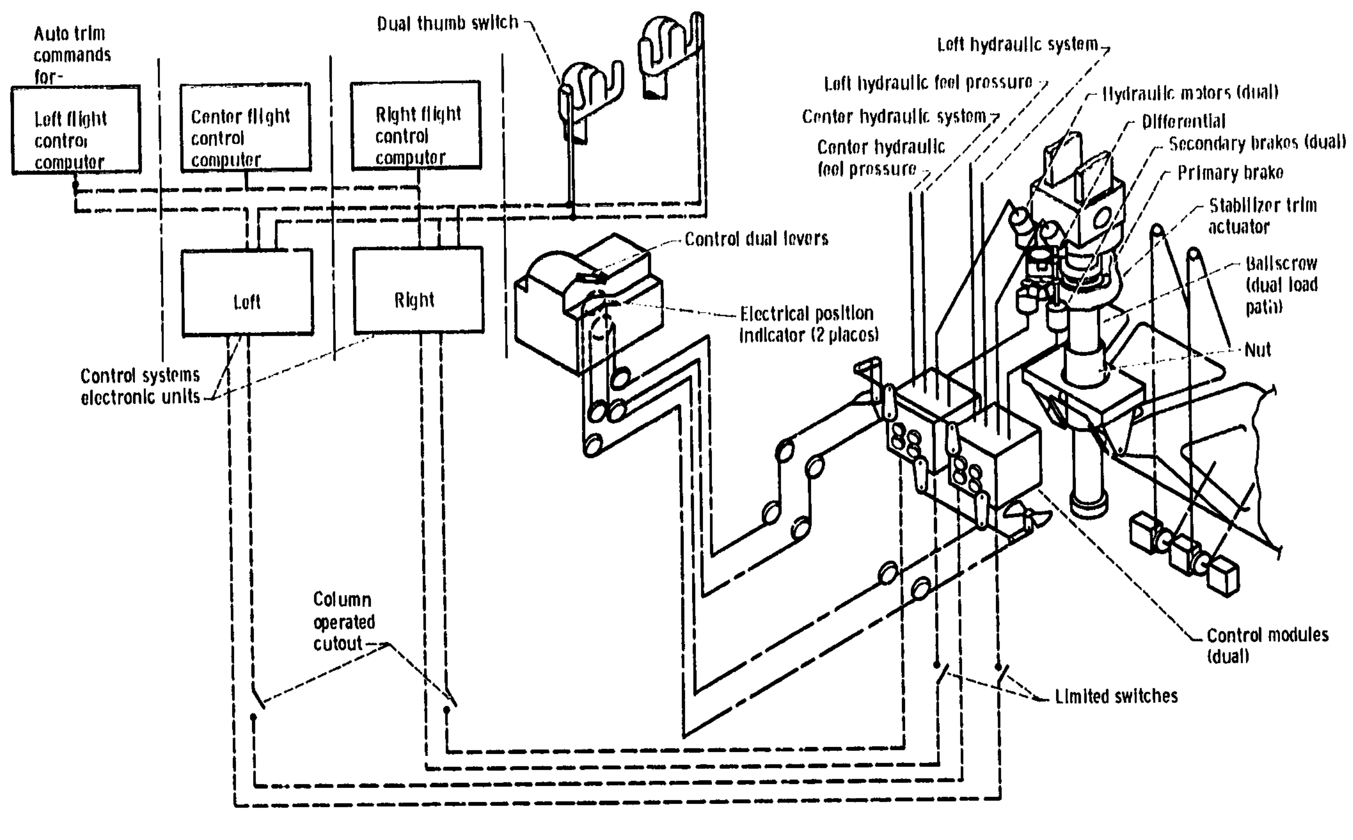

Figure 37.—Schematic of baseline stabilizer trim control system.

utilized channel sustains a malfunction, control is automatically transfered to the standby channel. A stick shaker mechanism, consisting of an electric motor driving an unbalanced mass, is located on each control column.

Lateral Control System

Two ailerons and six spoilers on each wing provide lateral control. The input signal from the pilot/copilot's control wheel is transmitted through a cable system to the central control actuators. These actuators, located in the main wheel well, drive the input cables to position the aileron surface actuators. The control wheels also drive transducers that send electrical signals to control modules in the control system electronics unit (CSEU). The CSEU sends electrical signals to position the spoilers. The aileron and spoiler control systems are shown schematically in figures 38 and 39. A transfer mechanism located at the base of the copilot's column links the pilot's and copilot's control wheels. Thus either control wheel can operate independently if the other side jams.

The ailerons are trailing-edge surface controls located inboard and outboard of the outboard flaps on each wing. The lateral control system employs three central control actuators (CCA), mounted in parallel. The CCA's amplify the lateral control signal forces exerted by the pilot and contain the autopilot input systems. When the autopilot is engaged, an integral autopilot servo in any CCA controls inputs to the CCA control-boost servos that actuate the aileron control system. The autopilot servo also backdrives the control wheels, where position transducer signals are generated to control the electrically commanded spoiler actuators. In the manual mode the control wheel positions an input lever on the central actuator that provides feel and lateral trim through a center positioning mechanism. The outboard aileron's lockout control subsystem permits full travel of the outboard ailerons as a function of control wheel deflection at low aircraft speeds and holds the outboard aileron in a faired position at high aircraft speeds. The elements of this subsystem are the CSEU and the outboard aileron's lockout actuators and lockout mechanisms.

Six flight spoilers are located on each upper wing surface just forward of the trailing-edge flaps. Each spoiler panel is operated by its own hydraulic actuator. At high speeds panels 4, 5, 8, and 9 are locked out while the remaining panels react, similar to the low-speed condition. As flight spoilers the surfaces operate in conjunction with the ailerons to provide lateral control of the aircraft. Each spoiler actuator receives its input by an electrical signal generated by the CSEU. The CSEU uses the control wheel deflection, speed-brake lever position,

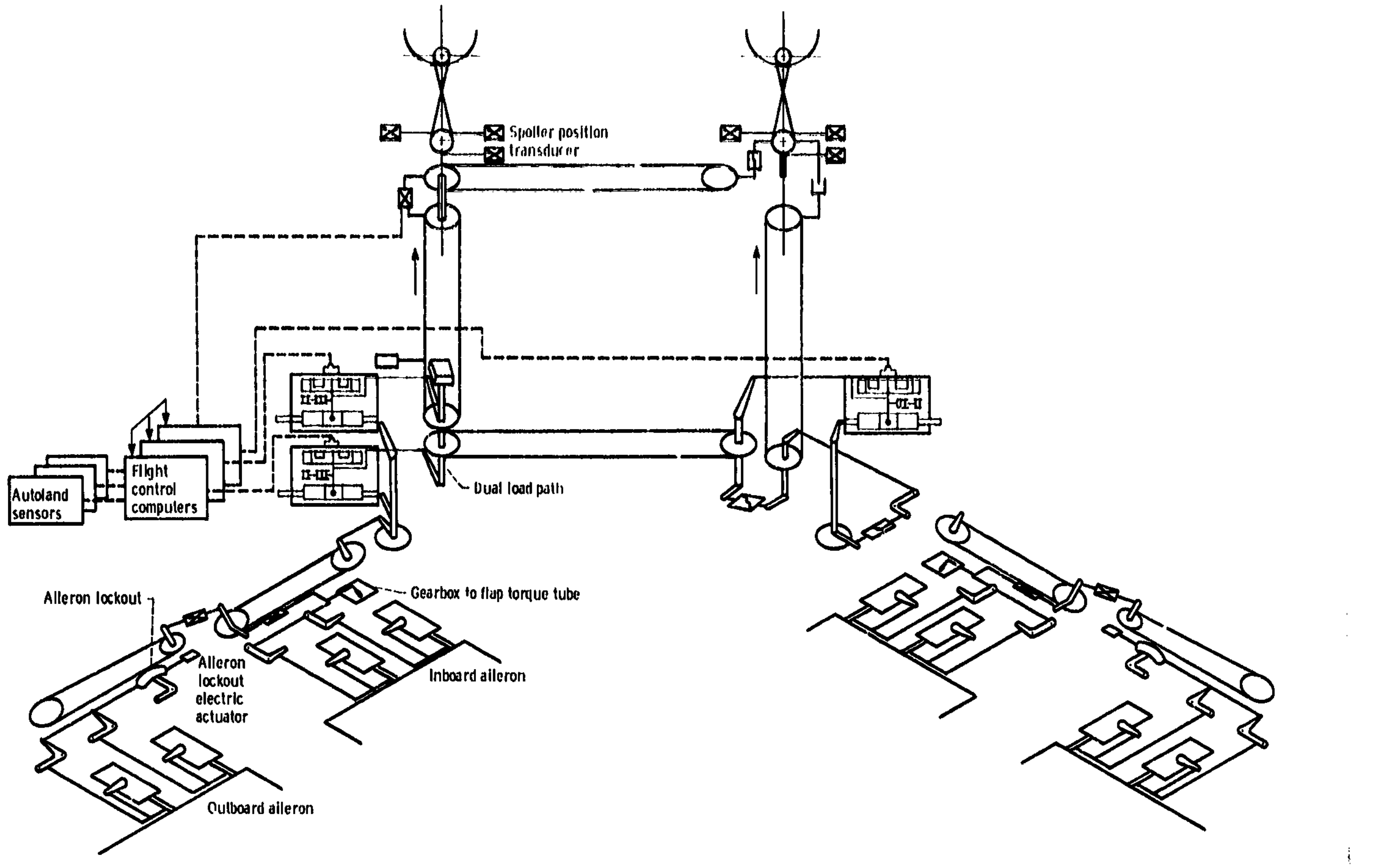

Figure 38.—Schematic of baseline lateral control system.

and air/ground status to control spoiler panel position. The position of each spoiler panel is sensed by an LVDT, whose output is used by the CSEU to close the servoloops. When used as ground speed brakes, all panels are deflected to 1.05 rad (60°). When used as in-flight speed brakes, panels 1, 2, 3, 10, 11, and 12 are deflected to 0.083 rad (46°) and panels 5, 6, 7, and 8 are deflected to 0.366 rad (21°) with panels 4 and 9 locked out. Each channel of the spoiler control has three sets of electronics comprising an active, standby, and monitor subchannel. Each channel is separate from and independent of all other channels, and each controls a spoiler pair.

Directional Control System

The rudder control system (fig. 40) provides directional control of the yaw axis. Three hydraulic actuators power the rudder. The rudder control system is operated by displacing either the pilot's or copilot's pedals. The cable system transfers the pedal motion to the aft quadrant. The motion then is transferred through a feel unit and a ratio changer to position the control valve within the actuators.

The rudder feel unit is a spring-loaded cam and follower device that provides a pedal force proportional to pedal position. The ratio changer is a controlled actuator that changes the ratio of pedal travel to surface displacement as a function of airspeed. This limits the amount of rudder that can be commanded at higher airspeed. The ratio changer control subsystem is part of the CSEU. Two monitored channels receive data from the air data computers. If a failure is detected in the normal channel, control is transferred to the standby channel. If the second channel fails, pressure to the actuator is shut off and bias springs drive the actuator to the low speed ratio. With a failed ratio changer hydraulic pressure to the rudder actuator powered by the left hydraulic system is shut off, thereby providing only two rudder actuators and limiting rudder travel.

Turning the rudder trim knob changes the rudder trim position. The knob activates an electric actuator to reposition the "feel," centering, and trim mechanisms. Two series yaw damper hydraulic actuators introduce a yaw damping signal, proportional to yaw rate, into the linkage system. The rudder autopilot system (rollout guidance) consists of three separate hydraulic servos, each controlled by independent electronic channels. The autopilot servos are connected to the aft quadrant by independent signals.

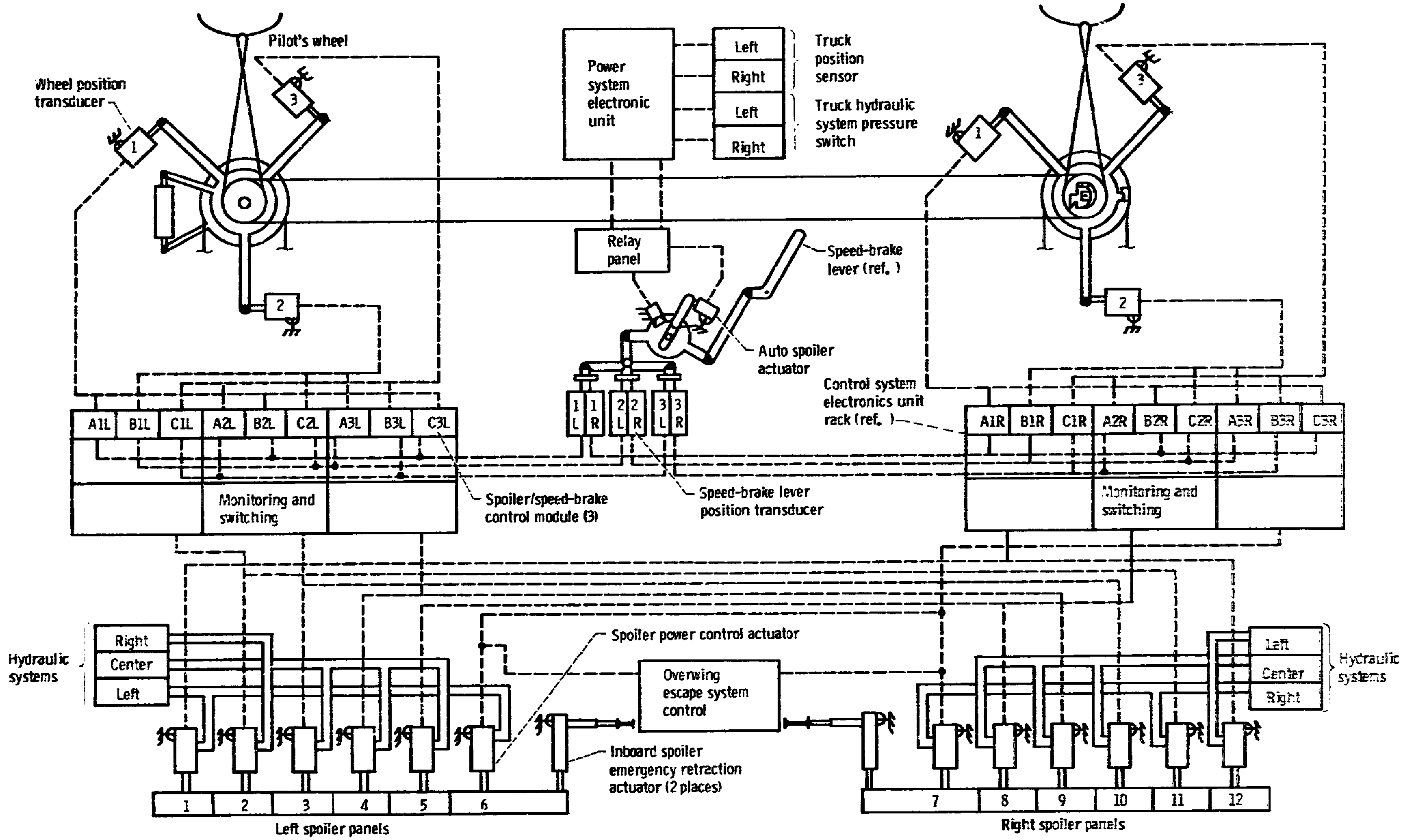

Figure 39.—Schematic of baseline spoiler control system.

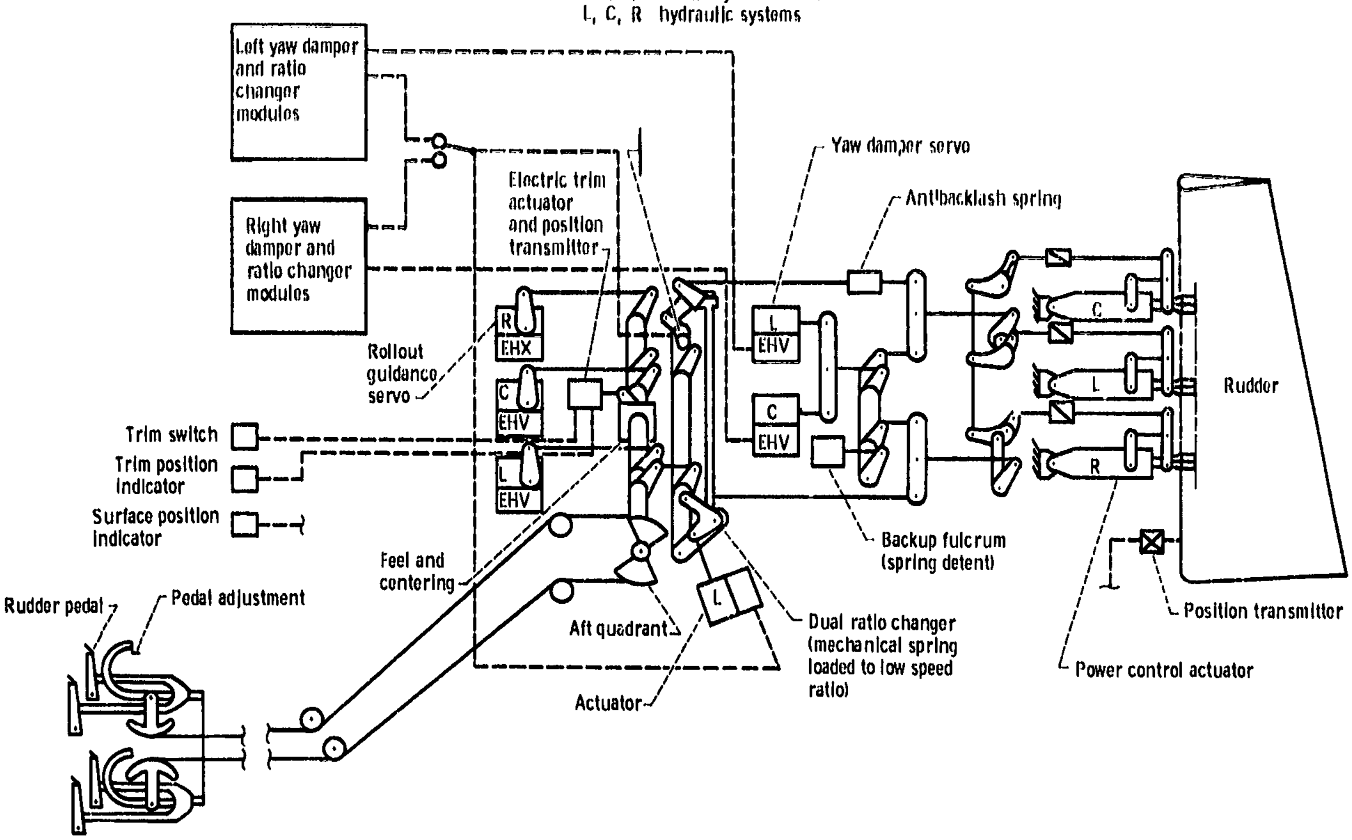

Figure 40.—Schematic of baseline rudder control system.

High-Lift Control System

The high-lift system (fig. 41) includes a double-slotted inboard flap and a single-slotted outboard flap, plus five outboard and one inboard three-position, leading-edge slats on each wing. The trailing-edge flaps and the leading-edge slats provide high lift for takeoff and landing. During normal operation the center hydraulic system is the primary source of power. In the event of hydraulic system failure the alternative power source is electric motors.

Each power drive unit (PDU) is a hydromechanical servomechanism that consists of a control valve module, a hydraulic motor, and a gear reduction unit. The PDU's transmit power to the rotary actuators via torque tube drive shafts and ancillary angle gearboxes. Alternative flap/slat control is provided by an electric motor drive assembly for each PDU. The alternative control loops for the backup electric motors are electromechanical closed loops with position sensors.

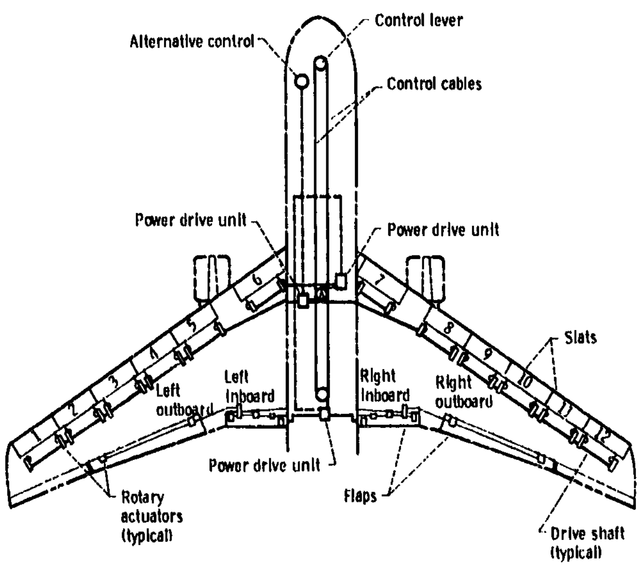

Figure 41.—Schematic of baseline high-lift control system.

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15. Supplementary Notes

16. Abstract

A concept for an advanced aircraft power system was identified that uses 20-kHz, 440-V, sine-wave power distribution. This system was integrated with an electrically powered flight control system and with other aircraft systems requiring secondary power. The resulting all-electric secondary power configuration reduced the empty weight of a modern 200-passenger, twin-engine transport by 10 percent and the mission fuel by 9 percent.

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