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FULL AUTHORITY DIGITAL ELECTRONIC CONTROL FOR THE PEGASUS POWERED HARRIER/AV8 AIRCRAFT

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1.0 SUMMARY

A Harrier aircraft (designated AV8-A in USMC service) flew in March 1982 with a full authority digital electronic fuel control. This was the first time any aircraft has been entirely dependent upon such a system for its safety of flight. The system is tailored to the requirement of the AV8-B aircraft and has evolved over 10 years of study and practical testing on Pegasus engines. It is the culmination of 23 years association of its designers with fixed wing VSTOL operators.

The use of full authority digital electronic control (FADEC) results in significant improvements compared with the current aircraft and, specifically, in improved control, reduced frequency of maintenance action, reduced crew workload, better repeatability, enhanced safety, greater mission effectiveness and increased operational readiness. The paper discusses and explains how each of these improvements is achieved.

The paper presents practical test results illustrating the performance of the system in various operating modes. It also illustrates failure responses of the system with step-free recovery from various faults.

The operating environment for the electronic section of the control is very severe. Special design methods have been used to give good thermal characteristics combined with mechanical integrity and tolerance of the high EMC demands often associated with aircraft carrier operation. These features are described in the paper which also deals with the maintainability/philosophy and the methods adopted to obtain high probability of fault detection and fault isolation.

Plenum Chamber Burning (PCB) will be the major function development in VSTOL operation with engines configured like the Pegasus. The paper ends by briefly discussing this type of operation and the control modes and control elements which are involved. It details the changes and extensions required of the dry engine system.

2.0 INTRODUCTION

Vertical take-off aircraft take two basic forms:-

- Rotor powered.
- Jet lift powered.

In spite of the common operating outcome in the ability to hover an aircraft, the two types of engine have characteristically different types of control. These differences are extensive and predominantly fundamental.

They stem from the different features of VTOL operation for each vehicle. Helicopters have relatively low power loading and are low speed vehicles. Jetlift aircraft have high thrust to weight ratios and are capable of very high speeds. The configuration, the control and the powerplants are different for the two types of aircraft as well as their operational characteristics. The more significant features of the VTOL operation are compared for rotary wing and powered lift aircraft in Table 1.

TABLE 1

FEATURES OF VTOL OPERATION

ROTARY WING	POWERED LIFT
. TURBO-SHAFT ENGINE	. TURBO-FAN ENGINE
. PREDOMINANTLY TWIN ENGINES	. ONLY SINGLE ENGINES TO DATE
. ENGINES MECHANICALLY COUPLED	. MECHANICAL COUPLING NOT YET PRACTICAL
. NO SPECIAL ATTITUDE CONTROL IN HOVER	. REACTION CONTROL OF ATTITUDE IN HOVER
. POSITIVE GROUND EFFECT	. NEGATIVE GROUND EFFECT
. LOW-SPEED AIRCRAFT	. HIGH SPEED COMBAT AIRCRAFT

Helicopter engines are smaller for the size of vehicle than those required to provide the high thrust/weight ratios for powered lift and high speed. Surprisingly, the torque/inertia ratio for these two types are similar although the sizes of the engines are different. The control response times are therefore of the same order.

2.0 Continued:

Powered lift aircraft benefit from no aerodynamic forces when in hover and zero or very low forward speed. The only way of controlling attitude is to use reaction controls. These take the form of small jets, at three or four points on the aircraft. In the Harrier, these jets are located in the wing tips, in the nose and in the tail of the aircraft. The medium used to generate the reaction forces from the nozzles is air bled from the engine compressor. As a result, the engine operation in hover for powered lift can differ from that in forward flight and it is important that varying bleed off-take for reaction control does not affect the thrust from the engine.

Powered lift aircraft, have relatively poor glide capability because, even in conventional landing, wing lift is augmented by vectored thrust and high wing loadings are normal. Not only is the glide capability limited, but it can only be entered from forward flight. It follows that safety considerations in vertical flight are more onerous for the powered lift type of aircraft.

The only powered lift aircraft in service are the Harrier/AV8A and the Russian Yak-36 Forger. The Harrier is a single-engined aircraft, the other is multi-engined but in terms of safety it corresponds to a single engine because a failure in either a lift engine or the vectored thrust engine is catastrophic during the hover.

Primary control of a powered lift engine must, of course, be of thrust. The variable which most closely approximates to thrust is fan speed. The engine will require overstress protection of several types, e.g. overspeed protection, over temperature protection and, in some cases, over pressure protection. The need for minimum thrust disturbance resulting from bleed transient for reaction control has been explained. It is the reingestion of exhaust gas that gives rise to the negative ground effects. The jet effluxes can also lead to ground erosion and the ingestion of foreign objects into the engine. The control should therefore be more tolerant than usual of foreign object damage. There is some evidence that VTOL aircraft are more susceptible to bird strike than are conventional aircrarft. Both of these factors influence a choice of control architecture which makes the function tolerant of degradation in engine performance.

3.0 BENEFITS OF DIGITAL CONTROL

The introduction of Full Authority Digital Control provides many benefits. These benefits can be allocated under two headings. Many of them provide a direct reduction in life cycle cost. Other benefits are intangible and cannot be readily quantified under a cost saving. The life cycle cost benefits can be grouped under a number of headings.

- Improved Control and Reduced Crew Workload
- Improved Maintainability
- Improved Operational Readiness
- Improved Mission Effectiveness
- Lower Costs.

3.1 IMPROVED CONTROL AND REDUCED CREW WORKLOAD

Improved control derives from several features of the digital system. Observation of limits is more consistent and precise than in a conventional system. It therefore allows the system to operate more closely to the ideal limits. The control developed for the Pegasus engine uses a non-dimensional closed-loop acceleration control. This control produces consistent and repeatable accelerations regardless of fuel type, fuel temperature or engine condition. The control uses schedules involving relationships between several variables and allows more accurate and repeatable control over the full flight envelope. It provides the potential for performance improvement and extension of the flight envelope beyond that usable with simpler systems as discussed later.

The system also provides automatic rating selection for the pilot under all operating conditions in vertical or conventional flight and in wet or dry operation of the engine. These features constitute a significant improvement in the control capability.

3.2 IMPROVED MAINTAINABILITY

The system exploits the capability of the digital microprocessor to detect and diagnose failure. The diagnosis is supported by BITE indicators mounted on the case of the electronics units.

The control laws have been selected to provide response which is independent of fuel type or temperature and is also insensitive to engine performance variation. This choice has allowed all in-service adjustments to be eliminated. Once installed, the control never needs to be reset.

3.3 IMPROVED OPERATIONAL READINESS

The operational readiness of the aircraft is enhanced in several ways. First, there is the elimination of in-service adjustments mentioned above. This feature also eliminates the need for confirmatory flight tests following adjustment. It therefore reduces the fleet fuel-burn as well as increases the aircraft availability. The use of built-in test indicators provides direct diagnosis of faults at the flight line and eliminates most unnecessary line replaceable unit removals. The time required to replace and diagnose an electronic defect is significantly shorter than for a hydromechanical unit. All of these factors reduce the total maintenance time required on the aircraft and enhance its operational readiness.

3.4 IMPROVED MISSION EFFECTIVENESS

The mission effectiveness is improved in several ways. The improved readiness mentioned above leads to higher system availability to be despatched on a mission. The organisation of the system largely frees the pilot from direct involvement in operating the engine during flight. This reduction in his workload permits a significant improvement in the effectiveness with which he can pursue his mission. Mission effectiveness is further enhanced by the performance improvements mentioned earlier. The structure of the system provides for redundant electronic controls with very low probability of causing an engine shut-down. Furthermore, the complexity of the hydromechanical fuel handling section of the control is reduced. These two features lead to a reduced in-flight engine shut-down rate and reduced mission failure rate. The combination of these features leads to a significant improvement in mission effectiveness.

3.5 LOWER COSTS

Operating costs are reduced in several respects. There is a significant fuel saving resulting from the elimination of confirmatory flight test following maintenance adjustment. The improved precision and repeatability of the control particularly in accelerations and in limiting extends the engine life for the same operating methods. Lower system failure rates and reduced pilot workload can be expected to result in lower attrition rate for the aircraft. Clearly the reduced maintenance actions affect not only the direct maintenance costs but also the spares inventory required for the system. The digital configuration provides a direct, simple and reliable interface with other avionic systems. This type of interface would be additional hardware with a conventional hydromechanical control. Finally because of the relative simplicity of conducting modifications in software, the non-recurring functional development cost and modification costs in service are reduced.

4.0 THE PRESENT SYSTEM

The present system on the Pegasus engine is a hydromechanical control, with an electronic temperature limiter. The hydromechanical control system for the engine is unusual. It has an emergency system fitted. The emergency system is required because any failure in the engine control itself which leads to loss of thrust or engine shut-down would have a catastrophic effect for the aircraft. The emergency system was fitted with the control from the outset, and as yet no accident of the aircraft has been attributed to control failure.

The emergency system itself is very simple. The control of the engine is by modulation of a valve coupled to the shut-off valve and directly operated by the pilots lever as shown in Figure 1. A changeover valve by-passes and disconnects the main metering valve in order to engage the emergency system. It also disconnects the by-pass valve for the limiter, and back-flow is prevented by a check valve in the downstream line from the metering valve delivery. In normal operation the shut-off valve is fully open and the manual flow control is therefore set to a flow determined by the lever position. For changeover from the normal operation to emergency operation the manual flow control is returned to the idle stop by means of the pilot's lever, and the changeover valve is operated by means of switched solenoid. Once the changeover valve has moved to the emergency position then the throttle lever can be advanced to an appropriate thrust setting. This procedure is required in order to prevent surge were the changeover valve to operate while the manual flow control was selecting a high flow.

Figure 2 shows the block diagram for the system with the selection of normal operation and the hydromechanical control plus temperature limiter to the manual flow control operation. The electronic limiter has a multiple datum selection and a muting switch.

The system was designed in the late 1950's. A design objective was that the system be modular with each module individually changeable. The new system shows a significant weight saving over the present system.

DSIC initiated a demonstrator programme of digital control for Pegasus in 1979. This programme is proceeding in two phases. Each phase has different hardware.

4.0 Continued:

A design target was to provide automatic failure response to an electronic failure. Three methods were possible:

a) Automatic changeover to a back-up manual control
b) Automatic changeover to a back-up electronic control
c) Automatic changeover to a second electronic control of identical performance.

All three methods have been assessed. The first was rejected because control reliability was eroded by the added complexity needed to ensure safety. No satisfactory resolution of this conflict was found although its feasibility was demonstrated on a test bed engine.

The second and third methods are embodied in the demonstrator programme.

5.0 DEMONSTRATOR SYSTEM

The Phase I system is shown in Figure 3. It uses two electronic control lanes, one of full capability, and one of reduced capability. The system normally operates using the main digital electronic control, but switches automatically to the emergency control when the main electronics fails. The probability of both the electronic controls failing is very much lower than the probability that the hydromechanical control which they replace fails. Frequency of reversion to the manual flow control is therefore now lower than with the existing system, and automatic changeover to it is no longer required. The same manual flow control as currently used can therefore be retained with this system. The main area of interest in the design is the definition of the capability of the partial control lane and the ability to derive other alternative systems from it. The main alternatives seen initially were:-

- Single channel electronic control with automatic hydromechanical reversion to a P1 compensated system.
- A '1½' lane system with the existing manual flow control, (i.e. as the Phase I demonstrator).
- Full dual lane system with existing manual flow control.

The first system is simple and reliable, but has reduced reversionary capability and mission success rate. The last configuration has a high defect rate, retains full reversionary capability and full mission capability after a failure. The intermediate system is relatively simple and, depending on negotiable complexity, can either retain a fair degree of mission capability or can be elementary.

One objective of Phase I was to define the system for Phase 2.

The system comprises two major components as shown in Figure 4.

5.1 INSTALLATION

The electronics module shown in Figure 4 is engine mounted, as illustrated in Figure 5. It is fuel cooled as a single unit and the electronics are withdrawn as a single block. The system is designed to be compatible with both AV-8B Harrier II and the existing Harriers.

The environment in which the control system is located is extremely hostile. Bay temperatures of up to 150°C can occur for short periods after shut-down, while heat is soaking from the engine into the engine bay. In normal operation, the fuel temperature used for cooling is low, but it can peak to temperatures above 80°C for fairly long periods during some types of operation.

5.1 Continued:

The main area of attention in this design is keeping good thermal paths between the components themselves and the coolant. Particular attention has been given to areas of metal to metal contact within the design of the unit and thermal washers and gaskets are used in order to secure low thermal impedances at critical points in the design.

The method of assembly, illustrated in Figure 6, uses metal frames which carry pairs of circuit boards. Each board has a copper facing which runs under the components and is clamped between the board and the frame. The frame itself is clamped to the case through which fuel circulates. The fuel passage lies immediately below the shoulder on which the frame is clamped. This arrangement provides the minimum number of interfaces and the shortest possible path between individual components and the fuel. The frame also stiffens the whole assembly and improves its mechanical integrity.

The front section of the case forms an enclosure behind the connectors. EMC filters are mounted on a bulkhead behind the back of this enclosure.

The power supply is installed in a second enclosed volume on one side of the unit. This arrangement provides two benefits. First, it excludes from the main electronics compartment interference generated by the switching regulators in the power supply. Secondly it allows high dissipation components to be mounted directly on the case.

Figure 7 shows one of the demonstrator units opened and the modules separated to expose all components. The wiring shown is strictly a feature of the prototype construction. Flexible film wiring would be used in production.

5.2 CONTROL LAWS AND PERFORMANCE

Figure 8 shows the control functions provided by the system. Water modulation is an optional feature.

Figure 9 shows a slam acceleration from idle onto the overspeed limiter and Figure 10 onto the temperature limiter.

The system is designed to respond correctly to a slam acceleration demand to a hot and decelerating engine. Figure 11 shows the response to this type of demand.

Figure 12 shows response to sharp throttle jabs and Figure 13 illustrates steady state governor performance. Figure 14 shows similar steady state conditions at altitude. Figure 15 shows an acceleration at altitude.

5.3 SAFETY AND MONITORING

The system provides automatic failure detection and response to failure. Where failure is detected the stepper motor drive is inhibited, the output is, therefore, frozen and the signal line to a changeover relay is activated. This relay sets in train the necessary actions to effect reversion to whichever system is next to be used.

Monitoring uses rate and range checks on all data. It also uses other methods. Outputs and actuators are monitored by position pick-offs. Some data is duplicated and comparison monitored. Simple models are used to monitor speed inputs and check sum procedures are used for memory contents. Dual microprocessors are used in the main lane and monitored by comparison. Either processor can detect failure and cause change-over to the back-up control.

Simulated failures have been demonstrated under many conditions. Figure 16 shows the effect of changing lanes repeatedly. The first change, during a slam acceleration, is a change from controlling with the main lane to controlling with the reversionary lane. When conditions have stabilised at the end of the acceleration the main lane is re-engaged. A third lane change, from main to reversionary is forced during a deceleration. There are some small parameter perturbations but they will not be discernible to a pilot flying the aircraft. Only a warning light will alert him to a failure.

Figure 17 shows an example of the system response to a second failure. In this situation the actuator is locked and fuel flow held constant. A red warning light is illuminated and the pilot engages the manual flow control. In Figure 17 the output is frozen under steady state conditions. The need for the warning light is shown very clearly by this recording. Figure 18 shows a failure causing the output to be frozen during a slam deceleration.

6.0 PHASE 2 (DEFINITIVE) SYSTEM

The demonstrator system for Pegasus uses 1½ lanes of electronics. The complexity of the reversionary lane depends on the amount of capability required to be retained after a failure in the main electronics control. In principle, the inputs to the partial lane can be reduced to two and the monitoring of the partial lane can be eliminated completely. Various increasing levels of complexity and sophistication are possible from this base depending upon the degree of capability required to be retained after the first failure. However, retention of mission capability after a first failure and simplified logistics led to the selection of a two lane system for Phase 2. The system is shown in Figure 19.

It uses the same basic methods and procedures as the Phase I system but is adapted to use two extended "main" lane controllers in place of the two dissimilar lanes. It embodies two additions to the Phase I system.

The first of these is the addition of a digital data interface for communication with other equipment in the aircraft. The second is the addition of an angle of attack input. This is used to reset the control to allow for the reduced surge margins experienced at high incidence.

7.0 FUTURE DEVELOPMENT

In the longer term, extensive development of the system can be envisaged to match the expected progress to VTOL aircraft design and operation. It is expected that supersonic versions of this type of aircraft will be designed and built. Such aircraft will probably embody plenum chamber burning (PCB). This corresponds to augmenter systems in conventional turbo-fan and turbo-jet engines and provides a significant increase in thrust. The addition of this feature to the engine and the supersonic capability to the aircraft will require extensions of the control capability in the future. The control requirements for PCB are similar to those used by DSIC in the control of PS50 and Adour engines *(7) and (8).

PCB control is essentially the same as a two gutter augmenter system. The dry engine system will require the additions shown in Figure 20. A variable nozzle is required and two metered fuel flows could be involved. The control laws used would be similar to those for normal augmenters with a pressure ratio control of nozzle area and non-dimensional schedules of fuel flow.

Changes required of the system will be the addition of one input variable and three output channels to each lane. Complete duplication will be mandatory if PCB is used in jet-borne flight. The control programme additions will result in only a small increase in the total memory size. All the dry engine control functions would be retained.

ACKNOWLEDGEMENTS

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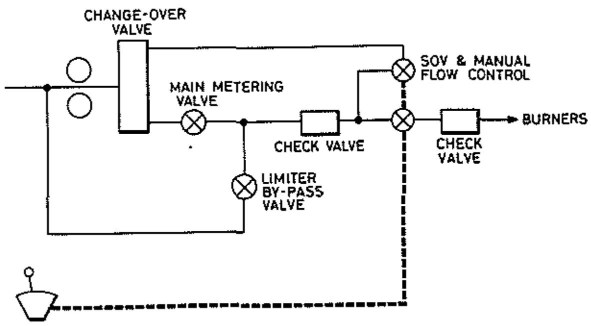


FIGURE 1

HYDROMECHANICAL CIRCUIT SCHEMATIC

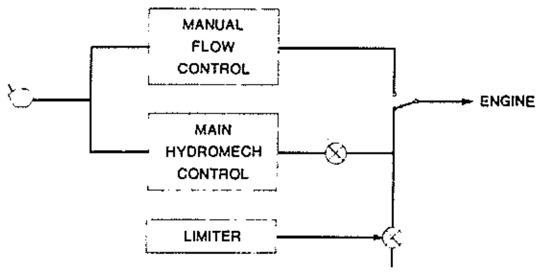


FIGURE 2

SYSTEM BLOCK DIAGRAM

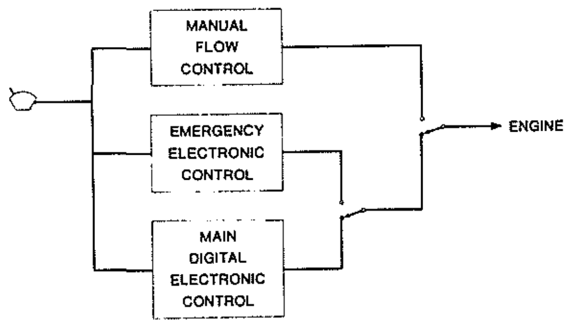


FIGURE 3

PHASE 1 DEMONSTRATOR SYSTEM

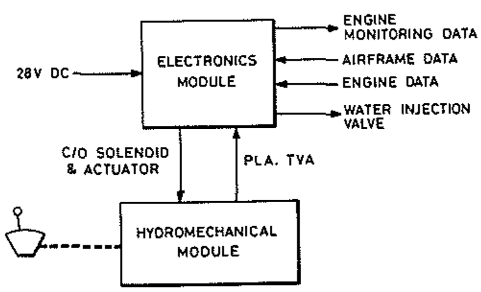
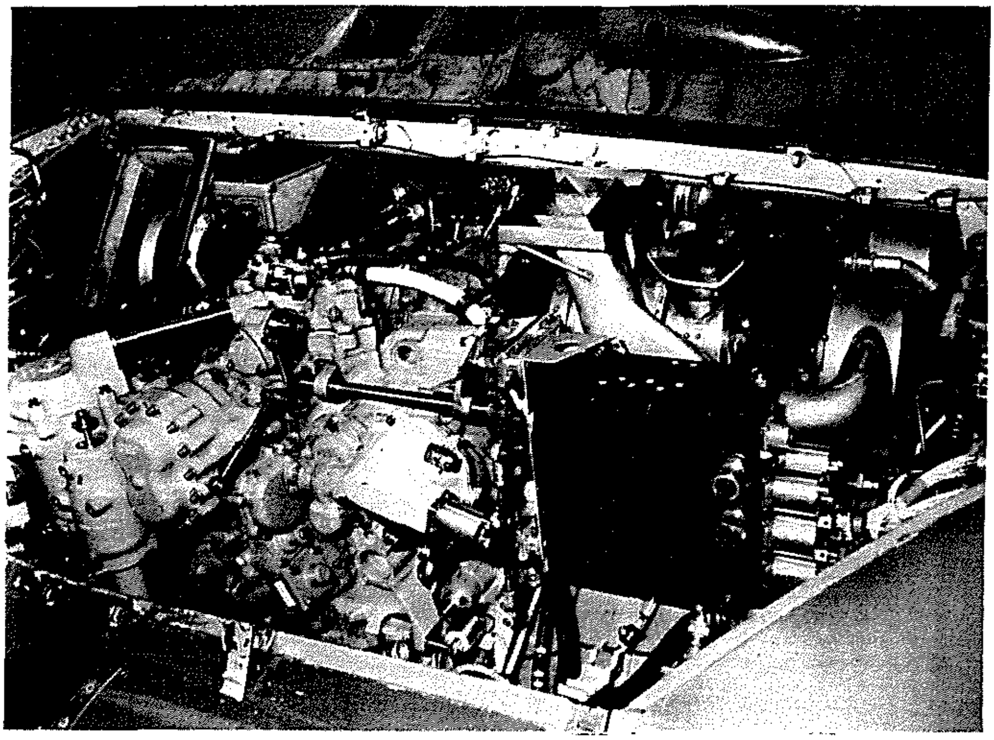
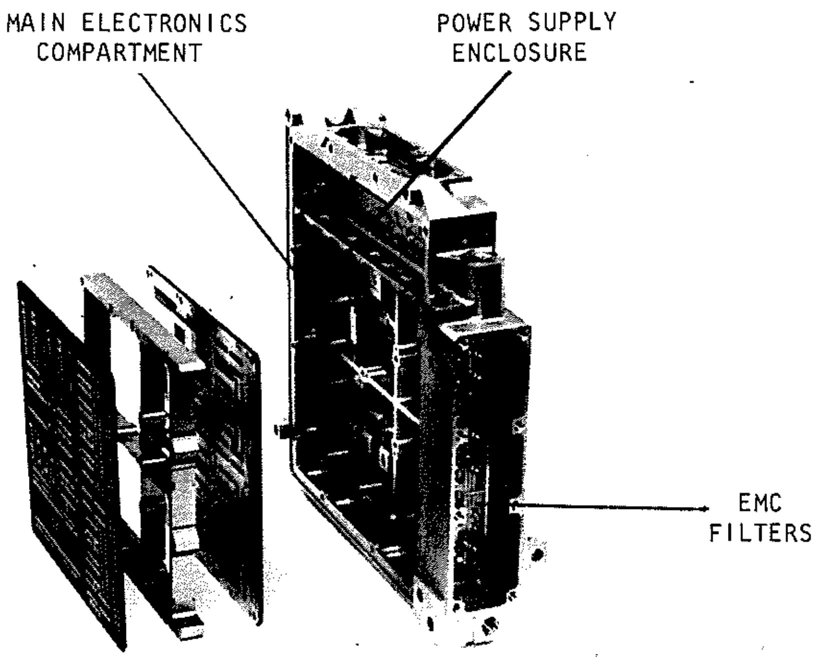


FIGURE 4

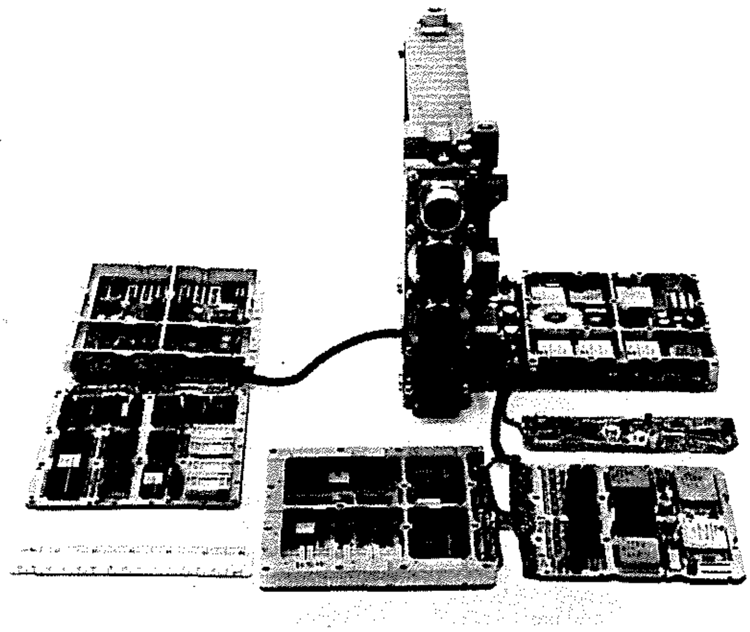
PEGASUS CONTROL SYSTEM COMPONENTS



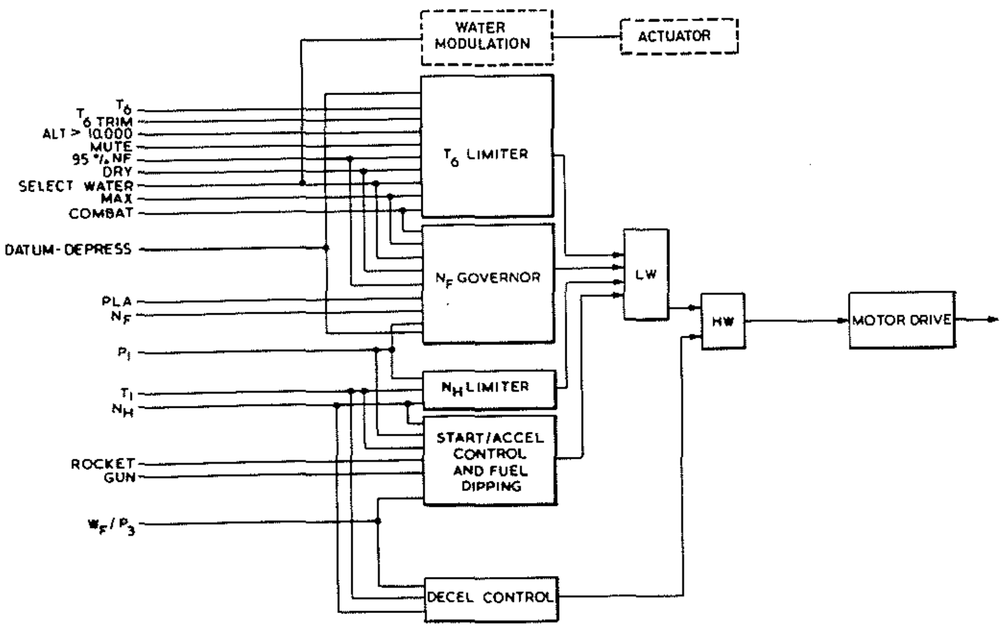
ENGINE MOUNTED ELECTRONICS MODULE FIGURE 5



REVERSONARY LANE CONSTRUCTION FIGURE 6



MAIN LANE INTERNAL VIEWS FIGURE 7



DIGITAL ELECTRONIC CONTROLLER SCHEMATIC FIGURE 8

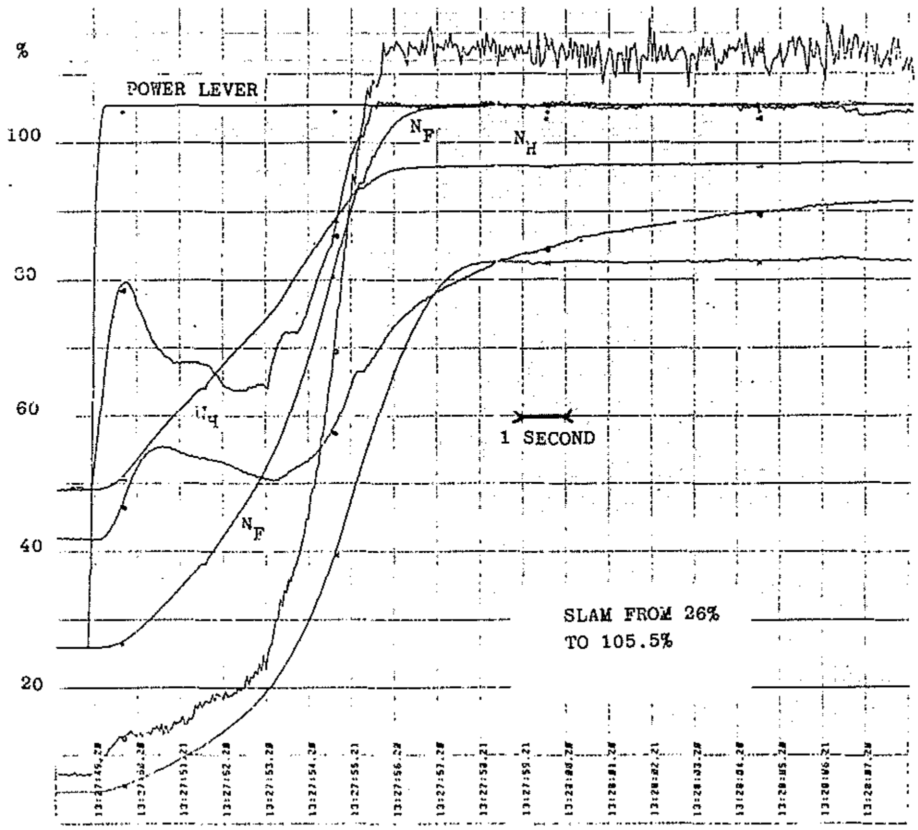


FIGURE 9

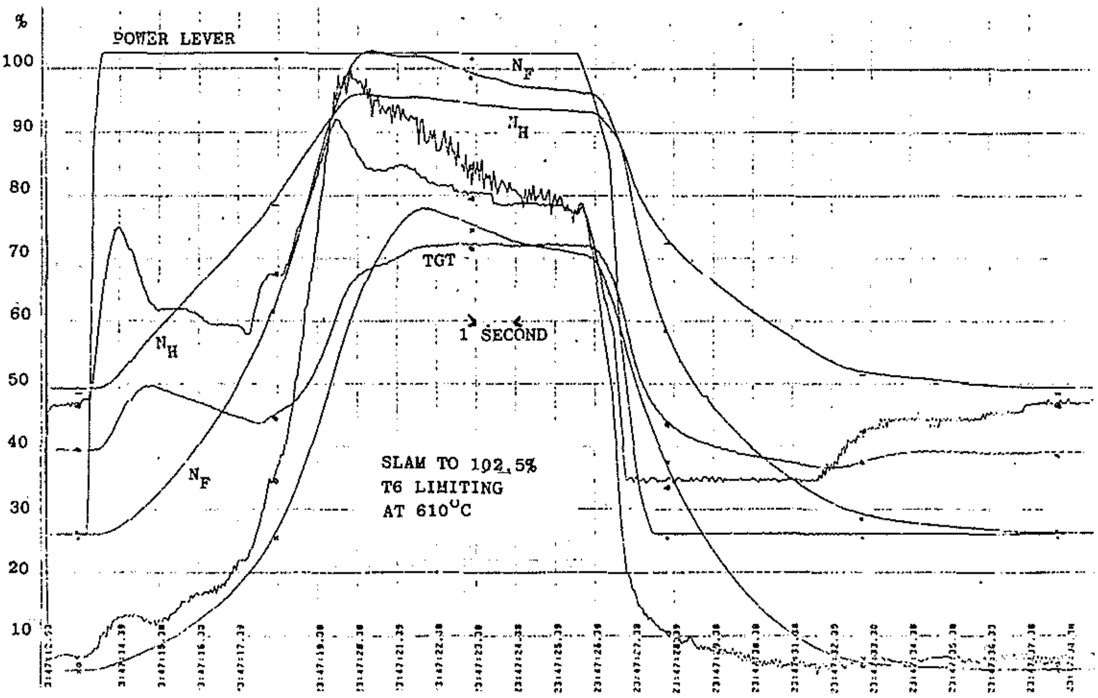


FIGURE 10

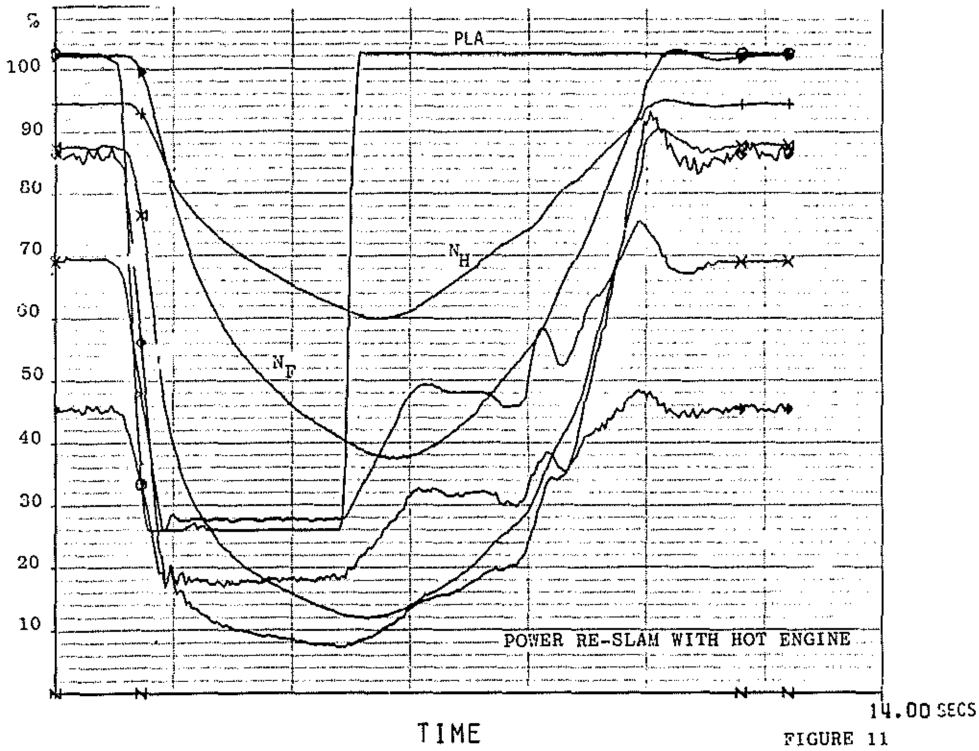


FIGURE 11

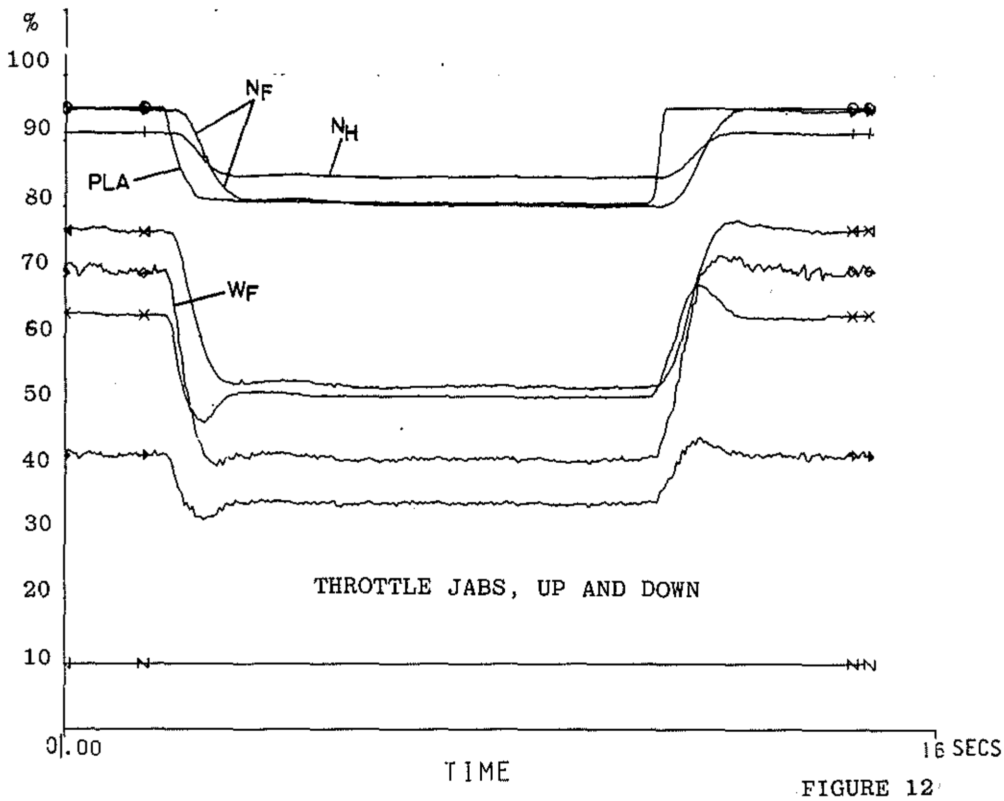


FIGURE 12

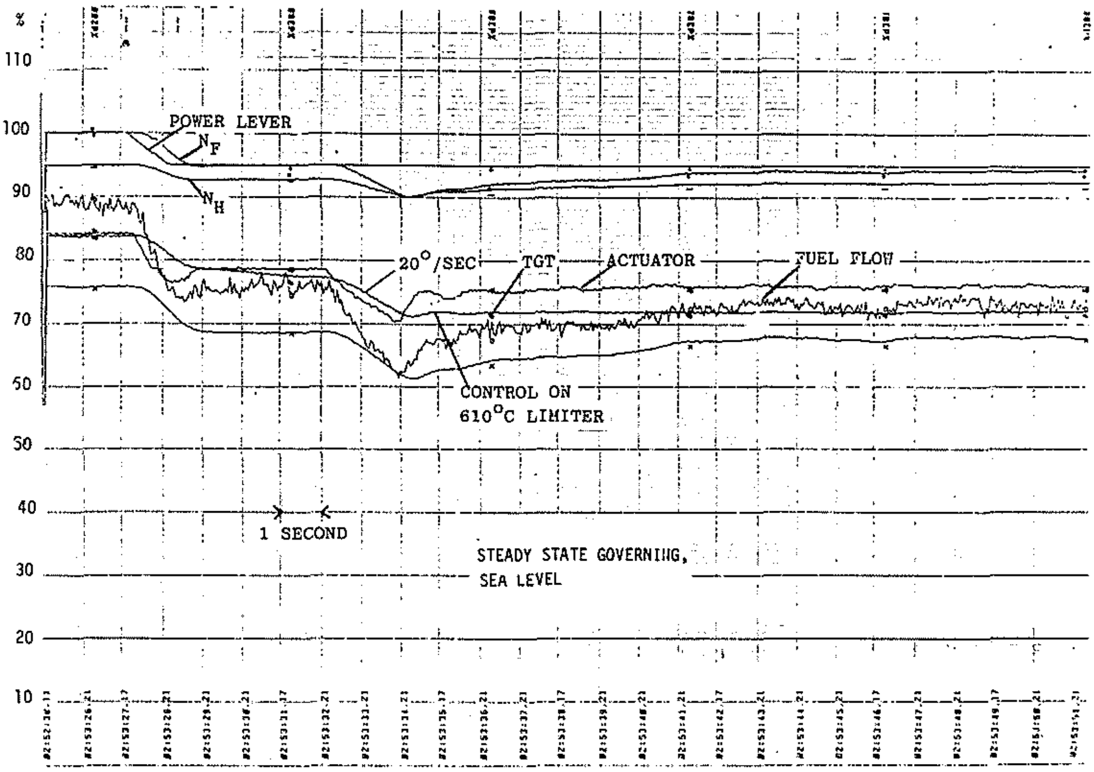


FIGURE 13

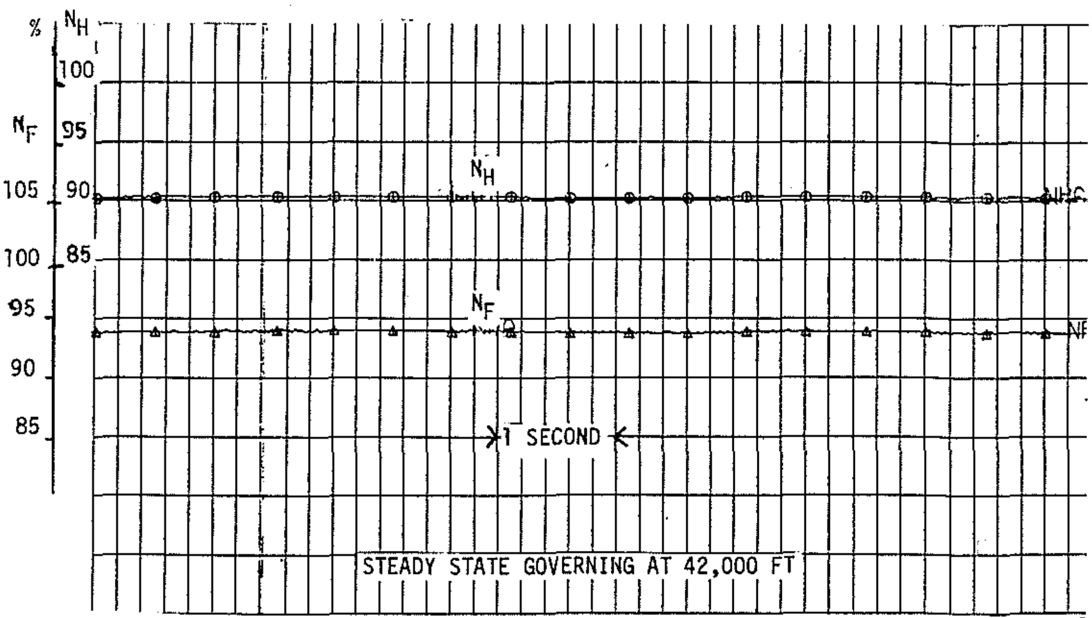


FIGURE 14

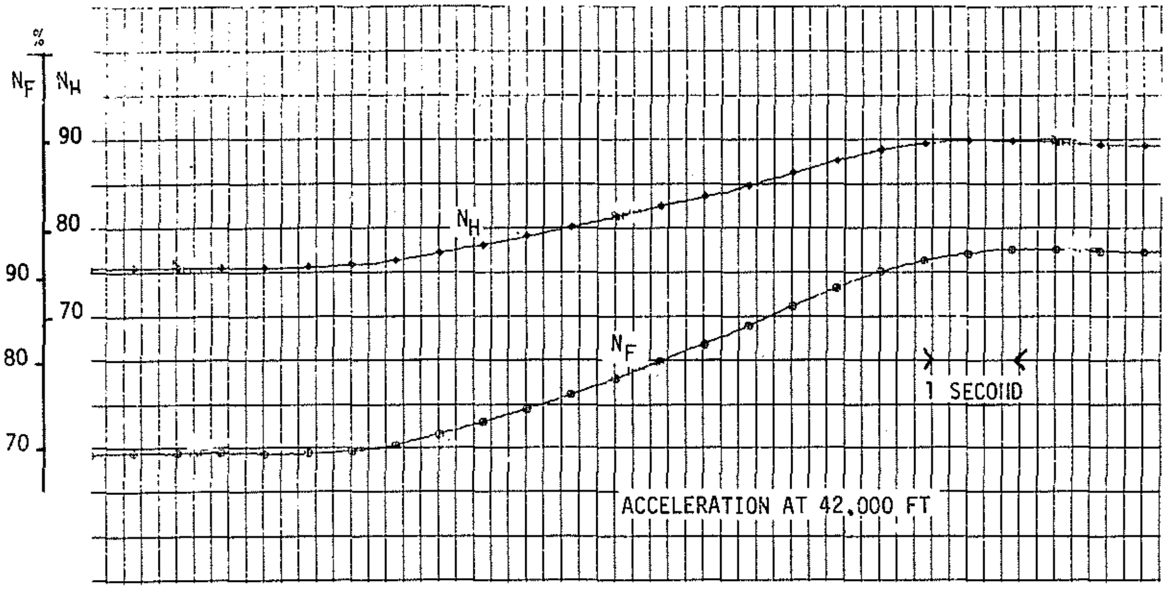


FIGURE 15

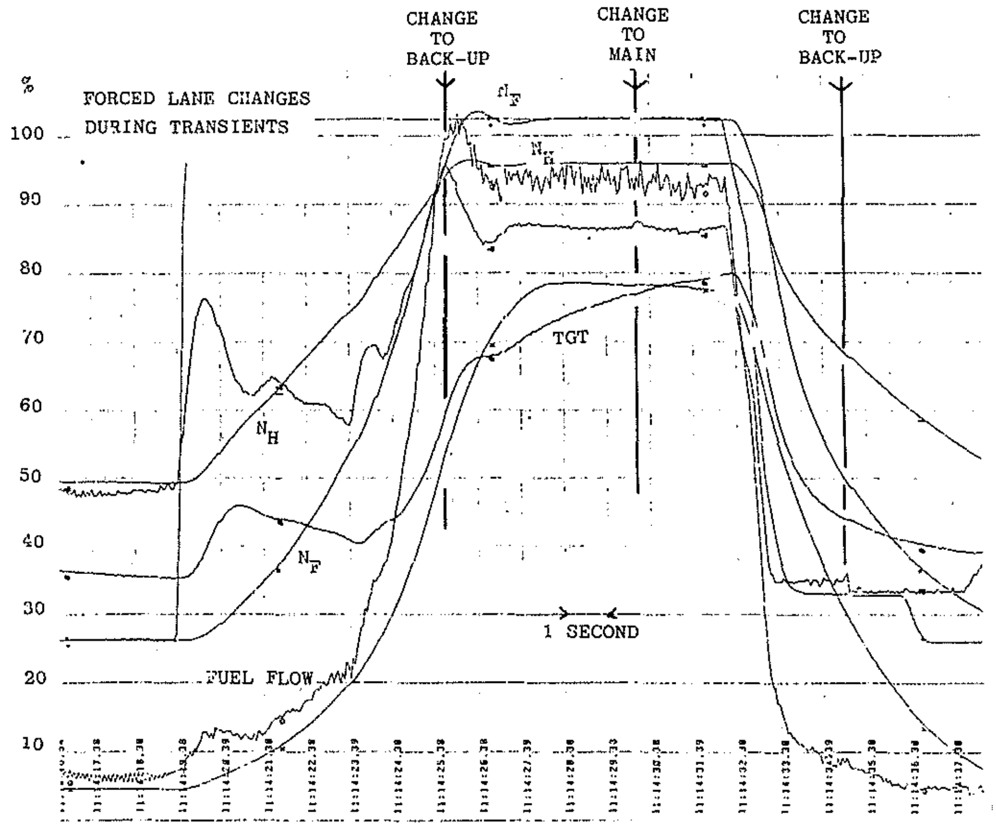


FIGURE 16

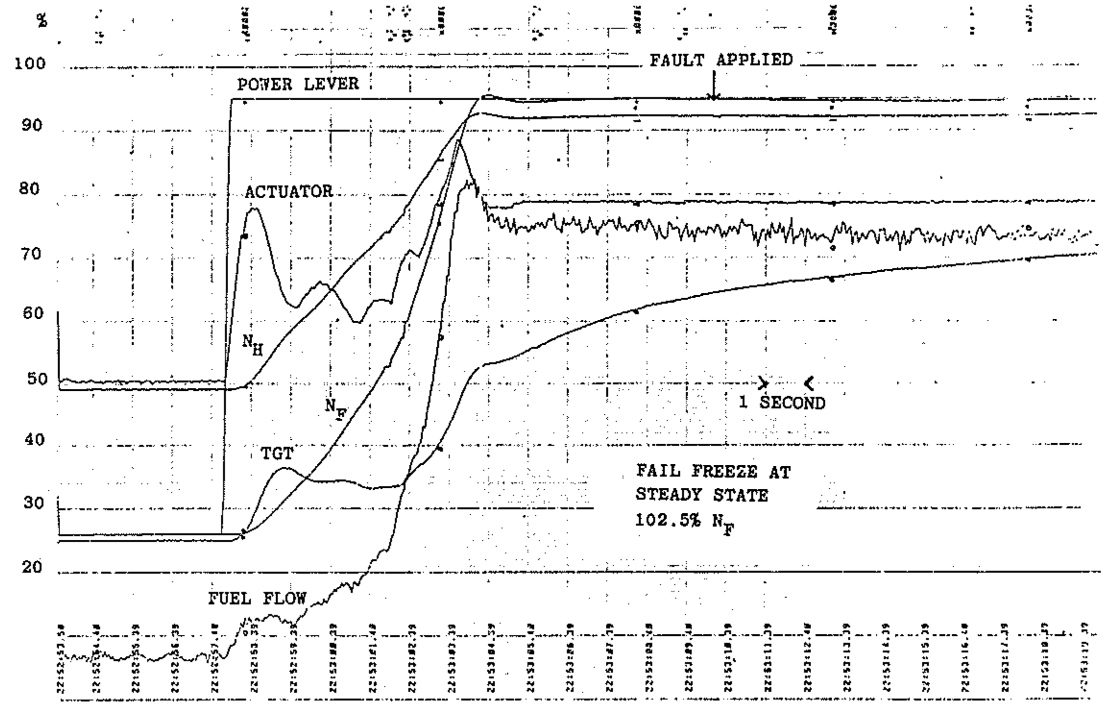


FIGURE 17

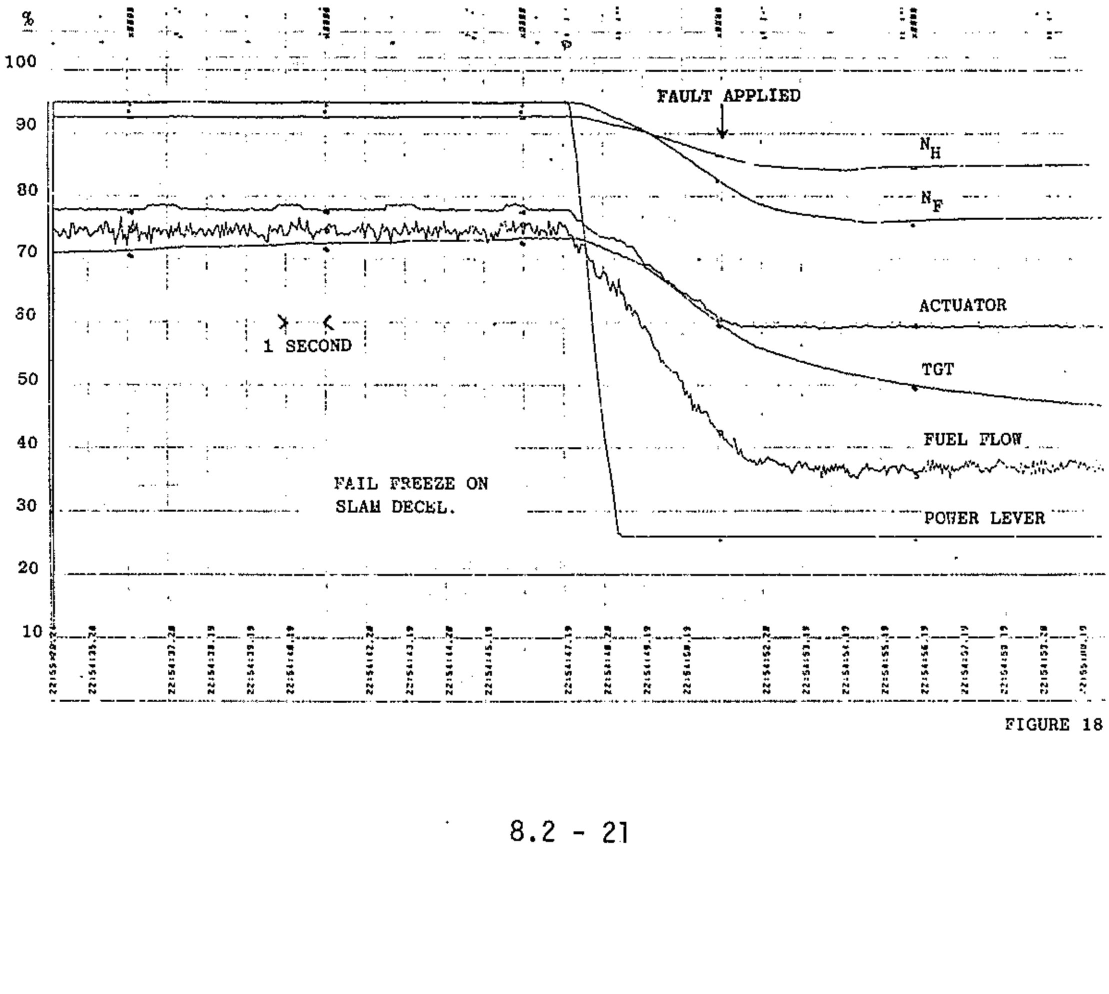
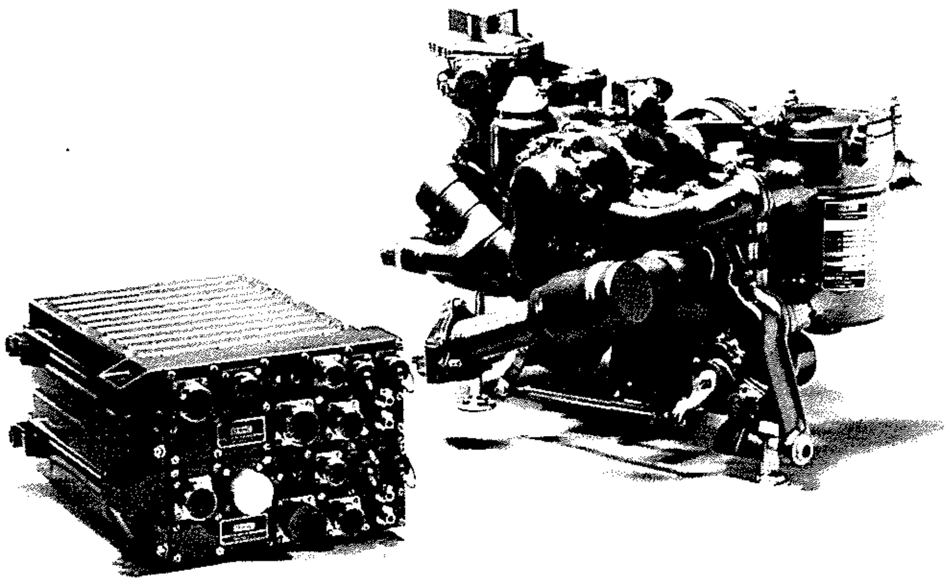
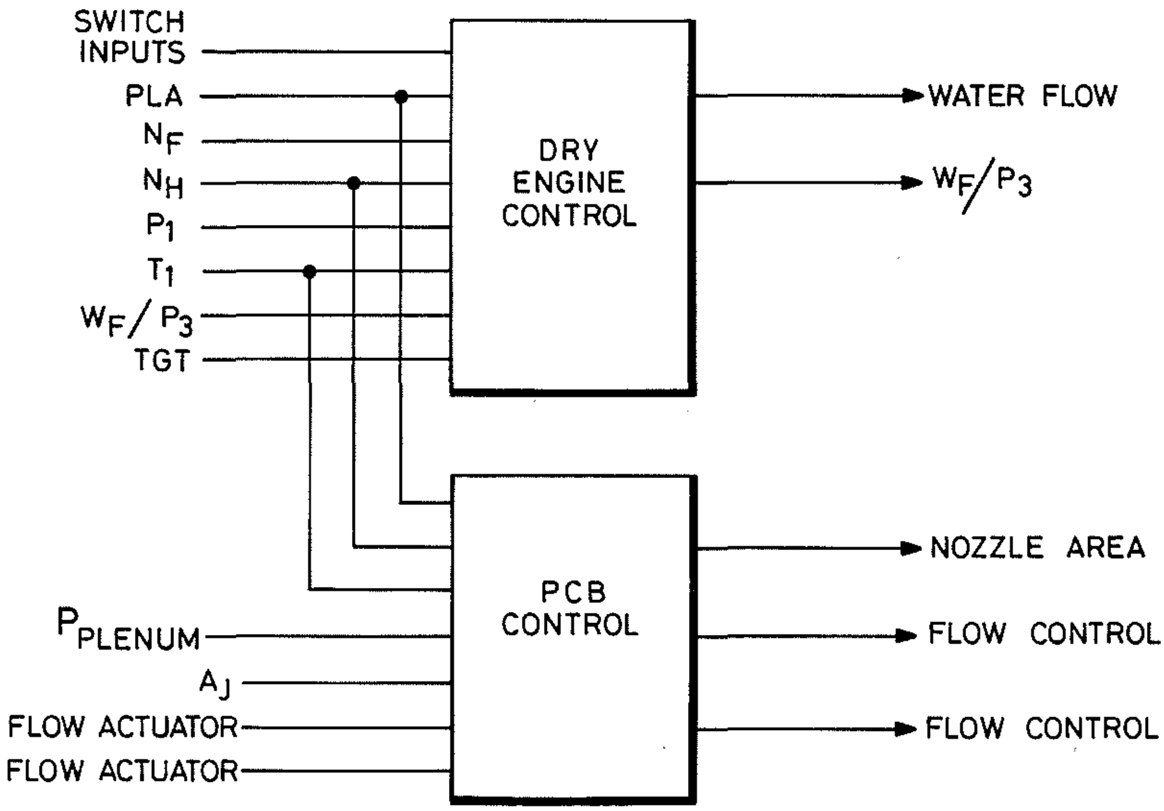


FIGURE 18



PHASE II SYSTEM FIGURE 19



PCB CONTROL SCHEMATIC FIGURE 20