Cranfield

College of Aeronautics Report No 8413
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Some Aspects of the Design of a "Fly-By-Wire" Flying Control System for a Supersonic V/STOL Fighter Aircraft

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ABSTRACT

The College of Aeronautics adopts a pragmatic approach to the teaching of aircraft design. Students will only be awarded an MSc degree if they have the ability to produce workable, realistic designs in which all of the major problems have been addressed. This ability is assessed by means of annual group projects in which relevant aircraft types are studied. Each student is given responsibility for the design of a major part of the aircraft, such as the subject of this report, the flying control system of a supersonic vertical take off fighter aircraft (Fig.1). This task involved the consideration of many design requirements, each of which influenced the system configuration.

The limited time available during the MSc programme meant that only rudimentary reliability calculations could be performed. It was decided to remedy this defect by a subsequent re-examination of the system using the following procedures:-

i) The derivation of safety, mission reliability and maintenance reliability targets.

ii) Prediction of the safety performance of the original system using empirical electronic and mechanical reliability data, augmented by MIL-217 and other generic reliability data.

iii) Progressive modification of the system until the stringent targets had been met.

These calculations showed that the original design, although using good design practices, fell considerably below the safety required for completely electronically signalled system. A large number of additional components was required to give the redundancy required.

Subsequent work concentrated on the investigation of the rudder pedal control systems in the cockpit with detailed design of the preferred scheme.

The final phase of work concerned the design of the roll reaction control nozzle.

This process showed the vital importance of adequate reliability calculations early in the design process. It also gave a good insight into how the system really works and how it might fail. Knowledge of these aspects of design is vital in the training of any designer of high-integrity systems and its importance cannot be overstressed.

The rather arbitrary safety target of two simultaneous failures in 10^7 hours leads to very costly systems and should be investigated more fully. The results of current FBW research should be incorporated into new requirements and designs.

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1. INTRODUCTION

1.1 The Reason for the Current Study

The College of Aeronautics adopts a pragmatic approach to the teaching of aircraft design. Students will only be awarded an MSc degree if they have proved that they have the ability to produce workable, realistic designs in which all of the major problems have been addressed. This ability is assessed by means of annual group projects in which relevant aircraft types are studied. Each student is given responsibility for the design of a major part of the aircraft such as the subject of this report, the flying control system of the S-80 supersonic vertical take off fighter aircraft (Fig.1). This task involved the consideration of many design requirements, each of which influenced the system configuration.

The limited time available during the original programme meant that only rudimentary reliability calculations and design calculations could be performed.

It was deceided that this work could profitably be continued and that the system's re-examination and design would form a suitable basis for a two-year study by the co-author, Mr. Meng. This project was thought to provide a good opportunity to examine an extremely modern, complex and powerful aircraft system, making extensive use of various reliability prediction techniques. It was also decided to perform the detailed design of a number of mechanical components of the system.

1.2 Description of the S-80 Project

It is necessary to describe the aircraft so that the current study will be seen in the correct context. The salient points of the description taken from Ref.1, are:-

1.2.1 Specification

The aircraft was designed to meet the U.S. Navy requirements for the Type B supersonic aircraft (1977):-

- Naval operations
- High performance V/STOL fighter/attack aircraft
- Supersonic dash capability with sustained Mach number capability of at least 1.6
- Operational from land and from ships smaller than CV's without catapults and arresting gear - good short take-off capability
- Sustained load factor of 6.2 at Mach 0.6, 10,000 ft altitude at 88% VTOL gross weight
- Specific excess power at 1 g (Ps 1 g) of 274 m/s (900 ft/s) at Mach 0.9, 10,000 ft altitude at 88% VTOL gross weights
- VTOL gross weights = 9072 to 15876 kg (20,000 to 35,000 lb)
- STO sea-based gross weight = VTOL gross weight plus 4536 kg (10,000 lb).

1.2.1 Configuration

Figures 1 and 2 show general arrangements of the aircraft.

The overall dimensions were chosen so that the aircraft could use the 55 ft x 32 ft flight deck lifts of the Invincible-class ships without folding wings.

The College of Aeronautics has been engaged in research into the construction of carbon and glass-fibre composite structures for the past 15 years. This included such areas as:-

i) Joints
ii) Material properties
iii) Buckling and post-buckled behaviour
iv) Impact and battle damage repair
v) Fatigue
vi) Design, construction and flight test of remotely-piloted vehicles with composite primary structures.

This experience prompted the decision to design the fighter to utilise carbon fibre reinforced plastic construction wherever possible.

Conservative estimates of weight savings were:-

Wing type components	=	15%
Fuselage	=	16%
Undercarriage	=	10%

Detailed descriptions of the aircraft and components are contained in individual students' project theses (Ref.2), but brief descriptions follow:-

Layout

A close-coupled canard arrangement was chosen because it gave improved lift characteristics for short take-off and combat. The foreplane enables the achievement of high angles of attack and improved area-ruling for reduced supersonic drag as well as providing active control to overcome the relaxed longitudinal stability of the aircraft.

The engine intakes are on the top of the rear fuselage. This reduces the radar cross-section, gives a better area distribution and reduces the inlet duct volume. This releases additional critical fuselage volume for fuel tanks and the remote augmentor duct.

Some doubts have been expressed about the efficiency of such intakes, but wind-tunnel tests of a generally similar configuration showed promising results. The under-fuselage pod is used to house the gun, the submerged part of the Skyflash missiles and the main undercarriage. The semi-submerged missiles reduce drag relative to wing pylons and the pod improves the area distribution.

Twin fins are a feature of some recent military aircraft and their feasibility for this study was investigated. On a conventional single fin design, directional stability reduces at high incidence because vortices, originating from separation on the forebody, both increase the destabilising effect of the body and destabilise the part of the fin. Twin fins can be located to either avoid these adverse interferences effects or obtain favourable interference. The twin fin arrangement also reduces the height of the aircraft which is particularly advantageous for naval aircraft.

Wing

The wing is a highly swept clipped delta with an aspect ratio of 2.2 and thickness/chord of 5%.

A small percentage of fuel is carried in the wing, which is also used to mount the main undercarriage, the reaction control system for roll control in the hover, and Sidewinder missiles.

Fuselage

The forward fuselage contains a partially reclined ejector seat surmounted by a large canopy with good pilot visibility.

Figure 3 shows the mock-up that was constructed to check pilot reaction to this cockpit layout.

The forward fuselage bending loads are taken by four longerons which are positioned such that the top two longerons form the cockpit coaming, while the bottom longerons are a continuation of the two longerons of the centre fuselage at floor level.

The carbon-fibre skin was designed to be effective in shear and torsion only, as the presence of large cut out areas had dictated the use of longerons to take the compressive/tensile loads.

The four-longeron construction was continued into the rear fuselage where they are used to mount the speed brake. Frames are used to mount the engines and fins. Removal of the engines is accomplished by pulling them aft along rails.

Powerplant and Vertical Take-Off System (Fig.4)

The aircraft uses the remote augmented lift system (RALS) being developed by the General Eelectric Company. Two propulsion units are used, mounted side by side in the rear fuselage, having variable cycle capability and a double bypass split fan to provide airflow to the single remote augmentor nozzle during vertical take-off and landing. Primary exhaust is through ADEN nozzles (Augmentor Deflector Exhaust Nozzle).

For vertical take-off the aircraft rises on three columns of high-energy air, one from the RALS nozzle and the other two from the deflected ADEN nozzles. For forward flight the RALS is shut off and the ADEN nozzles point aft. The RALS nozzle is gimballed to provide tilting to produce pitch and yaw control in the hover.

Roll control in the hover is achieved by the use of a reaction control system which bleeds high pressure compressed air and ducts it to wing tip mounted convergent-divergent nozzles. These ducts and nozzles have maximum operating temperatures of 480°C and are manufactured from nickel alloys.

These components posed great installation problems because of limited space in the very thin wing section. This section also initially produced aeroelastic problems from the wing-tip missile installation. These problems are resolved by a local increase in chord and careful siting of the nozzles.

The Flying Control System (original design)

The flying control system uses four channel electrical signalling to hydraulic actuators at the control surfaces. The aircraft is deliberately unstable in the pitch sense, control being achieved by the actively-controlled all-moving canard foreplane. The two halves of the foreplane are rigidly connected by a titanium torque tube. The foreplanes use a large centre spar allied to high modulus carbon fibre reinforced plastic skins supported by full-depth honeycomb core. Lateral stability is aided by twin fins. These have five carbon fibre spars each and use high-modulus carbon-fibre skins protected with an outer layer of KEVLAR. The twin rudders are constructed from glass-fibre reinforced plastic supported by honeycomb core. The design allows the wing trailing edge control to work as normal sealed high-speed ailerons in cruise flight, as drooped ailerons in high-speed combat and as slotted flaps with differential aileron motion to conventional and vertical take off and landing. Two control surfaces are included on each semi-span. The surfaces can be operated independently, or together.

These surfaces are constructed of carbon-epoxy laminates with a full depth honeycomb core. Actuation of the aileron motion is by a large hydraulic actuator on each control surface. Actuation of the flap is by three ball-screw linear actuators per control, powered by electrical motors in the fuselage.

Hydraulic System

The hydraulic power source comprises two completely independent systems, each with its own reservoir, pump, accumulators and piping with a stand-by system.

The two main hydraulic system pumps are driven from airframe mounted gearboxes. The stand-by system pump is electrically driven by an AC motor.

Both main systems share the operation of the flying control actuators. Each system supplies the actuator with 50% of the power required. In case of one system failure, the remaining system will supply the actuator (through a change-over valve) with the power required.

The stand-by system supplies most of the surfaces in case of both systems failure. It is switched on automatically by a duplex system pressure drop.

The system pressure is 4000 psi (27.58 MN/m^2).

Fuel System

The total internal fuel capacity of the aircraft is 4700 kgs, the tanks being located in the fuselage and innerwing. Two 100 gallon drop tanks are carried in the STOL mission under the outer wings.

The fuel system uses both integral and self sealing type tank construction. Collector tank transfer and feed system is incorporated, each engine being separately fed by a collector with a provision for cross-feed in case of emergencies. The transfer of fuel from the main tanks to the collectors is by tank mounted booster pumps. Transfer from the drop tanks is by means of compressor bleed air.

The centre of gravity limitations of the aircraft are met by suitable selection of tank locations and transfer sequence and no recourse is made to an automatic centre of gravity control system.

As far as possible,measures have been taken to use standard fuel components. Vulnerability to enemy action, reliability and accessibility have been given due consideration in the design.

Undercarriage

It was decided to make the undercarriage suitable for use from semi-prepared airstrips. This made the wheels rather large, but it was felt that it was necessary to give the aircraft the ability to operate from beach heads in the STOL mode.

The undercarriage configuration is of standard tricycle form using an aft-retracting nosewheel with twin wheels. The main legs retract inboard into the fuselage fairing and mount single wheels.

Armament

a) VTOL Mode

1 Vulcan 20 mm rotary cannon - M61 A1
2 Short-range Sidewinder missiles on wing-tips
2 Medium-range Skyflash missiles under fuselage.

b) STOL Mode

In this role the Skyflash missiles are replaced by four Martel or Sea Eagle anti-ship missiles and range is augmented by external fuel tanks.

1.3 The Project Work Programme

A major motive for the project was the improvement of background aeronautical knowledge. This was accomplished by attendance at numerous lecture courses, and tutorials, visits to operators and manufacturers of modern aircraft, and extensive literature surveys. The work associated directly with the project was envisaged as:-

i) Familiarisation with the existing S-80 design
ii) Investigation of the flying control system configuration
iii) Investigation of the cockpit controls
iv) Detail design of some mechanical components of the system
v) Compilation of a report to summarise the work.

2. FAMILIARISATION WITH THE S-80 AIRCRAFT

The initial part of this work was the careful reading of the relevant theses of Ref.2. This was followed by the production of the following engineering drawings to summarise the information on the original design:-

Drawing No.1 (see end of this report)

The drawing shows the position of all the main structural members of the aircraft, together with the access panels that had been defined.

Drawing No.2

This drawing shows the position of the ducting, nozzles and engines for the remote augmented lift system (R.A.L.S.) and the reaction control system (R.C.S.).

Drawing No.3

The positions of the flying-control system actuators are shown on this drawing.

It can be seen from the above drawings that there is a complex interaction between all the components and it was therefore decided to construct a scale,three-dimensional model to help the visualisation of the interactions. To assist this process, Drawing No.6 was produced, which shows the fuselage cross-sectional shape at a number of positions down the fuselage.

The model was produced and the various components incorporated as shown in Fig.5.

3. INVESTIGATION OF THE FLYING CONTROL SYSTEM CONFIGURATION

3.1 Introduction to the System

The aircraft is expected to take off vertically and to fly at high speeds up to M 1.6, thus its flight control system is very complicated.

In the conventional flight mode, pitch control is performed by a moving canard, roll control is by flaperon and yaw control by twin rudders, (Fig.1).

In VTOL flight mode, all three control systems make use of jet-exhaust reaction (Figs.4 & 6). Pitch control is completed by lift variation of a remote augmentor lift system (R.A.L.S.) at the front and two augmentor deflector exhaust nozzles (A.D.E.N.) at the back, roll control by lift variation of reaction control system (RCS) nozzles in wing tips, and yaw control by lateral tilt of a front RALS nozzle only.

Summarising:-

CHANNELS / MODES	PITCH	ROLL	YAW
Conventional	All-moving canard	Flaperons	Twin rudders
VTOL	Lift variation of RALS and ADEN	Lift variation of RCS	Lateral tilt of RALS only

The pilot of the S-80 will be subjected to high acceleration levels, but this has been alleviated by the use of a highly raked seat. It was decided to use a side-stick controller for the pitch and roll axes because this makes control easier under high 'g' conditions. Conventional rudder pedals were chosen for the yaw axis.

Having defined both ends of the control system, i.e. the surface or nozzles and the cockpit controls, it is now necessary to describe the middle portion of the system.

Paragraph 1.2.2 states that the aircraft utilises active control in the pitch sense. This requirement, together with the need for fully-powered controls for such a high performance aircraft, drove the design in the direction of electrical signalling (fly-by-wire). This was chosen because conventional cable or rod signalling would not react quickly enough to compensate for the inherent instability of the aircraft. Fibre optic signalling (fly-by-light) was not chosen at this stage because there was insignificant information available concerning its reliability. This should be re-examined when more information becomes available.

The electrical signalling has additional benefits in terms of weight, simplified installation, and control, to ensure that the pilot works to the limits of manoeuvrability without over-stressing the aircraft.

The use of electrical signalling necessitates the use of large numbers of electronic components, each with a specific probability of failure. This also applies to the rest of the system, indeed of the whole aircraft. The major task of the system designer is to select the right number of the right types of components to ensure that performance targets can be met with a given level of safety and reliability. Subsequent paragraphs describe this process.

3.2 Safety and Reliability Targets

3.2.1 Aircraft Safety Targets

When the project was started there was little available information about the safety levels to use. Reference 3 quotes loss rates for F-4 Phantom aircraft as 5.8×10^{-6} per flight how due to flight control system failures. Reference 4 requires civil transport aircraft catastrophic failures to occur from 1×10^{-7} to 1×10^{-9} per flight, this includes all systems. The initial target for this study, was taken to be between these values and was chosen to be 1×10^{-7} per flight, attributable to the flight control system only.

The system was designed to meet this target, but the requirements shown in Ref.5 gave a more exact definition of the target for an actively-controlled fly-by-wire system. This was:- "It is designed to achieve a one-in-ten million failure probability over a one hour flight, surviving any two electrical failures or a combination of a hydraulic and an electrical failure in any order".

The system was therefore re-designed to meet this exceedingly severe requirement.

3.2.2 Mission Reliability Targets

This is the probability that the aircraft will be able to perform a given mission without any failures or defects that will have an operational effect. A modern combat aircraft has been recently designed to have an operational reliability of 0.95 and this target was used for the S-80. The usual method of recording operational defects is to monitor the whole aircraft fleet over a number of months and produce an operational defect rate. This is defined as:-

$$\frac{\text{No. of operational defects for the fleet in the period}}{\text{Fleet flying hours x 1000}}$$

Operational defects cover such things as accidents, mission cancelled, mission partial failure, flight safety hazard, etc.

Thus the target mission reliability is 50 operational defects per 1000 hours. Reference 9 quoted a target for the flying control system of 4.8% of this value, based on comparable aircraft, i.e. 2.4 op. defects per 1000 hours.

3.2.3 Aircraft Confirmed Defect Rate Target

The RAF has a comprehensive reliability data recording system with a central computer at the Maintenance Data Centre (MDC) at Swanton Morley. Records are maintained of all defects reported on RAF and RN aircraft. Not all defects are confirmed as genuine defects because some of the reports may have been due to poor faul diagnosis. The confirmed defect rate includes operational defects, above. Reference 9 called for a target of 24 confirmed defects per 1000 flying hours for the flying control system. This is a good measure of the maintenance effort required to keep the aircraft flying.

3.3 Design to Meet the Safety Target

3.3.1 Initial Process Using a 1 in 10^{7} Failures Per Flight Hour Target

3.3.1.1 General

It was decided to distribute the target between all channels equally in the preliminary calculations.

In our case, there are three control axes: pitch, roll and yaw, and each axis contains controls for two flight modes, conventional and VTOL respectively. So we have six control channels in all. Therefore the safety target for each control channel equals $1 \times 10^{-7} \div 3 \div 2 = 1.67 \times 10^{-8}$, the number of catastrophic failures for each channel should not be allowed to exceed this value. As we can see this safety requirement is very severe and very difficult to meet. It was this safety target that posed a decisive effect on multiplication and configuration of control system channels.

On the basis of reliability theory, when calculating system reliability, the whole system was divided into several sections each being considered as series or parallel sections. The total system reliability was determined by analysing the reliability of each component and checking the effect of this value on the next highest sub-assembly, assembly, sub-system until the reliability of the entire system was determined. (see Fig.7)

In the safety analysis, the sub-systems were divided into two sections, the inside and outside the cockpit sections. This is because the environmental conditions inside and outside the cockpit are very different. There are no environmental extremes of pressure, temperature, shock and vibration in the cockpit but the environments outside are much more severe. We know from reliability theory that environmental stresses exert a tremendous effect on the failure rate of components especially for electric-electronic components.

Another important factor in reliability analysis is mission time. Of the two flight missions of the aircraft, the intercept mission takes about 1.5 hours and the sea patrol last up to 2.6 hours. So 2.6 hours was chosen as flying time for the conventional control system channels.

It was assumed that the time taken for vertical take-off and landing cycles would be 5 minutes per flight.

The failure probability distributions of different components will differ considerably. Most electronic components follow a negative exponential distribution, but this does not apply to wear-out of mechanical components. The mechanical components, however, are parts of relatively complex assemblies, such as actuators, pumps etc. which usually exhibit a negative exponential distribution when considered as assemblies (Ref.6). This study, therefore, assumes negative exponential distributions throughout and it is thought that this will be a good approximation to reality. The failure rates used for individual components are shown in Table 1.

3.3.1.2 Examination of the Roll Control Circuit

a) Inside Cockpit

All components, except a feel mechanism, are quadruple for both conventional and VTOL roll controls. The reliability of quadrupled sections are very close to unity, and therefore the failure rate of the feel unit alone determines the reliability of inside cockpit section Rin, which turns out to be 0.9999999948 (Appendix 1).

b) Outside Cockpit

i) Conventional

The aircraft has four pieces of flaperons with two pieces each wing, but only the outboard flaperons serve as ailerons, therefore the inboard flaperons have nothing to do with roll control during normal operations but were necessary for stand-by conditions. The schematic diagram of flaperon transmission is shown in Fig.8.

When the hydraulic torque motors rotate, ball-screw actuators will carry left and right flaps deflecting around point B in phase, this makes up flap control. When the hydraulic aileron actuators move, they will drive trailing edge vanes plus flaps together (flaperon) rotating around point A, the port and starboard flaperons deflect in opposite directions playing the role of roll control.

Although the hydraulic motor channel does not actuate the aileron, it fills the role of supporting point C. If this function of supporting fails, the roll control also fails, therefore this channel reliability must be considered.

Initial calculations showed that the derived failure rate of this layout was far worse than its safety target, because of the very high failure rate of several of the mechanical components.

Extensive calculations were performed with various components configurations until the configuration shown in Fig.13 was arrived at. This produced a predicted failure rate of 1.48 per 10^8 flights.

ii) VTOL Roll Control

In this flight mode, the roll control system nozzles are controlled by the movement of outboard flaperons through rod-bellcrank mechanisms. In calculating VTOL reliability we must add this mechanical system to the conventional roll control system in series.

It could not initially meet the target and it was therefore decided to duplicate the roll control nozzles.

This was achieved by using one duct per wing, but making use of two nozzles at the end of each, one pointing up, and one down. In the event of failure of the ducting/nozzle system in one wing, the other duct system could produce roll control in either direction. This is now a fully redundant system and its reliability improves considerably and meets the target, (see Appendix 1).

Subsequent study showed that this system was adopted on the roll control system of the Harrier (Ref.8)

3.3.1.3 Examination of the pitch Control Circuit

a) Inside Cockpit

We have the same configuration as roll control inside cockpit for both conventional and VTOL.

b) Outside Cockpit

i) Conventional

In the configuration of Fig.9, only using the canard, we could not meet the safety target, because the failure rate of the bearings is high. This rate was taken from Ref.7 and its wear-out failure rate accounts for 73%. Because the 73% wear-out failure can be removed by scheduled replacement, the failure rate used equals:

$2.718 \times 10^{-6} \times 27\% = 0.734 \times 10^{-6}$.

We can also resort to the inboard flaperons as a stand-by branch. In addition to the flap function, we would rotate it around point A in phase, performing the pitch control function. (Fig.8).

At the very moment of failure of the canard, we would actuate this channel and it therefore serves as a stand-by system.

The stand-by system of inboard flaperon has a reliability of 0.99999998, the port and starboard flaperon was considered as series system. Applying the stand-by formula of reliability, we obtain the result equal 0.5×10^{-8}, which will meet the safety target.

ii) VTOL

The layout of Fig.10 was investigated and its reliability turned out to be very poor.

After this, we reduced the failure rate of the nozzle to 5×10^{-6} taken from B 747 (Ref.16) and took flying time as 5 minutes. The target could still not be met

There were two alternative solutions to use.

Firstly, we could put a much stricter failure rate for the nozzle to the manufacturer, which would inevitably involve high cost and long development time.

Secondly, we could use two extra nozzles, one each in the nose and the tail of the aircraft with their own special ducts bleeding gas from the engines. The two nozzles should be considered in series, and the whole system as a stand-by system to VTOL pitch control.

Assuming they posses the same reliability as that of RALS and ADEN nozzles and applying the stand-by formula, we found that we could meet the target of VTOL control, but this scheme would produce several severe difficulties, such as arrangement and installation problems, maintainability problems and engine lift decay problems due to additional bleeding, etc.

3.3.1.4 Examination of the Yaw Control Circuit

a) Inside Cockpit

Again we have the same arrangement as the roll control inside cockpit for both conventional and VTOL yaw control. The reliability of this section equals to 0.9999999948.

b) Outside Cockpit

i) Conventional

The mechanical transmission sections of each rudder were duplicated and the two rudders taken as parallel. After calculation, its failure rate was 0.6×10^{-8} which was less than the target, thus meeting the safety requirement.

ii) VTOL

In the VTOL flight mode, yaw control is provided by lift tilt of RALS nozzle only. This was found to be far too unreliable. The nozzle control section was quadruplicated, and the chain drive duplicated, but the target could not be met, because of the nozzle failure rate. The only solution was to duplicate the nozzle by running an additional yaw duct from the engines, in a similar manner to the pitch circuit. Using the stand-by formula the failure rate was calculated to be 0.2×10^{-8}, thus meeting the target, but at the expense of considerable complexity and cost.

3.3.1.5 Summary of Results of the Initial Safety Analysis

The following table highlights the results of the above analyses:-

Flight Modes	Channels	Targets	Failure Rate	Methods
Conventional	roll	1.67×10^{-8}	2.48×10^{-8}	Duplication of hydraulic motor.
	pitch	"	0.5×10^{-8}	Inboard flaperon stand-by.
	yaw	"	0.6×10^{-6}	Two rudder as parallel system.
VTOL	roll	1.67×10^{-8}	0.2×10^{-8}	Double nozzles each wing tip.
	pitch	"	0.02×10^{-8}	Emergency nozzles in the nose and the tail.
	yaw	"	0.2×10^{-8}	Nose emergency nozzle

It can be seen from the above table that all but one of the sub-systems meets its individual target and that the whole system performance of 4×10^{-8} exceeds the target of 1×10^{-7} flights. The system changes required to do this have been extensive in terms of the requirement of extra hydraulic channels and more severely, the provision stand-by ducts for the pitch and yaw circuits. These changes inevitably incur weight, cost and maintenance penalties.

3.3.2 Final Design Process Using the Criterion of 2 Simultaneous Failures

The system was designed initially to meet the safety target of 1 catastrophic failure in 10^{-7} flights for the flying control system. After this process was completed, Ref.5 was published which described the safety requirements for the flight control system of the active control Jaguar prototype aircraft. This stated:-

"The active control Jaguar is designed to achieve a one-in-ten million failure probability over a one hour flight, surviving any two electrical failures or a combination of a hydraulic and an electrical failure in any order."

The S-80 is also an "actively controlled" aircraft which utilises "fly-by-wire" signalling so it was decided to re-design the system to meet this exceedingly stringent target.

The method of approach chosen was based on the assumption that when any two failures take place at some time, the most severe condition is that they are in two different channels, i.e. the two channels are to be cut off, therefore the multiplication number of the affected control system will be reduced by two.

3.3.2.1 Check of Two Simultaneous Electrical Failures

Previous work in paragraph 3.3.1 produced complex sub-systems with highly reliable performance for both inside and outside parts of the systems. The loss of two electrical channels affected the "in-cockpit" part and reduced the redundancy from four to two channels. Initial checks showed that under these circumstances, the target could not be met. Several alternative solutions were checked, but the only suitable one appeared to be specifying more reliable flight control and air-data computers. The flight time was reduced to 1 hour in line with the target requirement. If the computer reliability could be doubled, the predicted performance would be:-

Channel	Conventional Controls	VTOL Controls
Roll	3.17×10^{-8}	$.03 \times 10^{-8}$
Pitch	3.17×10^{-8}	$.03 \times 10^{-8}$
Yaw	3.18×10^{-8}	$.03 \times 10^{-8}$

It can be seen that the rather arbitrary individual sub-system targets are not all met.

In the most critical case,that of failure of the conventional yaw channel, the overall system failure rate would be this value plus the failure probabilities of the other channels. In this case these channels would not be subject to dual failures as that has already been considered to have occurred int he conventional yaw channel. The total system failure probability was therefore predicted to be 6.17×10^{-8} flight hours, which exceeds the requirement of 1 in 10^{-7} hours.

3.3.2.2 Check of One Electrical, Plus one Hydraulic Failure

The single electrical failure reduced the redundant electrical channels from quadruplex and triplex, which still retained high integrity. The loss of single hydraulic components had little effect on most of the sub-systems because of their inherent hydraulic redundancy. The exception was the VTOL roll channel where a hydraulic failure, associated with an electrical failure gave a value of 39.6×10^{-8} hours. This was alleviated by the use of differential ADEN nozzle actuator as a stand-by measure. The use of this technique led to the following predictions:-

Channel	Conventional Controls	VTOL Controls
Roll	.18	4.02
Pitch	.18	0.02
Yaw	.21	.02

It can be seen that even with the use of differential ADEN nozzles the individual roll VTOL target is not met, but the aircraft system prediction of 4.63×10^{-8} exceeds the system requirement of 1 in 10^7 hours.

3.4 Design to Meet the Mission Reliability Target

The mission reliability performance is rather difficult to predict, as it depends on what are considered to be defects which impair a combat operation. The Aircraft Confirmed Defect rate performance is described in para.3.5 below and was calculated to be 5.75 for 1000 flying hours with the original computer failure rates.

Study of failure modes of mechanical components indicated that catastrophic failure rates comprised approximately one third of total defects. These are the failures that would have significant effect on the aircraft's operational performance. It was further decided that in a wartime situation failure of half of the quadruplicated components would constitute operational failures. These factors were applied to all of the components in the system to give a total of 2.19 operational failures per 1000 hours which met the target of 2.4 per 1000 hours.

The rather simplistic assumptions above should be checked more accurately if the aircraft design were to be refined.

3.5 Design to Meet the Overall Confirmed Defect Rate Target (Maintenance Reliability)

The overall defect rate is a simple sum of all the failure rates of the system. The calculation method is to multiply the failure rate by its number for any component, then add up all the products of the components on the flight control system. The failure rates used for individual components are shown in table 1. Table 2 shows the calculations for the overall defect rate with the original failure-rate computers. The overall rate was calculated to be 5.75×10^{-3} flying hours.

This figure is considerably below the system target of 24×10^{-3} per hour. The discrepancy occurs because the current analysis only concerns the primary flying control system of the aircraft, i.e. that required for safe stability and control of the aircraft. The remaining parts of the flying control system such as the flaps, airbrake, leading edge flaps etc. would contribute defects which would then approach the overall system target.

4. RUDDER PEDAL DESIGN PROCESS

4.1 Requirement

The layout and the characteristics of our aircraft's rudder pedal must conform to all the requirements of Ref.11, but this specification has no relevance to the highly raked seat.

As we know, using a highly raked seat is a general trend in recent years to overcome the effects of high 'g' loads upon the human body. In our case the rake angle of the ejector seat is high with a value of 35°.

It was decided to determine suitable requirements to suit this situation. Reference 10 was studied and dimensions determined for aircrew with the 5th and 95th percentile proportions. These were shown on drawing No.7. The eye reference point must be maintained for all sizes of pilots. This is achieved by means of sliding the ejector seat up and down the inclined seat rail. The foot neutral position therefore moves horizontally and vertically as shown in the drawing. Av.P 970 (Ref.11) calls for an adjustment of 12 inches but recent designs have shown that this is excessive. Study of the 5th and 95th percentile mass showed that 9.5 in would be adequate.

Reference 11 also specifies the strength requirements for the assembly.

The pedal requirements were discussed with Cranfield Institute of Technology's very experienced Chief Test Pilot. He proposed two additional requirements:-

i) The pedal adjustment must be performed easily, and quickly
ii) The pilot's heels should be able to touch the cockpit floor to support his legs during high 'g' manoeuvres. Drawing No.7 shows that this was accomplished by the incorporation of a false floor angled at 12° to the horizontal.

4.2 Alternative Schemes

The initial scheme to be considered was the pendant type (Drawing No.8). This has two hanging pedals - left and right - supported by a long frame off the front bulkhead with pedal adjustment provided by a spring loaded cross bolt in slots in the adjuster boxes, each pedal being adjusted separately and brake application provided by ankle action, (see Drawing No.8).

In attempting to achieve the required adjustment the vertical movement of the pedal becomes excessive and it was felt that too must space behind the cockpit panel was used by the rudder pedal. This scheme was therefore abandoned and attention concentrated on three alternative floor-mounted schemes:-

i) Scheme 1 This is a simple double-pivot system based on that used by the MiG-21 aircraft. The original scheme, however, had the disadvantage that the rudder pedal adjustment was very difficult. This has been simplified by use of a flexible drive from the instrument panel (Drawing No.9).

ii) Scheme 2 This scheme uses a parallel motion linkage using a single pivot (Drawing No.10). This scheme was based on those used by the BAe. Hunter and Harrier aircraft. The pedal adjustment is by means of a spring-loaded pin.

iii) Scheme 3 This scheme was designed to obtain the full 12 inches of movement and is shown in Drawing No.11. The pedal motion is linear and is achieved by the use of a series of parallel sliding components. The rudder pedal is mounted on the slide block which can slide between two slideways with its wheels in the troughs inside the slideways (two wheels for each side). The slideways in turn can slide between the guides, which are fixed to the cockpit false floor by bolts. The guide and slideways are attached to each other by an adjusting pin.

4.3 Use of Weight Estimation to Aid the Choice of the Preferred Scheme

Simple stressing calculaitons were performed and estimates made of the types of metal used in each of the schemes. An estimate was made of the volume of material required to satisfy the strength requirements. This information was used to compare the relative weights of the alternatives:-

Scheme 1	11.50 kg
Scheme 1	10.2 kg
Scheme 3	31.7 kg

The weight estimates were very simple and probably innacurate, but did show that Scheme 3 was considerably heavier than the others primarily because of the steel sliding faces. This scheme was therefore rejected. The accuracy of the calculations was insufficient to differentiate between Schemes 1 and 2 and so it was decided to check their relative reliabilities and ease of maintenance by means of a Failure Mode and effects analysis in para.4.4 below.

4.4 Use of Failure Modes of Effects Analysis to Aid the Choice of The Preferred Scheme.

FMEA normally begins at the component level and proceeds upward towards the highest level of assembly. As part of the analysis process, estimates were made of the likelihood of failures occuring. The potential failures were ranked according to criticality, that is, the component failure having the combination of the most serious effect and the highest chance of occurence was ranked first, and subsequent failures listed in declining order of their combined criticality and probability numbers.

The criticality factor, C_E, was determined using the method of Ref.12:-

$$C_E = F_1F_2F_3F_4F_5$$

F_1 = class of failure factor

F_2 = system effect factor

F_3 = Likelihood of occurrence factor

F_4 = Ease of detection factor

F_5 = Design familiarity factor

Drawings No.9 and 10 show the two competing schemes. Each drawing has a table of the parts used to make up the assembly. Each scheme utilised an output transducer, artificial feed jack and a brake servo jack. These items were identical for each scheme and thus would have equal reliability. They were therefore, excluded from the FMEA. The remaining parts on each scheme were then examined individually using a standard FMEA sheet. An example of this process is shown in Table 3 which shows that each failure mode was examined and its criticality determined by using the above formula.

Summing the criticality factors of both schemes we arrived at the following totals:-

Scheme 1	82.2
Scheme 2	130.8

These figures do not necessarily show the absolute reliability of each scheme but they do clearly show that Scheme 1 was considerably more reliable.

Examination of both schemes showed that both of them were simple to adjust and para.4.3 shows that their weights were similar. The FMEA shows that Scheme 1 was more reliable than Scheme 2 and it was therefore chosen as the preferred scheme.

4.5 Further Work on the Preferred Scheme

Drawing No.12 shows how the preferred rudder pedal scheme is installed in the cockpit of the aircraft. The mounting casting is rigidly connected to the sloping floor and the adjustment arrangement is shown. Drawing No.9 shows that each pedal can be adjusted in the fore-and-aft direction by means of a rack and pinion assembly driven from the instrument panel gearbox by means of a flexible rotary drive. Drawing No.14 shows details of the adjustment gearbox and how it is mounted to the panel. Drive shafts are routed to each pedal so that by turning the handle the pedals are adjusted by an equal amount. It if felt that this design would be simple light and easily maintained. There is ample, easily altered adjustment available to suit a wide range of pilots. The sloping floor arrangment should provide comfortable pedal oepration during high 'g' manoeuvres.

5. DESIGN OF THE ROLL CONTROL REACTION NOZZLE

The initial design of the system utilised only one reaction nozzle per wing. Calculations in para.3.3.1.2 showed that this had insufficient integrity. It was decided to improve this by duplicating the nozzles at the end of the original duct.

5.1 Scheme 1

The initial scheme is shown in Fig.11. The assembly consists of a V-shaped nozzle - one branch downward and another upward, two coaxial shutters, bellcranks stopper and springs. When the aileron is in the neutral position, the two shutters are closed as a result of a balance between the springs and the neutral position stopper. When the aileron is up, the control pin pushes the right bellcrank against the spring and the shutter opens, hot air comes up, so the wing does down. At the same time the shutter remains closed. When the aileron is down, the control pin pulls the left bellcrank and the down shutter opens meanwhile the up shutter does not move, therefore the wing goes upwards.

This scheme has the advantage that the high-energy hot air is only turned through 90° in either direction and, if the radius is relatively large, the pipe losses will be small.

The disadvantages of the scheme are the large number of components, which leads to maintenance, cost and weight penalties and difficulties in fitting the components with the restricted space near the wing-tip.

5.2 Scheme 2

The second scheme was adopted from the system used on the BAe. Harrier. This scheme is shown in Dwg.17 and it can be seen to be simpler and hence more reliable than Scheme 1. When the aileron is deflected down, the shutter swings to the right and opens the nozzle, the hot air comes downwards and the wing goes up. When the aileron deflects upwards, the shutter swings left, so that the nozzle connects with the 'bucket' part of the shutter, the hot air turns 180° flowing upwards, therefore the wing goes down.

The potential disadvantage of the 180° flow was felt to be more than compensated for by the simplicity and robustness of the design and it was decided to use this scheme.

Subsequent design work is shown on the final system shown in Drawing No.17.

6. DISCUSSION

6.1 Fly-By-Wire Aspects

The S-80 Aircraft Project was one which placed particularly severe demands on the design of the flying control system. The combination of conventional aerodynamic controls, vectored thrust controls and active control technology led to an extremely complex system. The use of active controls, associated with artificial stability, precluded the use of conventional signalling between the pilot and the control surface actuators and led to the adoption of electrical signalling - "fly-by-wire'. The system, as originally designed for the MSc. group project was extremely complex (see para.3.1). This complex system, however, had insufficient redundancy to meet the original safety target of 1 failure in 10^7 flights and had to be modified. The modifications described in paragraph 3.3.1.5 enabled the original target to be met. The even more strict target of 2 simultaneous failures could only be met by the use of more reliable computers and additional input from the control system to use differential ADEN nozzle deflection to augment roll control in the hover.

The necessarily inexact calculations showed that, given these modifications, the target could be met, but at considerable cost in terms of complexity, weight, maintenance effort and equipment cost.

The rather arbitrary adoption of the two simultaneous failures target would impose unacceptably high penalties in terms of weight, complexity, initial and life cycle costs and maintainability. This requirement is considerably more stringent than those for conventional aircraft and it is suggested that the original target of 1 failures in 10^7 flights without performance degradation is more reasonable. Any subsequent failures would entail reduced operational capabilities or handling. The setting of such safety targets requires more consideration when the results of FBW flight tests become available.

Analysis described in paragraphs 3.4 and 3.5 showed that the final system should just be able to meet the mission reliability target and comfortably exceed the confirmed defect rate target (maintenance reliability).

Thus it can be seen that the final system exceeds all of the pre-determined safety and reliability requirements, but at considerable cost. Reference 17, however, shows that use of active control, fly-by-wire technology has considerable benefits and should be persued:-

"The benefits of advanced wing design can be achieved fully only if coupled with artificial stability in pitch, made possible by full-time Fly-By-Wire (FBW) controls. This is the dominant reason for adopting full-time Fly-By-Wire on the next generation of combat aircraft. In addition to the benefits in combat performance conferred by artificial stability, the fuel savings are substantial. Improved maximum lift/drag ratio yields improved cruise and loiter economy, reduced lift-dependent drag in manoeuvre means less fuel for combat, conventionally expressed as a number of sustained turns. The total savings in a typical air superiority mission, where 40 per cent of the total fuel is expended in combat, can yield either 20 per cent improvement in combat endurance, or 25 per cent in radius of action, or 40 per cent improvement in combat air patrol time, all of these being additive to the greatly enhanced combat success conferred by the enhanced turning performance.

Before we adopt full-time FBW control, of course, we have to justify the additional weight and complexity of the control system, with its multiplexed sensors, computing and additional power supplies. A recent study showed that the mass penalty for a full-time FBW control system compared with a conventional 'rods and levers' system with small authority autostabilisation was about 50 per cent of the control system mass. This sounds a lot. However, if we rescale the aircraft with natural stability and mechanical controls by increasing its wing area to achieve the same sustained manoeuvre performance as the aircraft with artificial stability, and increase fuel content to restore range, the penalty in take-off mass is many time larger than the penalty in control system mass. This and many other similar paper calculations have amply justified the adoption of full-time FBW for the next generation of combat aircraft. To pave the way for this generation in the UK, a quadruplex digital full-time FBW system designed to producion airworthiness standards has been successfully tested in flight on the Jaguar FBW demonstrator.

Another advantage of FBW is that it enables the aircraft to be manoeuvred more efficiently. In conventional aircraft it is possible for pilots to over-stress the aircraft or to put the aircraft in conditions whereby flight instabilities might occur. This requires great pilot skill if these conditions are to be avoided and inevitably leads to aircraft not been flown to their full limits so that a safety margin is retained. Active controls and 'fly-by-wire have two advantages inthis field and lead to what is known as 'carefree manoeuvring'. The advantages are that the system will automatically take the aircraft to the very limit of performance without overstressing the aircraft and will also automatically alleviate some of the instabilities that a human pilot could not cope with. These properties, however, highlight the initial importance of having reliable computer programs available to control the flying control system.

The use of FBW technology has greatly reduced the need for the design of mechanical components in the signalling part of the system but introduces in involvement by other disciplines. These include electronics, computer architecture, software design, failure monitoring, redundancy control and therefore much co-operation and co-ordination will be required during the design, development and operation of such systems.

6.2 Mechanical Design Aspects

The concentration on the FBW aspects of the design only allowed time to consider the detail design of the rudder pedals and the roll reaction control nozzle. Many valuable lessons were learned, however. The use of a highly raked ejection seat showed that the conventional design requirement for foot position and travel were inadequate. Anthropomorphic statistics and discussion with a test pilot gave useful new criteria.

Several alternative rudder pedal arrangements were considered and use was made of weight analysis to limit the available options to two. Failure Mode and Effects Analysis (FMEA) was found to be useful in discriminating between the remaining solutions to give the arrangement which would have the best reliability and maintainability characteristics.

The original roll reaction control nozzle could only be operated to give roll in one direction, thus both wing nozzles had to be operational to give roll control in the hover. This would not meet safety targets and so the design was modified so that each nozzle would give roll control in either direction, thus giving redundancy. A scheme based on the design of the Harrier aircraft was finally chosen because of its simplicity and high reliability.

6.3 General Points

The use of reliability techniques in both the safety analysis and in the design of the rudder pedals showed how vital this process was. It is impossible to make rational design decisions on such highly complex designs without them. Failure Modes and Effects Analysis also gave valuable insights into how components might fail and how to allow for failures.

7. CONCLUSIONS AND RECOMMENDATIONS

i) This study showed the initial importance of adequate reliability calculations during the design of a complex, high-integrity system.

ii) Active control of an aircraft, using 'fly-by-wire' signalling leads to a complex design, but the performance benefits should more than compensate for this.

iii) Reliability calculations for the S-80 showed that the original design, although using good design practices, fell considerably below the safety required for completely electronically signalled system. A large number of additional components was required to give the redundancy required.

iv) The arbitrary safety target of 2 simultaneous failures in 10^{-7} hours should be investigated more fully as this leads to very complex and costly systems. 1 failure in 10^{7} flights seems more realistic with subsequent failures giving reduced performance. The current flight test work on FBW aircraft such as the Jaguar should be monitored and any lessons learnt incorporated into future FBW designs.

vi) The calculations performed showed that, with some modification, the S-80 should meet all of the safety and reliability flying control targets.

vii) Design work showed the inadequacy of current design requirements for rudder panels of aircraft with highly-raked seats and these should be investigated more fully.

viii) The application of Failure Modes and Effects Analysis gave a good insight into how a component works and could also be used to aid in the choice between competing design solutions.

ix) Knowledge of reliability techniques is vital in the training of any designer of high-integrity systems and it should form part of the curriculum alongside other disciplines such as aerodynamics, stress analysis, cost engineering, etc. This exercise showed the value of actually applying these techniques in a realistic manner rather than a superficial exposure to them during lectures.

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ASHOORI	Leading edge flaps
BALDWIN	Main undercarriage
BURLEIGH	Centre fuselage
CHAKRABARTI	Inner wing
FROSST	Trailing edge auxiliary surface (fixed geometry wing)
HOOD	Nose undercarriage and front bulkhead
HUGHES	Survivability
LIM	Control systems, outside cockpit
LUBBE	Outer wing
MATTHEWS	Rear fuselage
MORGAN	Fins
NADGIR	Rudders
NAWI	Air intakes and engine mounting
RAJAN	Fuel system
ROBINS	Cockpit layout, controls, seat, environmental control
SCOTT	Nozzles and powerplant offtakes
STOCKING	Foreplane
TOWELL	Variable camber wing, main structure
VERGHESE	Variable camber wing, moving surfaces
YAQUB	Nose fuselage.

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TABLE 1 Failure Rate Information

Component	Failure Rate $x10^{-6}$F.Hours	Ref. No. of Source	Comments
Air Data Computer	120	3	
Ball Screw Jack	1.32	13	
Bearing	0.73	7	This is 27% of total failures - only those which are serious.
Bellcrank	0.008	13	Similar to quadrant.
Bolt	0.008	13	
Chain Drive	10		
Clutch	0.59	7	
Connector	0.26	14	Using relevant stress factors.
Discharge Valve (Nozzle)	4	16	
Electrical Actuator	12	3	
Electronic Computer	120	3	
Feel Mechanism	0.002	3	
Gearbox	0.12	13	
Gyro	25	3	
High Temp. Bearing	16.52	7	
Hyd. Actuator	50	3	
Hyd. Motor	30	18	
LVDT	3	3	
Pitot Tube	33	16	
Potentiometer	5	13	
Rod	0.3	15	
Spring Cartridge	0.5	15	
Torque Tube	0.3	13	
Trim Mechanism	48	3	
Universal Joint	5.341	7	

TABLE 2 System Confirmed Defects

Components	No.	Failure Rate (10^{-6}/flying hour)	F.R.xNo.	Note
CONVENTIONAL ROLL CONTROL SYSTEM INSIDE COCKPIT				
Trim mechanism	4	48	192	
Feel mechanism	1	0.002	0.002	
LVDT	4	3	12	
Pitot	4	33	132	Shared by 6 channels
Air data computer	4	120	480	Shared by 6 channels
Roll gyro	4	25	100	
Connectors	24x4=96	0.286	27.456	
Computer	4	120	480	Shared by 6 channels
$\Sigma = 1423 \times 10^{-6}$ Σ 2x(192+0.002+12+100+27.4f6)x10^{-6} $= 2086 \times 10^{-6}$				The layouts are the same for all channels with 3 components shared by all.
OUTSIDE COCKPIT				
Electric actuator	4	12	48	Shared by con. and VTOL Roll Control System
Hydraulic actuator	4	50	200	
Potentiometer	4	5.0	20	
Connectors	10x4=40	0.286	11.44	
Electric actuator	4	12	48	
Hydraulic motors	4	30	120	
Gear box	4	0.12	0.48	
Torque tubes	6+7=13	0.3	3.9	
Bolts	52	0.008	0.416	
Clutches	2	0.954	1.188	
Ball screws	6	1.32	7.92	
Potentiometers	4	5	40	
$\Sigma = 490$ $\Sigma x2 = 980$				Symmetry for left and right

TABLE 2 - continued

Components	No.	Failure Rate (10^{-6}/flying hour)	F.R.xNo.	Note
VTOL ROLL CONTROL SYSTEM				
Gear box	1	0.12	0.12	
Spring cartridges	2	5.551	11.102	
Bellcranks	6	0.008	0.048	
Rods	2	0.3	0.6	
Nozzles	2	4	8	
Σ = 19.87 Σx2 = 39.74				Symmetry for left and right
CONVENTIONAL PITCH CONTROL SYSTEM				
Electric actuator	2x4=8	12	96	Two channels canard and inboard flaperon
Potentiometer	2x4=8	5	40	
Hydraulic actuator	2x4=8	50	400	
Gear box	2	0.12	0.24	
Bolt and bellcrank	6	0.016	0.096	
Bearing	2	0.734	1.468	
Σ = 539				
VTOL PITCH CONTROL SYSTEM				
Electric actuator	8	12	96	duplication
Potentiometer	2x4=8	5	40	
Pneumatic actuator	2x4=8	50	400	
Gear box	2x1=2	0.12	0.12	
Bolt and bellcrank	2x1=2	0.016	0.032	
Nozzle	2x1=2	4	8	
Connectors	2x40=80	0.286	22.88	
Σ = 567 Σx2 = 1134				With stand-by system

TABLE 2 - continued

Components	No.	Failure Rate (10^{-6}/flying hour)	F.R.xNo.	Note
CONVENTIONAL YAW CONTROL SYSTEM				
Electric actuator	4	12	48	
Potentiometer	4	5	20	
Hydraulic actuator	4	50	200	
Connector	10x4=40	0.286	11.44	
Bolts	2	0.072	0.144	
Bellcrank	2	0.024	0.048	
Σ = 280				
VTOL YAW CONTROL SYSTEM				
Electric actuator	4	12	48	
Potentiometer	4	5	20	
Pneumatic motor	4	50	200	
Connector	40	0.286	11.44	
Gear box	1	0.12	0.12	
Chain drive	2	10	20	
Bearings	2	16.52	33.04	
Nozzle	2	4	8	
Σ = 341 Σx2 = 682				With stand-by system
Overall failure rate for system = 5745×10^{-6} = 5.745				

TABLE 3

SUB-SYSTEM Pedal Assembly (Scheme 1) .... DRAWING NUMBER ....

FUNCTION Transfer foot movement to electrical signal as input of computer .... RELIABILITY MODEL DOCUMENT NUMBER ....

RELIABILITY PREDICTION DOCUMENT NUMBER ....

RELIABILITY MODEL EVENT	EVENT FAILURE MODE	ITEM	ITEM FAILURE MODE	FAILURE CAUSE	EFFECT ON SUB-SYSTEM OUTPUTS	EFFECT ON		PREVENTIVE ACTION		COMMENTS					CRITICALITY
						RELIABILITY	SAFETY	DESIGN	QA	F_1	F_2	F_3	F_4	F_5	
	Main bar	1	Fracture	Excess bending moment, fatigue	Loss of electrical signal. No braking		No yaw control Poor landing	Toe straps give control	Maintenance fatigue checks	5	2	0.5	1.3	0.8	5.2
			Lug failure	Excess shear force Poor material	No braking		Poor landing	Toe straps give control	Maintenance fatigue checks	5	2	0.5	1.3	0.8	5.2
			Bolt break	Excess shear force	No braking		Poor landing	Toe straps give control	Maintenance fatigue checks	5	2	0.5	1.3	0.8	5.2
			Bar distorts	Jam	Wrong signal	Performance reduced				5	1	0.5	1.3	0.8	2.6
	Parallel bar	2	Fracture	Excess bending moment	Wrong signal	Performance reduced			Maintenance and checks	1.0	1.0	0.5	1.3	0.8	0.52
			Loss of stability	Excess axial force	Wrong signal	Performance reduced			Maintenance and checks	1.0	0.5	0.5	1.3	0.8	0.26
			Lug failure	Excess shear force	Wrong signal				Maintenance and checks	1.0	1.0	0.5	1.3	0.8	0.52
			Bolt break	Excess shear force	Wrong signal				Maintenance and checks	1.0	1.0	0.5	1.3	0.8	0.52
	Support casting	3	Break	Too big bending moment	Loss of electrical signal		No yaw control Poor landing		Checks	5	2	0.5	1.3	0.8	5.2
			Bolt break	Too big shear force	Loss of electrical signal		No yaw control Poor landing		Checks	5	2	0.5	1.3	0.8	5.2
	Rack	4	Break	Excess bending moment due to bias effort	Loss of electrical signal		No yaw control Poor landing	Toe straps	Checks	5	2	0.5	1.3	0.8	5.2
			Teeth break	Excess shear force	Loss of adjusting function	Performance			Checks	1.0	0.5	0.5	1.3	0.8	0.52

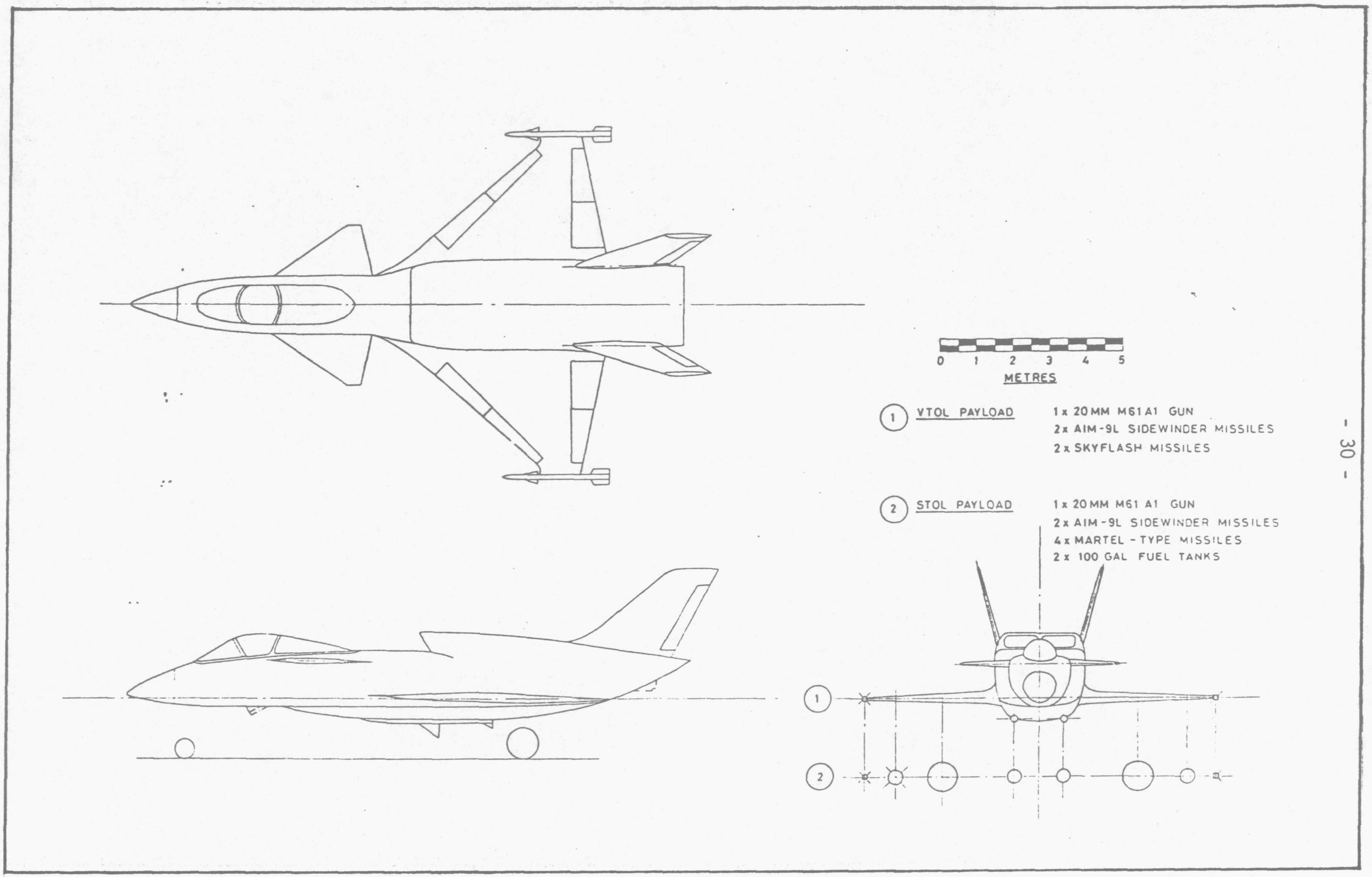


FIG.1 S-80 General Arrangement

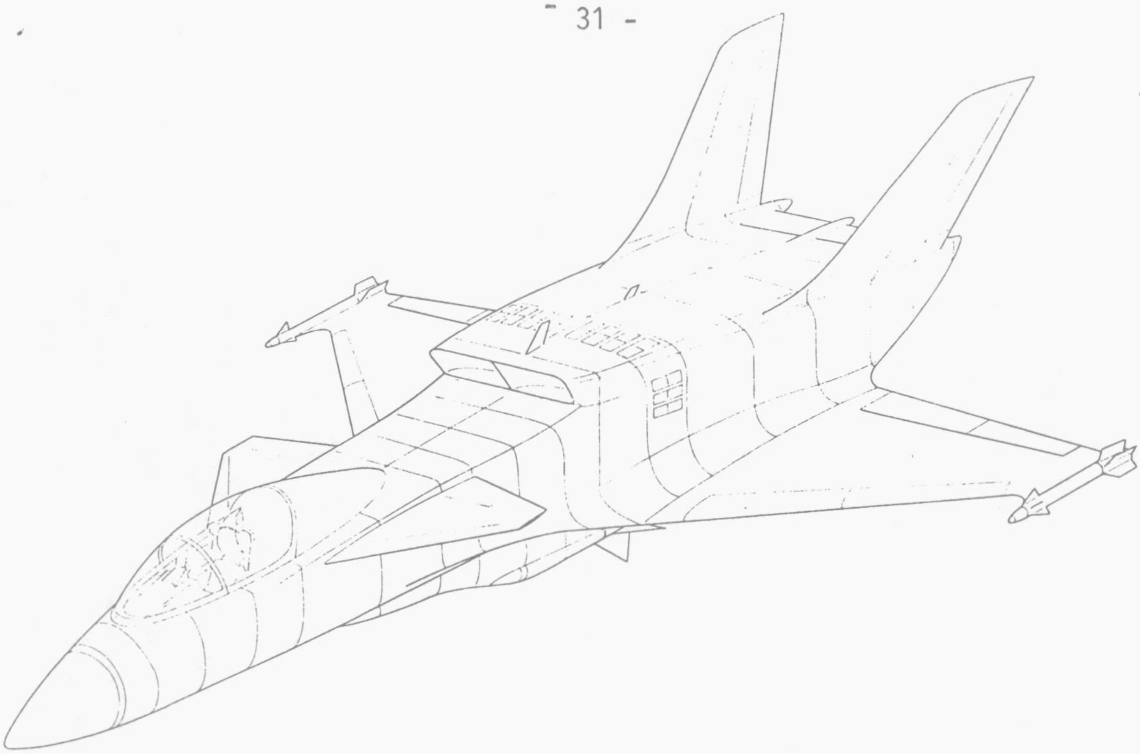


FIG.2 Perspective

FIG.3 Photograph of Cockpit Mock-Up

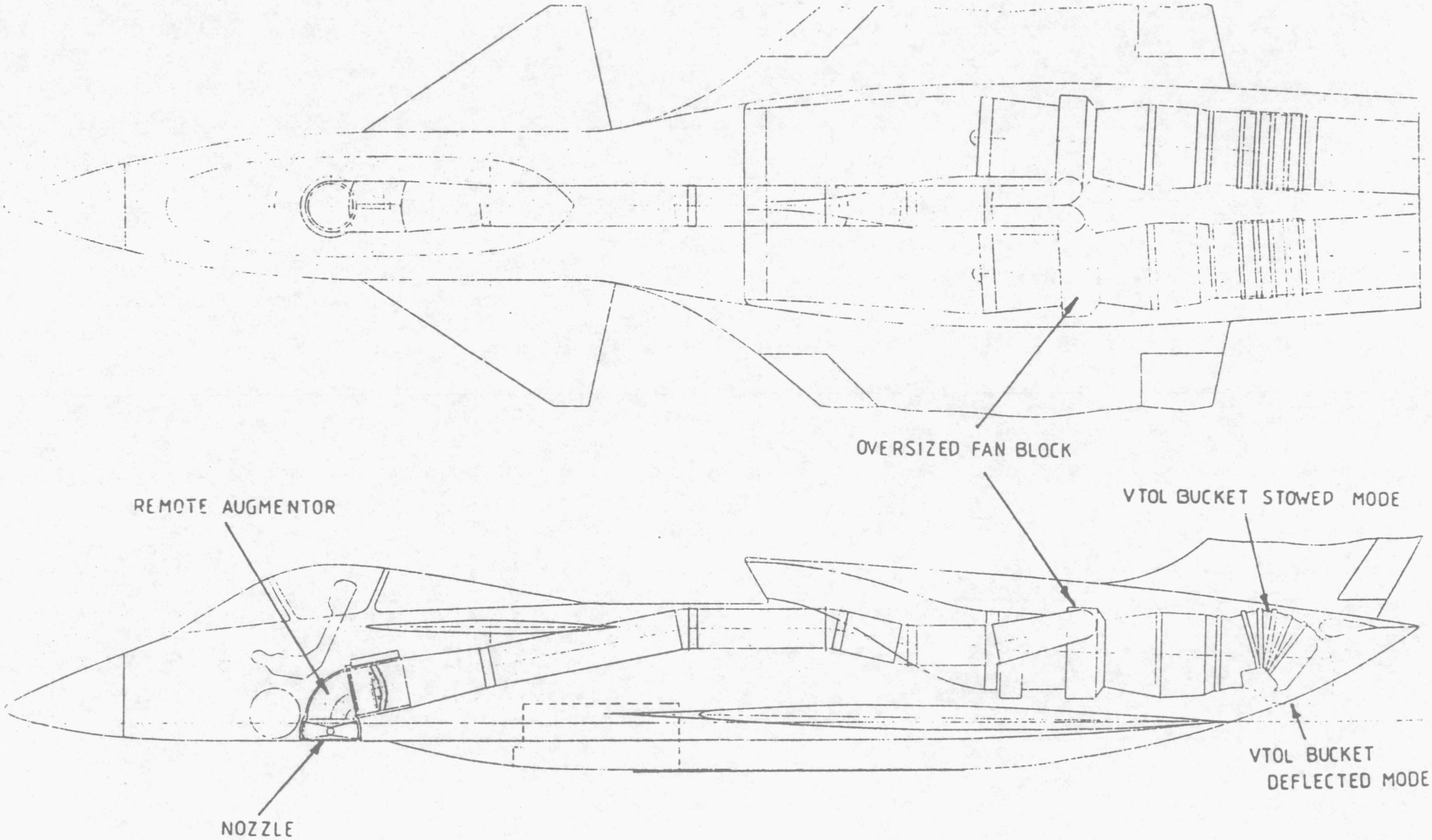


FIG.4 Sketch of the RALS System

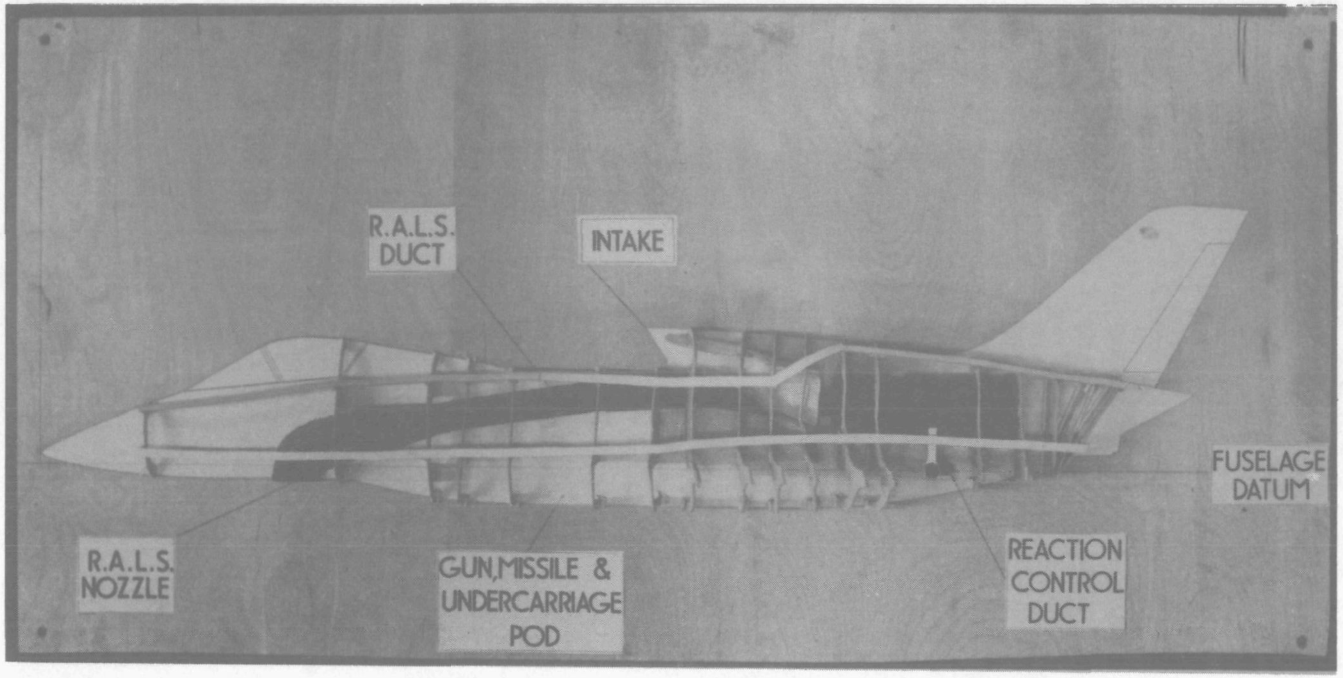


FIG.5 Photograph of S-80 Model

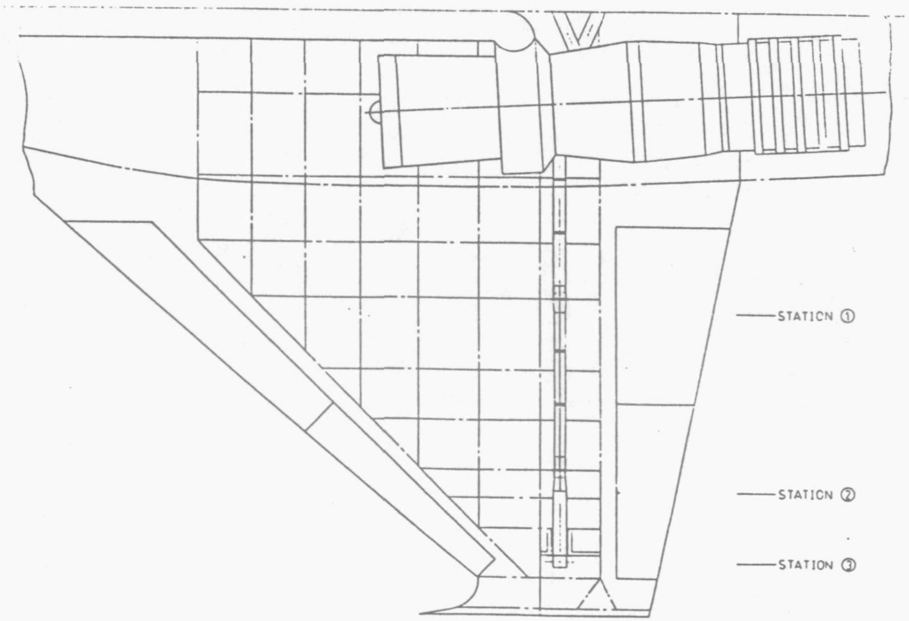


FIG.6 Sketch of Roll RCS System

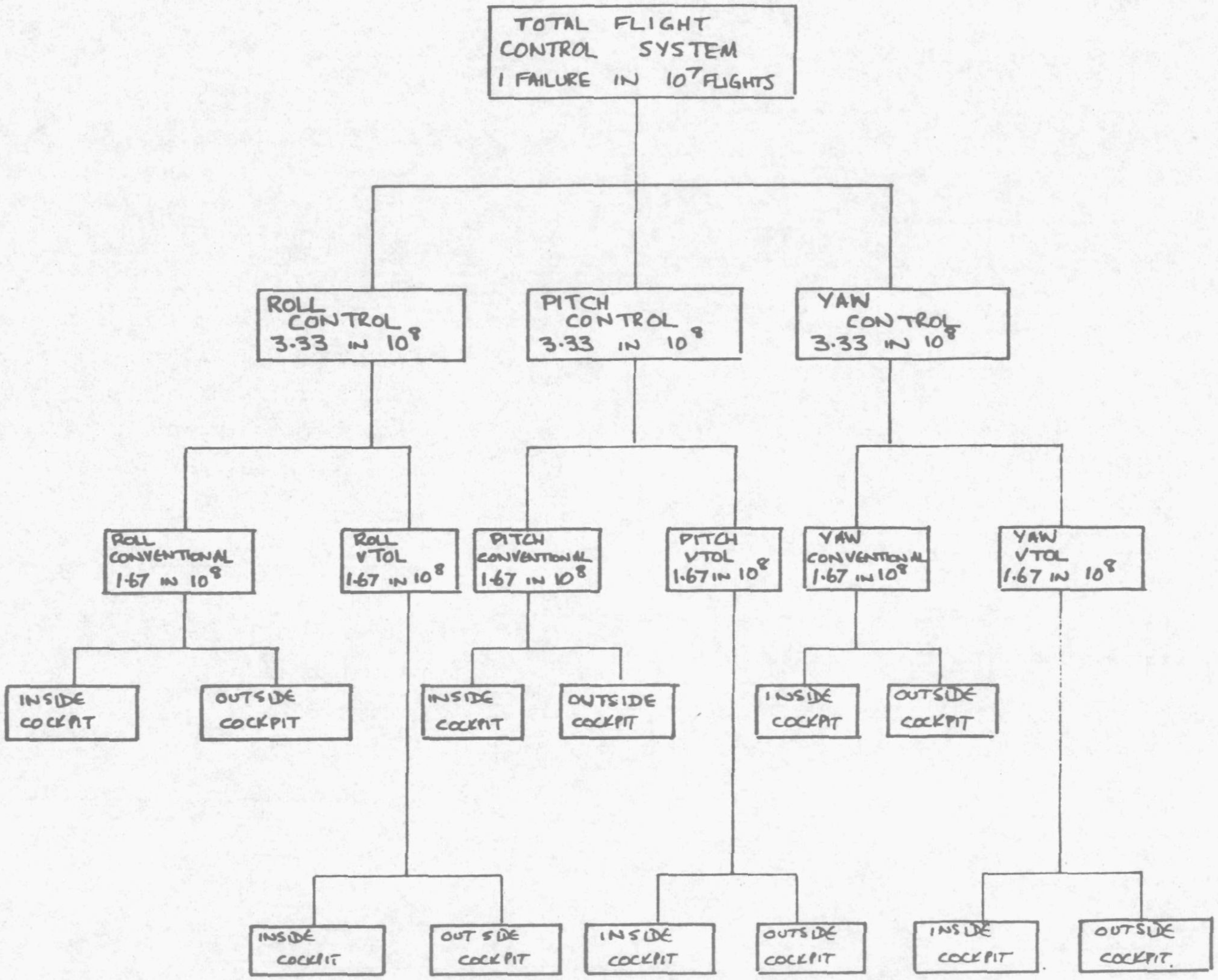


FIG.7 Schematic of Sub-Assemblies

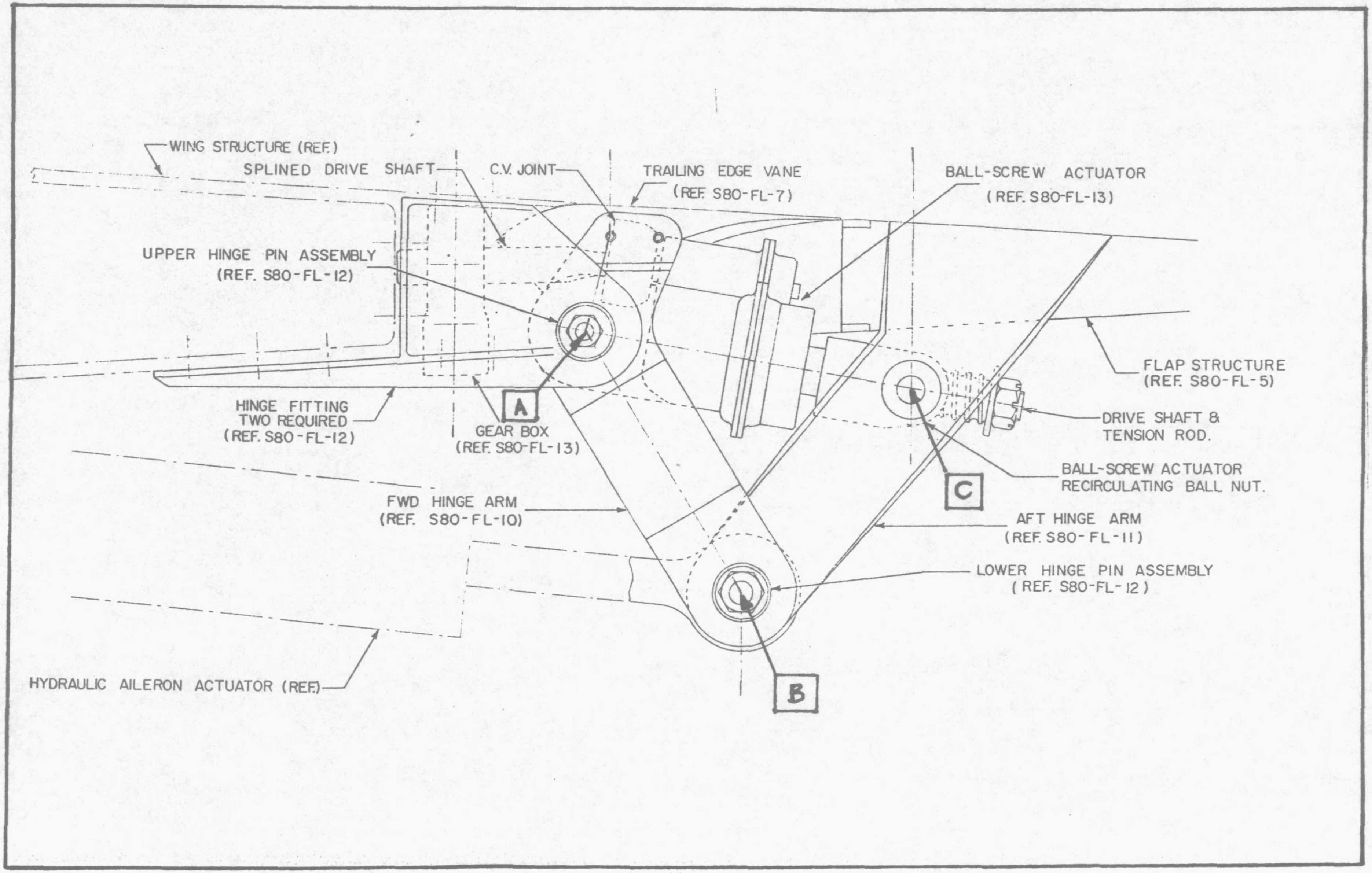


FIG.8 Schematic of Flaperon Transmission

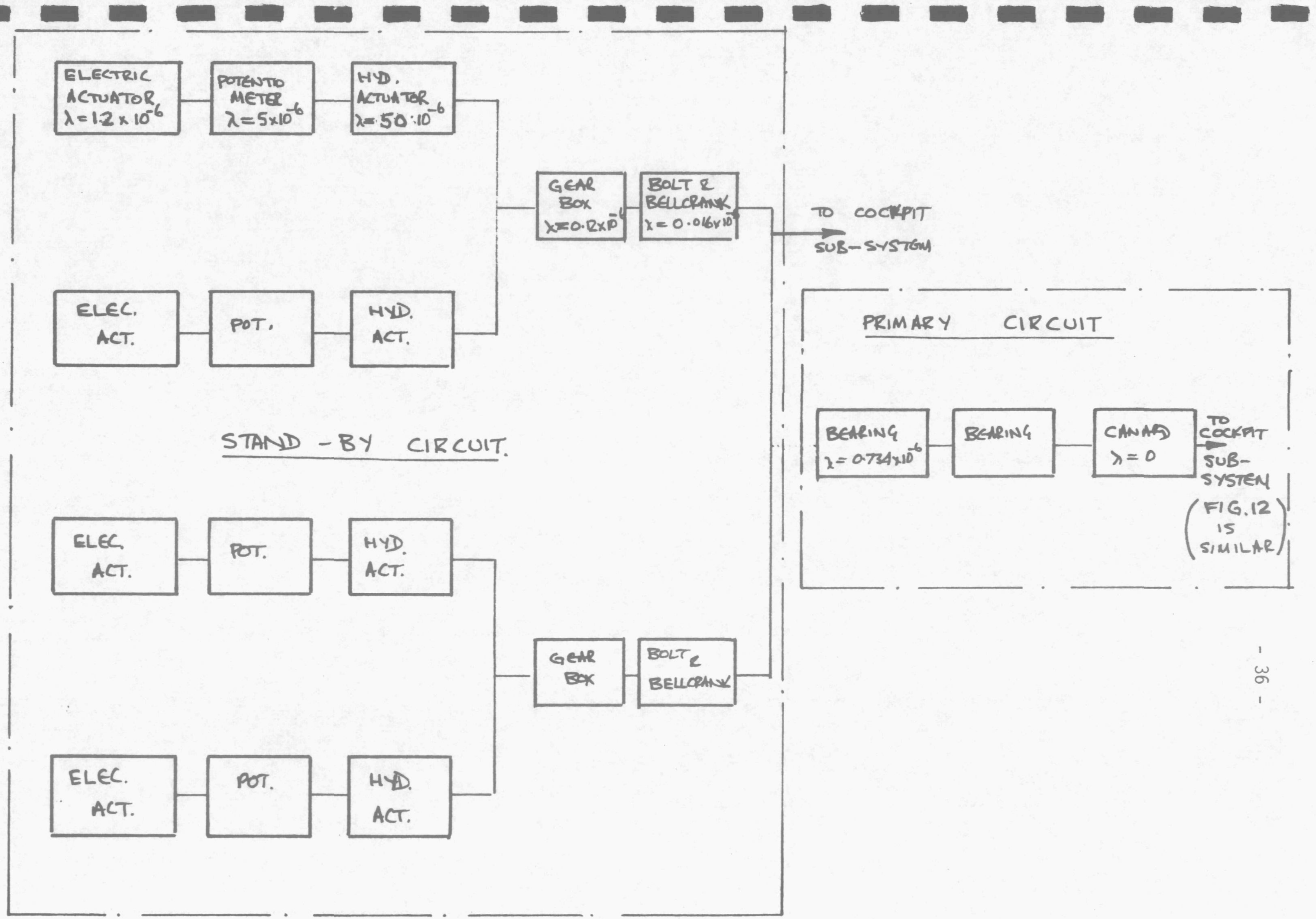


FIG.9 Schematic of Original Pitch Circuit Outside Cockpit - Conventional

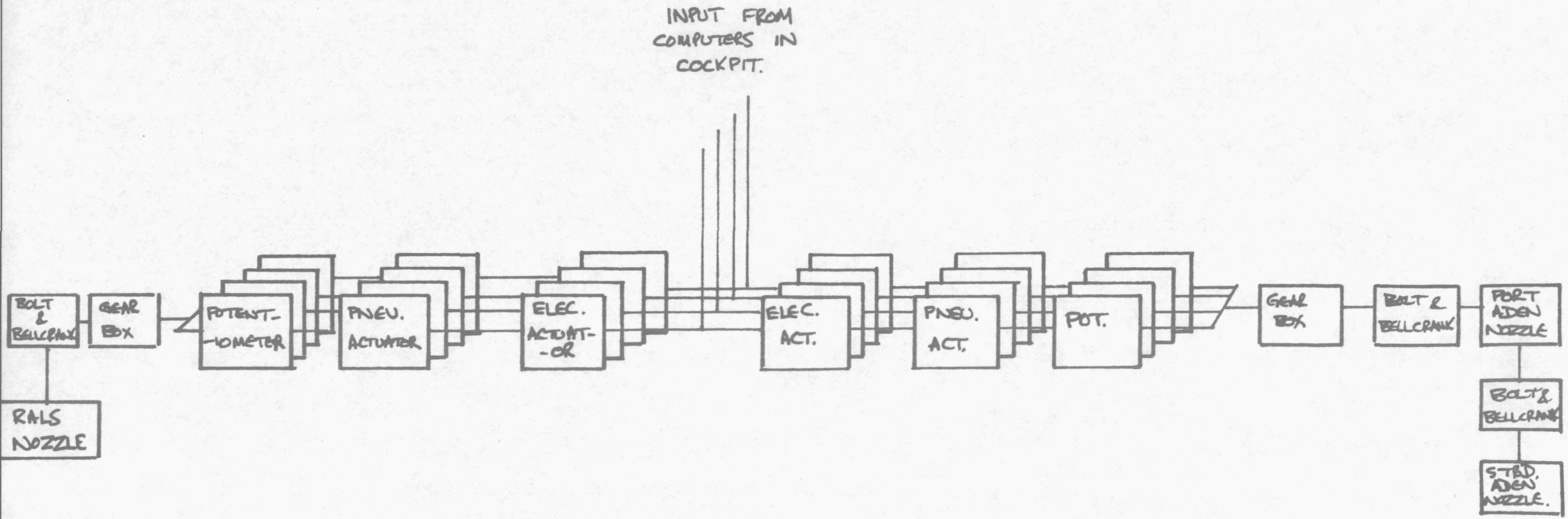


FIG.10 Schematic of Original Pitch Circuit Outside Cockpit - VTOL

FIG.11 Reaction Control System - Scheme 1

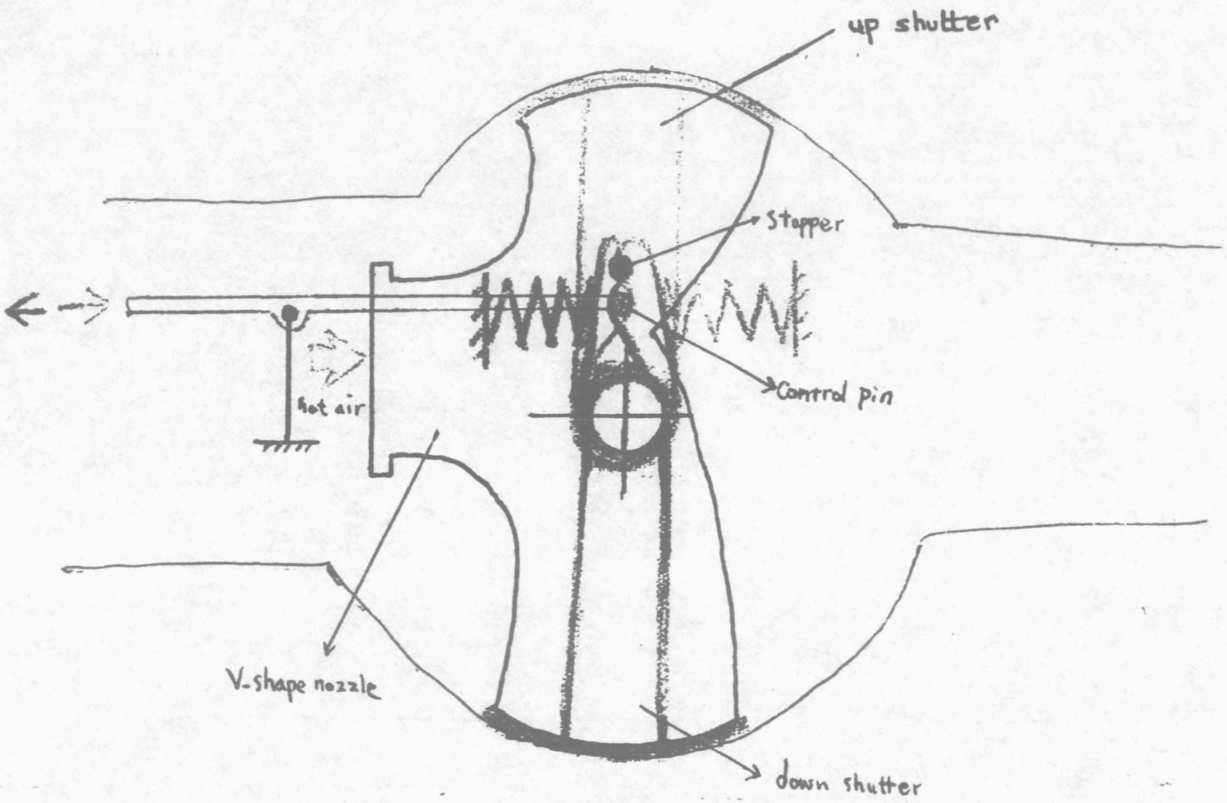
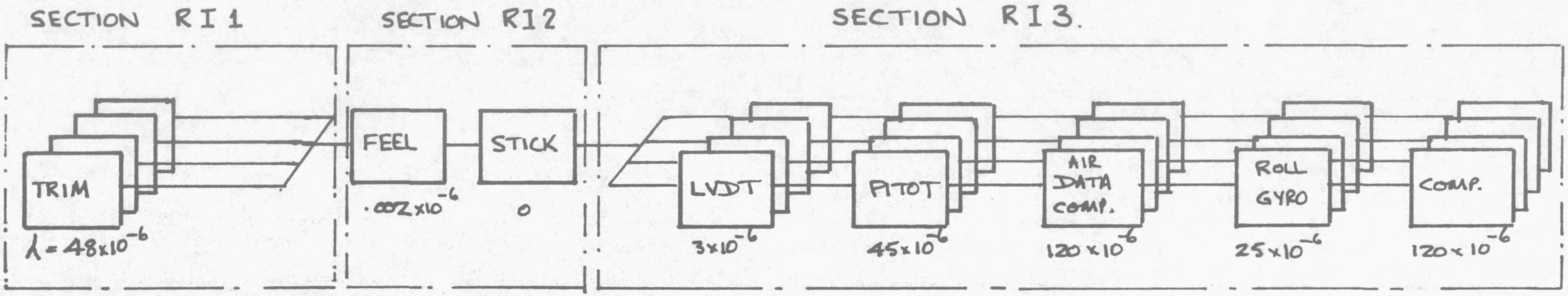


FIG.12 Schematic of Roll Control - Inside Cockpit



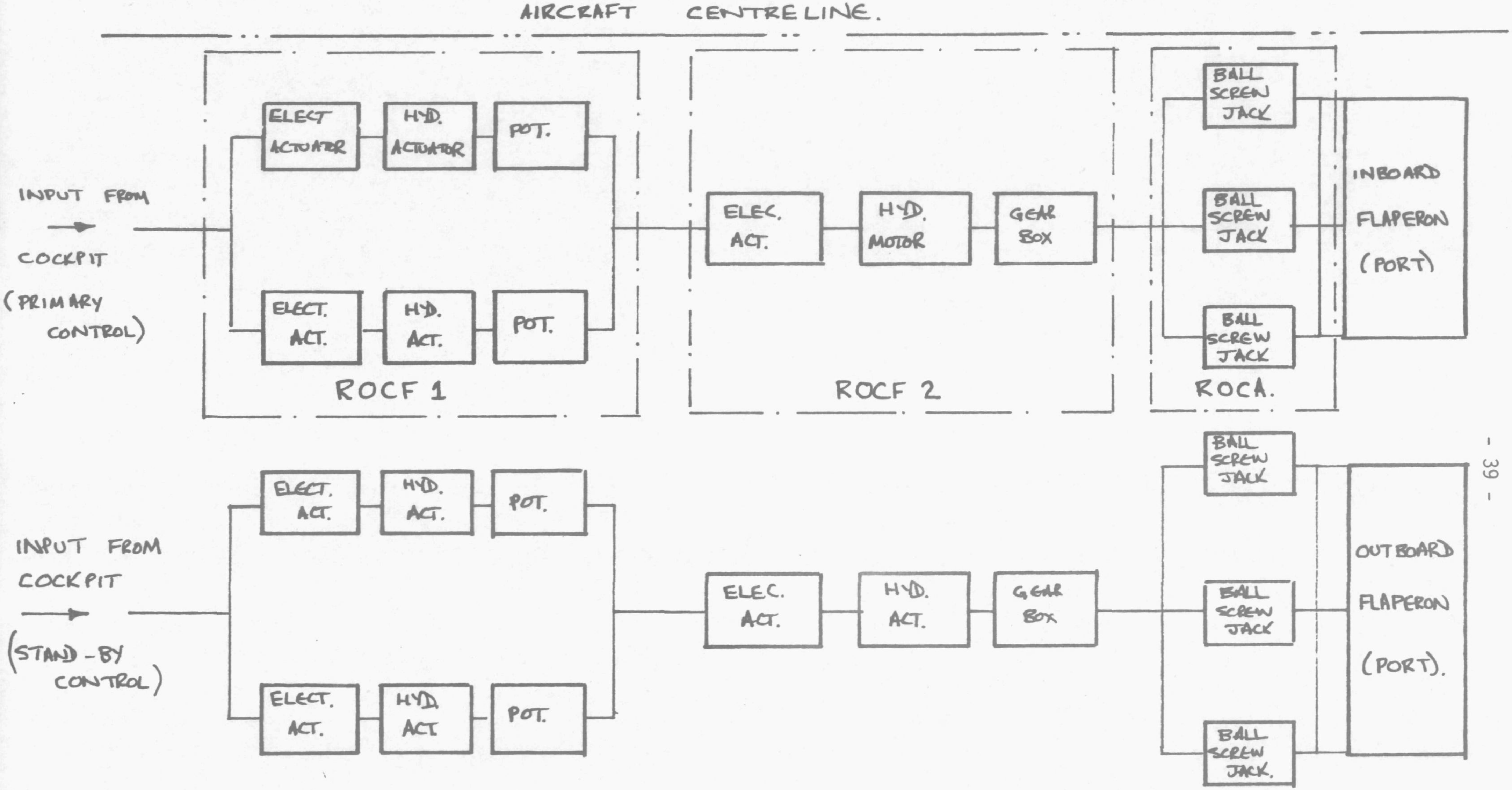


FIG.13 Schematic of Roll Control Outside Cockpit - Conventional

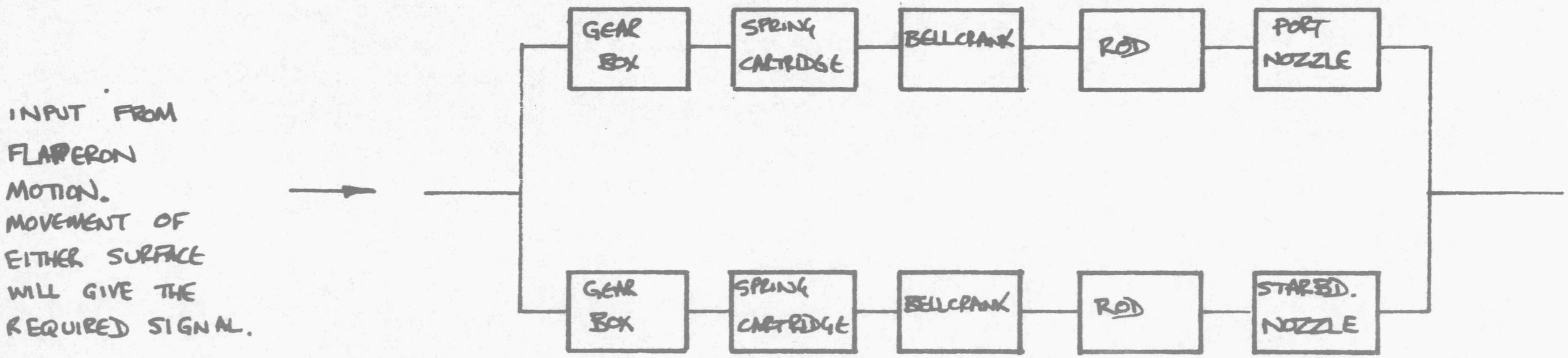


FIG.14 Schematic of Roll Control - Outside Cockpit - VTOL

APPENDIX 1 FLYING CONTROL SYSTEM SAFETY ANALYSIS

1. INTRODUCTION

Fig.7 shows a schematic of the entire flight control system, high-lighting the sub-division into roll, pitch and yaw channels, and subsequent sub-divisions. The targets shown were those used in the initial calculations (see para.3.3.1 of the main text). Calculations were performed for each channel, but this appendix only describes those for the roll channel. The same method of calculation was used for the other channels and also for the subsequent calculations for the more rigorous safety targets (para.3.3.2).

The failure rate data used for the calculations were taken from the information shown in Table 1.

2. SAFETY ANALYSIS OF ROLL CONTROL INSIDE THE COCKPIT

The components in this sub-system are used in both conventional and VTOL modes of operation.

Paragraph 3.3.1.1. of the main text gives the reasoning behind the choice of the negative exponential failure distribution for the purposes of these calculations. A further simplification was made possible by the use of the approximation quoted in Ref.15, which was that:-

For equal parallel sections the exact formula is:-

$$R = 1 - (1-e^{-\lambda t})^2$$

whereas the approximation is $R = 1 - (\lambda t)^2$. The maximum error following from the use of this method was $(\lambda t)^3$ and, as λ is very small, this error is negligible.

Fig.12 shows a schematic diagram of this sub-system, which includes three groups of components. Section R_{RI1} contains the quadruplex trim actuators, R_{RI2}, the feel unit and control column in series and R_{RI3}, the quadruplex computer and air data components. The reliability of the sub-system is:-

$$R_{RI} = R_{RI1} \times R_{RI2} \times R_{RI3}$$

where

$$\begin{aligned} R_{RI1} &= 1 - (\lambda t)^4 \\ &= 1 - (48\times10^{-6}\times2.6)^4 \\ &= 1 - (2.46\times10^{-16}) \simeq \underline{1} \\ R_{RI2} &= 1 - (.002\times10^{-6}\times2.6) \\ &= \underline{0.9999\ 9999\ 48} \\ R_{RI3} &= 1 - (\Sigma\lambda t)^4 \\ &= 1 - (317.58\times10^{-6}\times2.6)^4 \\ &= 1 - (4.65\times10^{-13}) \simeq \underline{1} \end{aligned}$$

$$\therefore \quad R_{RI} = \underline{0.9999\ 9999\ 48}$$

For VTOL mode the time is reduced to 0.08 hours giving

$$R_{RIV} = \underline{0.9999\ 99999\ 8}$$

3. SAFETY ANALYSIS OF ROLL CONTROL OUTSIDE THE COCKPIT

3.1 Conventional Flight Mode

Fig.13 shows a schematic diagram of this sub-system which uses inner or outer flaperons for roll control, thus providing a parallel system. The analysis is complicated by the fact that the hinge point for the aileron motion is provided by the actuation system required to drive the flaperons in the symmetric flap mode. The reliability of the aileron surface itself is taken to be 100% in this analysis.

The sub-system reliability is therefore:-

For Inboard Flaperons

$$R_{ROCI} = R_{ROCA} \times R_{ROCF}$$

where

$$R_{ROCA} = 1 - (\Sigma\lambda t)^3$$

$$= 1 - (1.32\times10^{-6}\times2.6)^3$$

$$= \underline{0.999999999}\ 6$$

Thus the reliability of the triplicated screw jack system was $\doteq$ 1.

$$R_{ROCF} = R_{ROCF1} \times R_{ROCF2}$$

$$= [1 - (43\times10^{-6}\times2.6)^2] \times [1 - (27\times10^{-6}\times2.6)]$$

$$= \underline{0.99993}$$

$$\therefore\quad R_{ROCI} = \underline{0.99993}$$

This figure applies to the inboard flaperon on one side of the aircraft, therefore:-

$$R_{ROCI_{BOTH}} = R_{ROCI}{}^2 = .99986$$

When we introduce the outer flaperon as the parallel channel we have:-

$$R_{ROC} = 1 - (1-R_{ROCI_{BOTH}})^2$$

$$= \underline{0.99999998}$$

Total sub-system is R_{ROC}

$$= R_I \times R_{ROC} = \underline{0.999999975}$$

3.2 VTOL Flight Mode

Fig.14 shows the schematic diagram of this sub-system. The RCS nozzle gives roll control in each direction for each nozzle. The control for the nozzle is derived from the motion imparted to the aileron by its hydraulic actuators.

In this arrangement the normal aileron motion is quadruplex and its $R \approx 1$.

The remainder of the sub-system

$$R_{ROV} = 1 - (\Sigma\lambda t)^2$$

$$= 1 - (4.844 \times 10^{-6} \times .08)^2$$

$$= \underline{1 - 1.5 \times 10^{-13} \simeq 1.}$$

Total sub-system reliability is therefore

$$R_{RV} = R_{ROV} \times R_{RIV} = 0.9999\ 99998$$

$$\therefore \quad \underline{\text{Failure Rate} = 2 \times 10^{-9} \text{ flights}}$$

4. SUMMARY OF PERFORMANCE

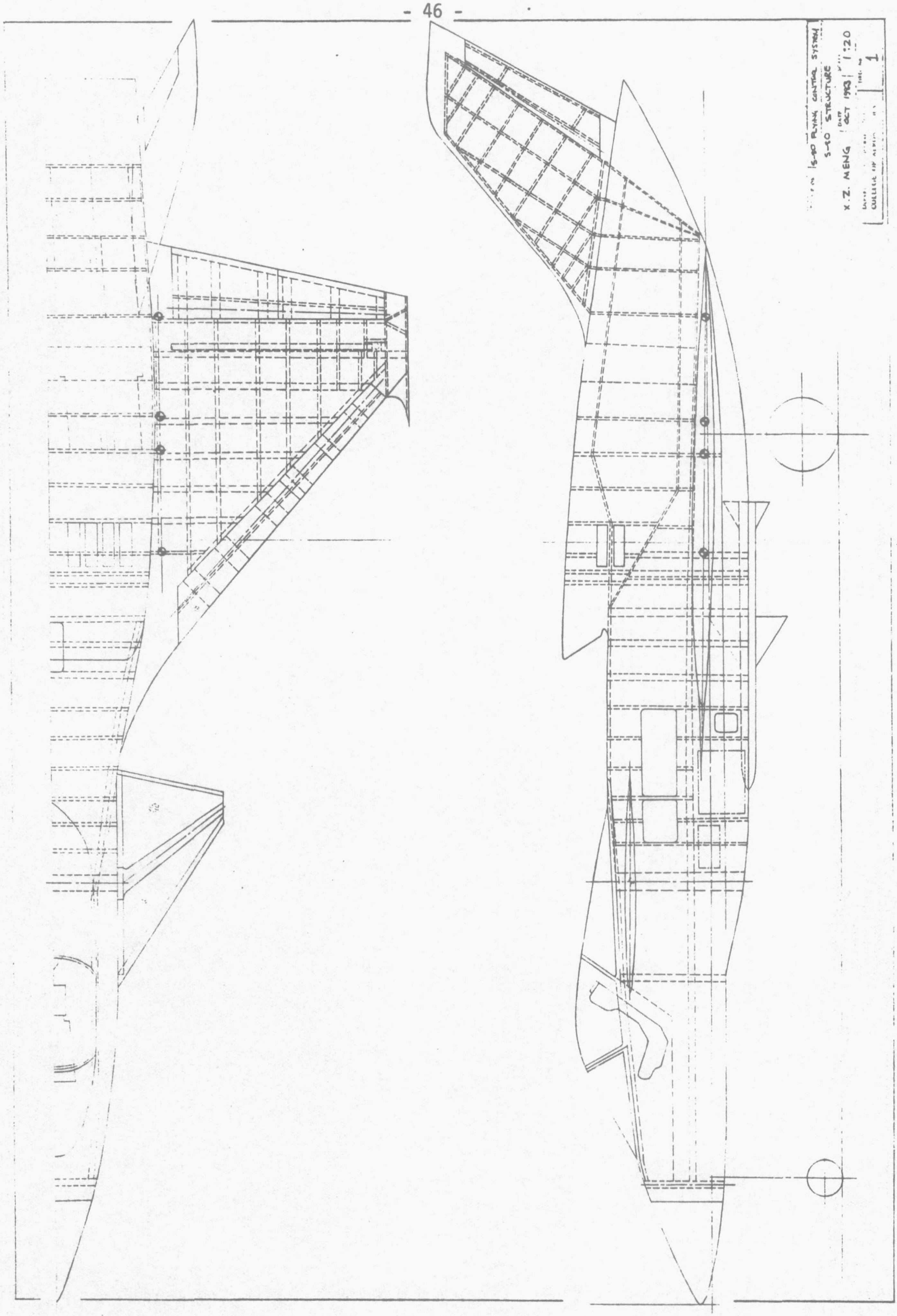
The above calculations show that the predicted performance of the roll control parts of the flying control system was:-

Channel	Target Failures/10^8 Flights	Predicted
Roll-Conventional	2.48	1.67
Roll-VTOL	0.2	1.67

The targets were somewhat arbitrary, but subsequent calculations for the pitch and yaw circuits yielded the predicted failure rates which are shown in para.3.3.1.5 of the main text. These show that the <u>total</u> system prediction of 4 x 10^{-8} exceeds the target of 1 x 10^{-7} flights.

The main text also describes how the final design was determined after use of the more stringent targets of para.3.3.2. The analysis used in that process was similar to that used in this Appendix, but will not be reproduced for brevity.

DRAWINGS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S-80 STRUCTURE
X.Z. MENG
OCT 1983
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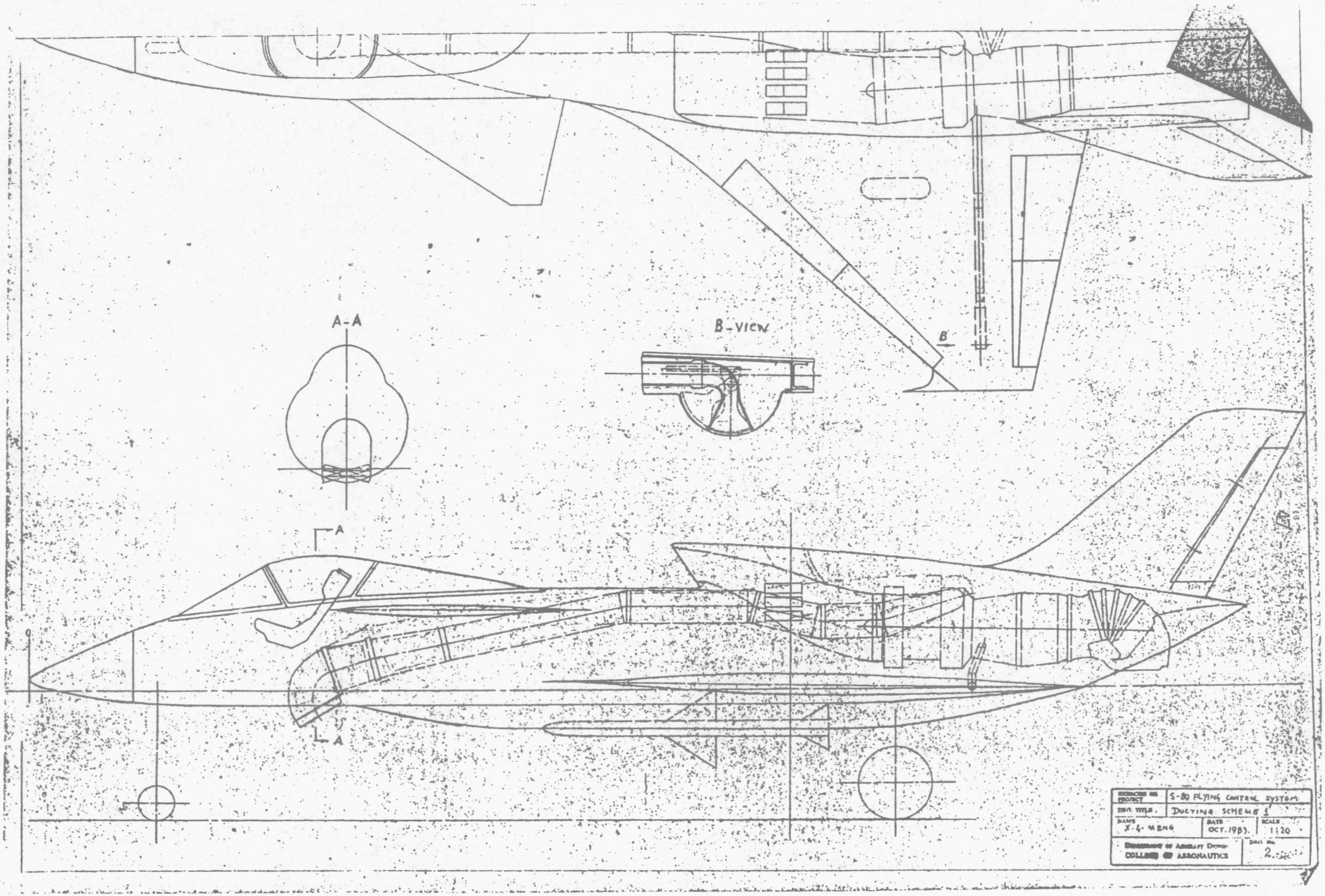
A-A
B-VIEW
B
A
A
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X. L. MENG
OCT. 1983.
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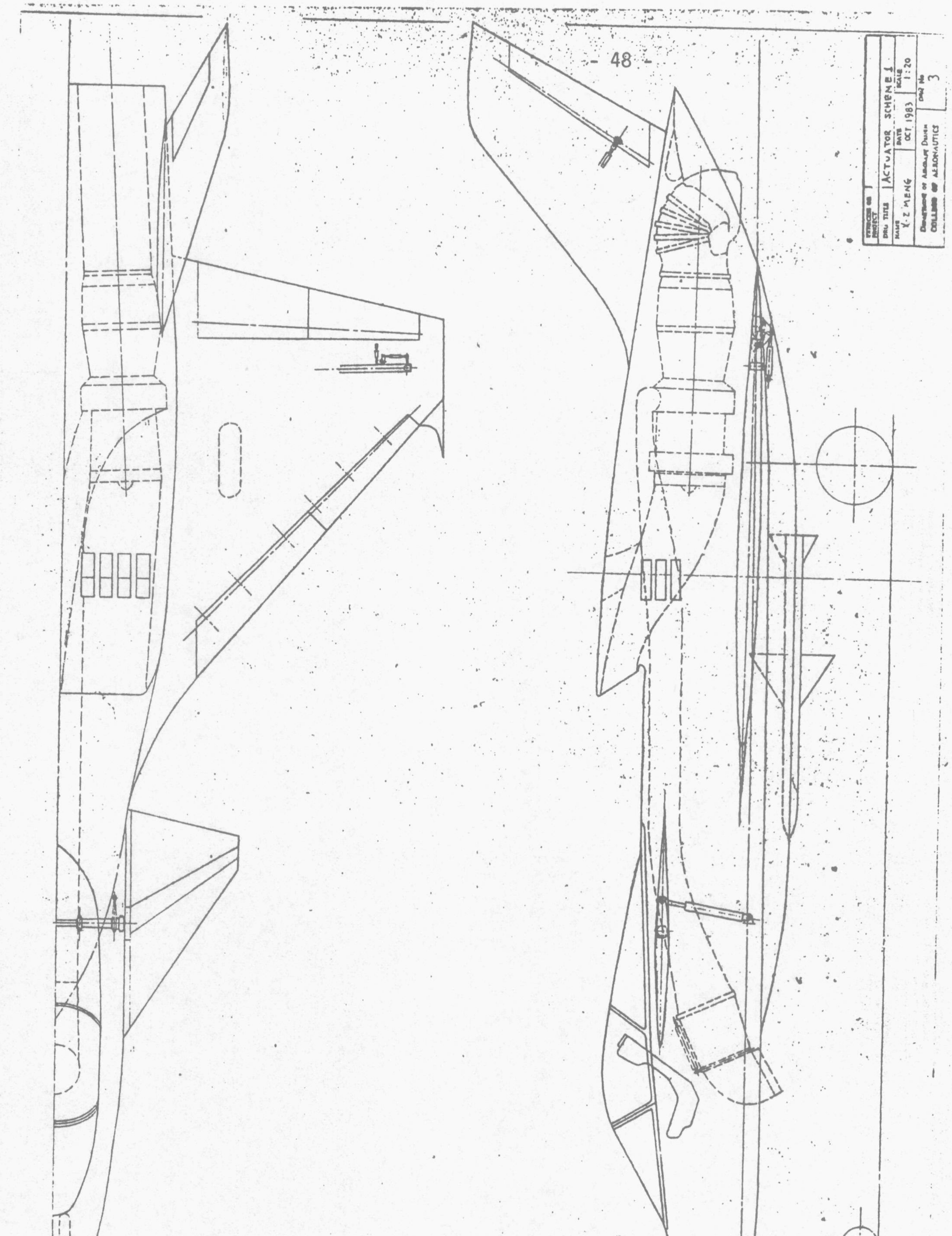
2.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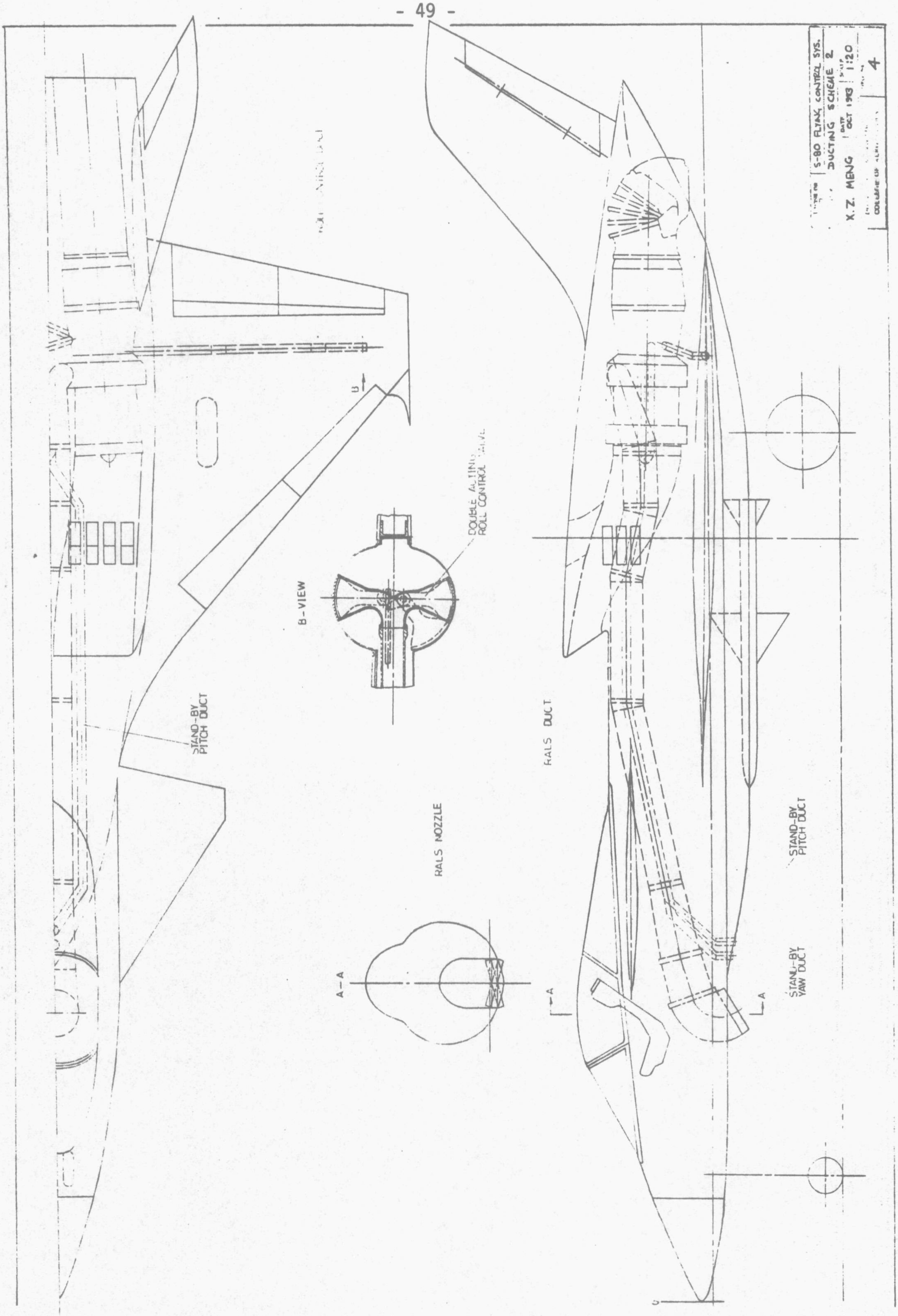
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OCT 1983
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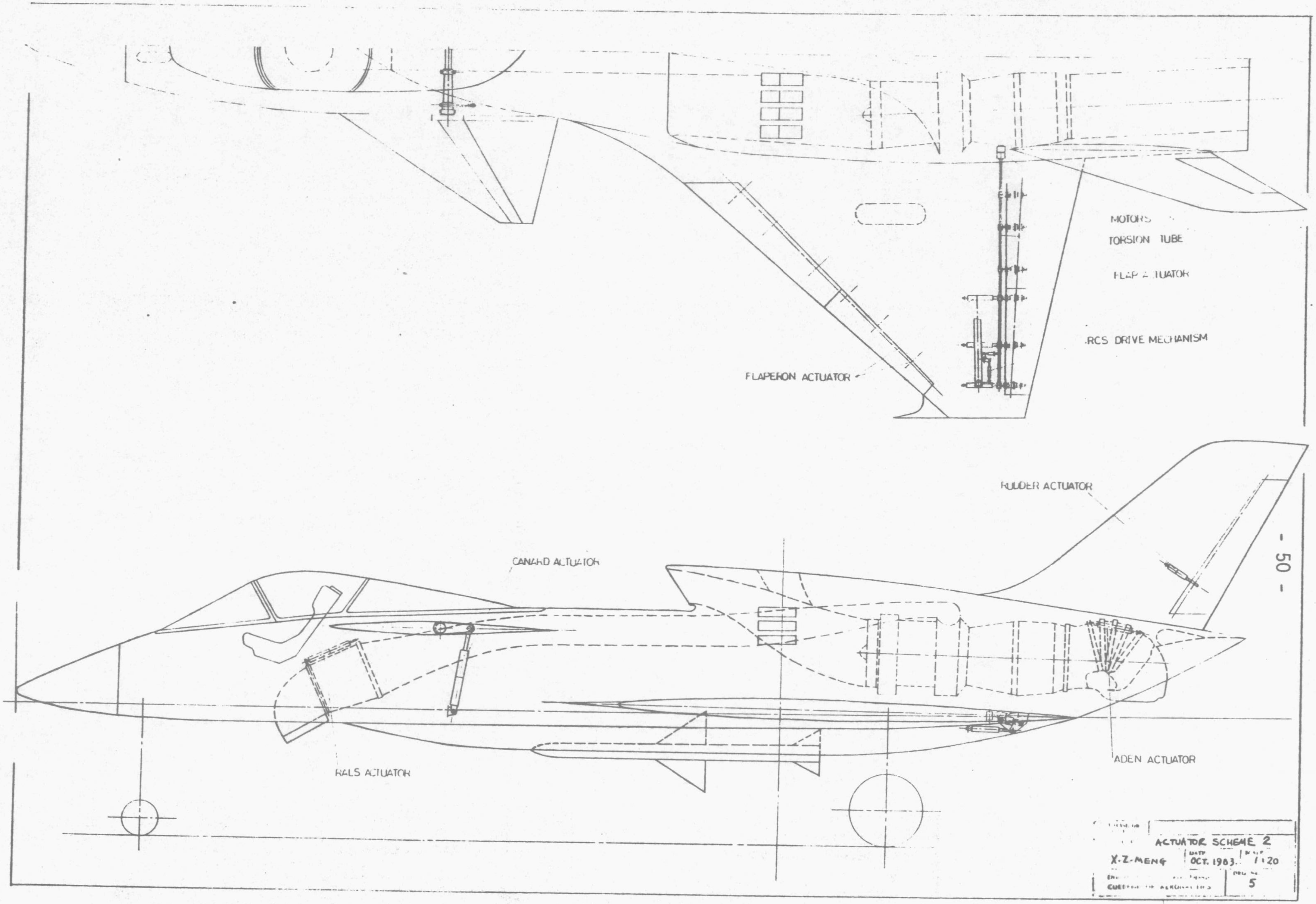
RALS DUCT
RALS NOZZLE
A-A
STAND-BY PITCH DUCT
STAND-BY YAW DUCT


MOTORS
TORSION TUBE
FLAP ACTUATOR
RCS DRIVE MECHANISM
FLAPERON ACTUATOR
RUDDER ACTUATOR
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RALS ACTUATOR
ADEN ACTUATOR
ACTUATOR SCHEME 2
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OCT. 1983.
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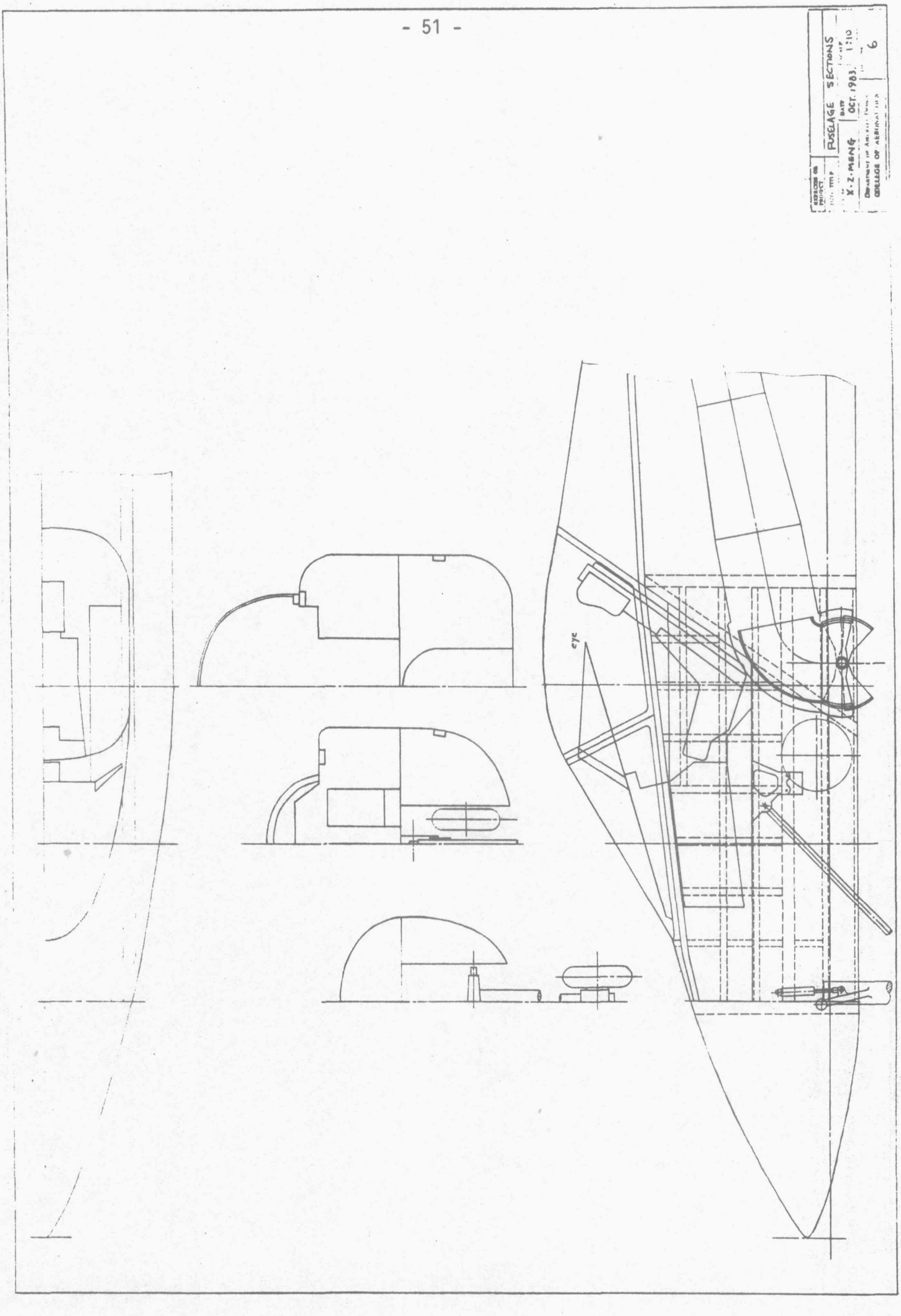
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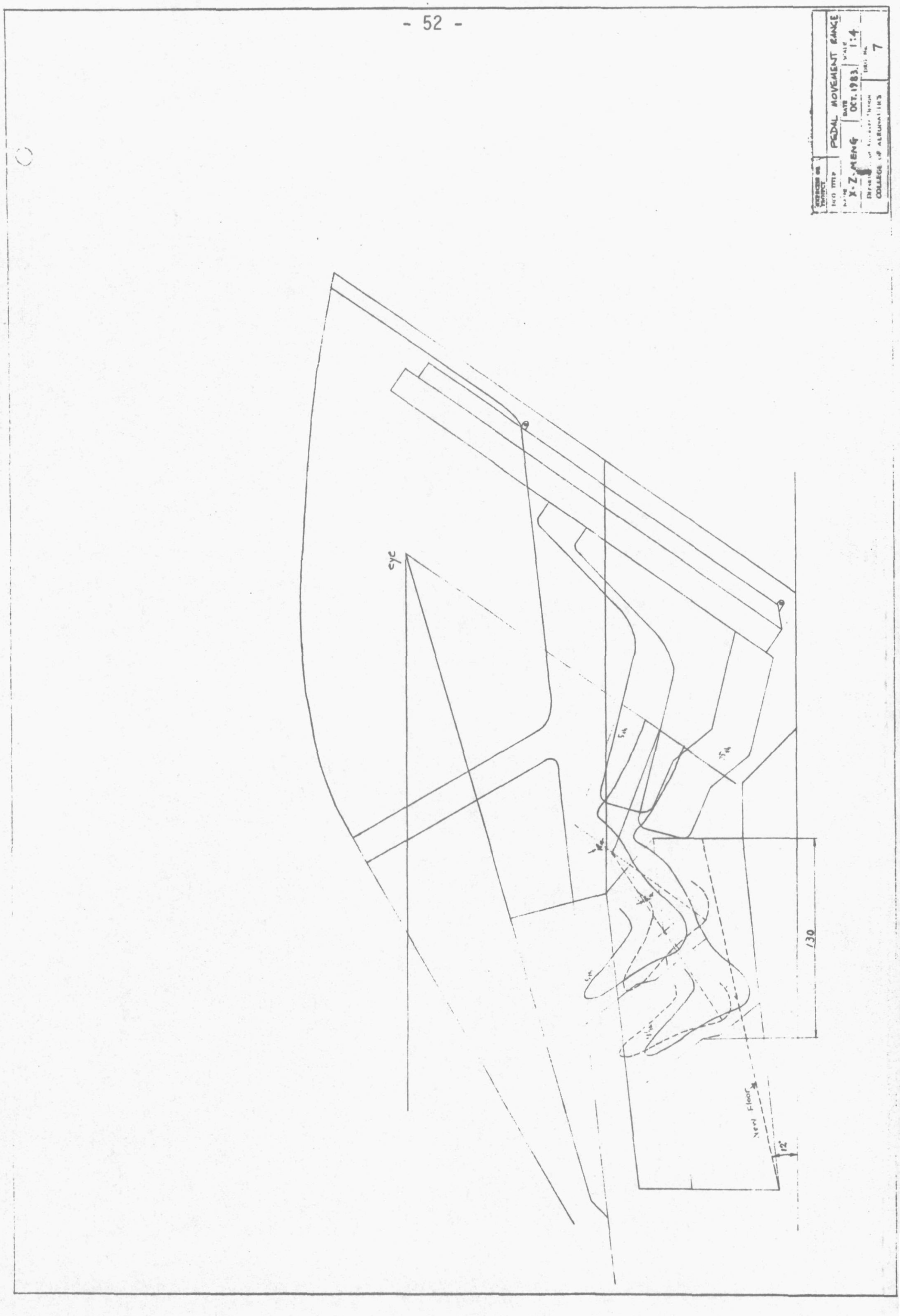
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DATE
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7
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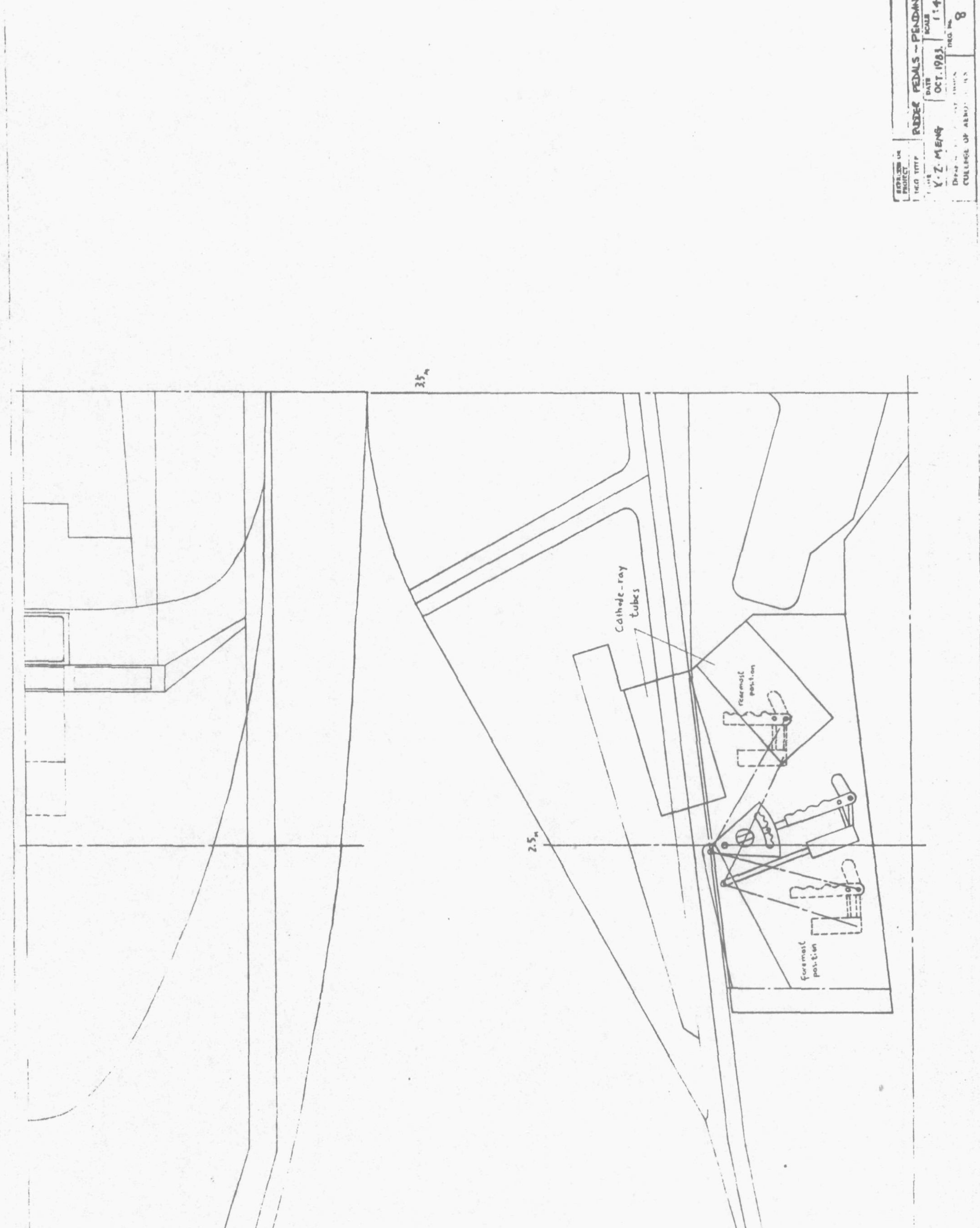
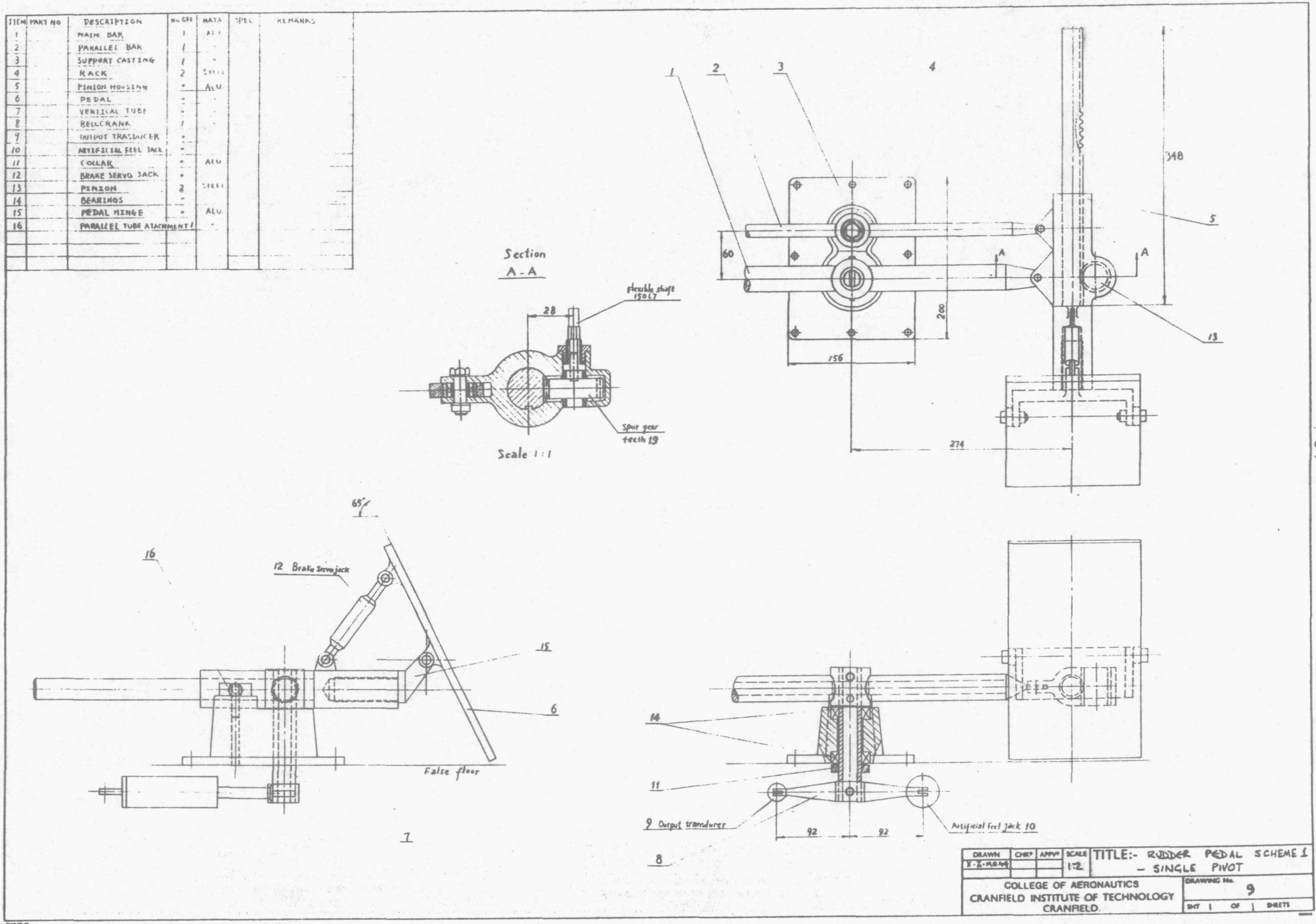
eye
130
new floor
12°


RUDDER PEDALS - PENDANT
Y. Z. MENG
OCT. 1983.
1:4
8
3.5 m
2.5 m
1.52 m
Cathode-ray tubes
rearmost position
foremost position



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2		PARALLEL BAR	1	"		
3		SUPPORT CASTING	1	"		
4		RACK	2	STEEL		
5		PINION HOUSING	"	ALU		
6		PEDAL	"	"		
7		VERTICAL TUBE	"	"		
8		BELLCRANK	1	"		
9		OUTPUT TRASDUCER	"			
10		ARTIFICIAL FEEL JACK	"			
11		COLLAR	"	ALU		
12		BRAKE SERVO JACK	"			
13		PINION	2	STEEL		
14		BEARINGS	"			
15		PEDAL HINGE	"	ALU		
16		PARALLEL TUBE ATACHMENT	1	"		

ITEM	PART NO	DESCRIPTION	NO OFF	MAT'L	SPEC	REMARKS
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2		BELLCRANK CASTING	1			
3		MAIN TUBE	1	"		
4		PARALLEL TUBE	1	"		
5		TOE PEDAL	2	"		
6		HEEL PEDAL	2	"		
7		SUPPORT BRACKET	"	"		
8		ADJUSTMENT STRUT	"	"		
9		ADJUSTER BAR	1	STEEL		
10		MAIN TUBE	1	ALU		
11		SLIDE	1	"		
12		BRAKE SERVO JACK	"			
13		ARTIFICIAL FEEL JACK	"			
14		OUTPUT TRANSDUCER	"	ALU		
15		PEDAL SUPPORT	2	"		
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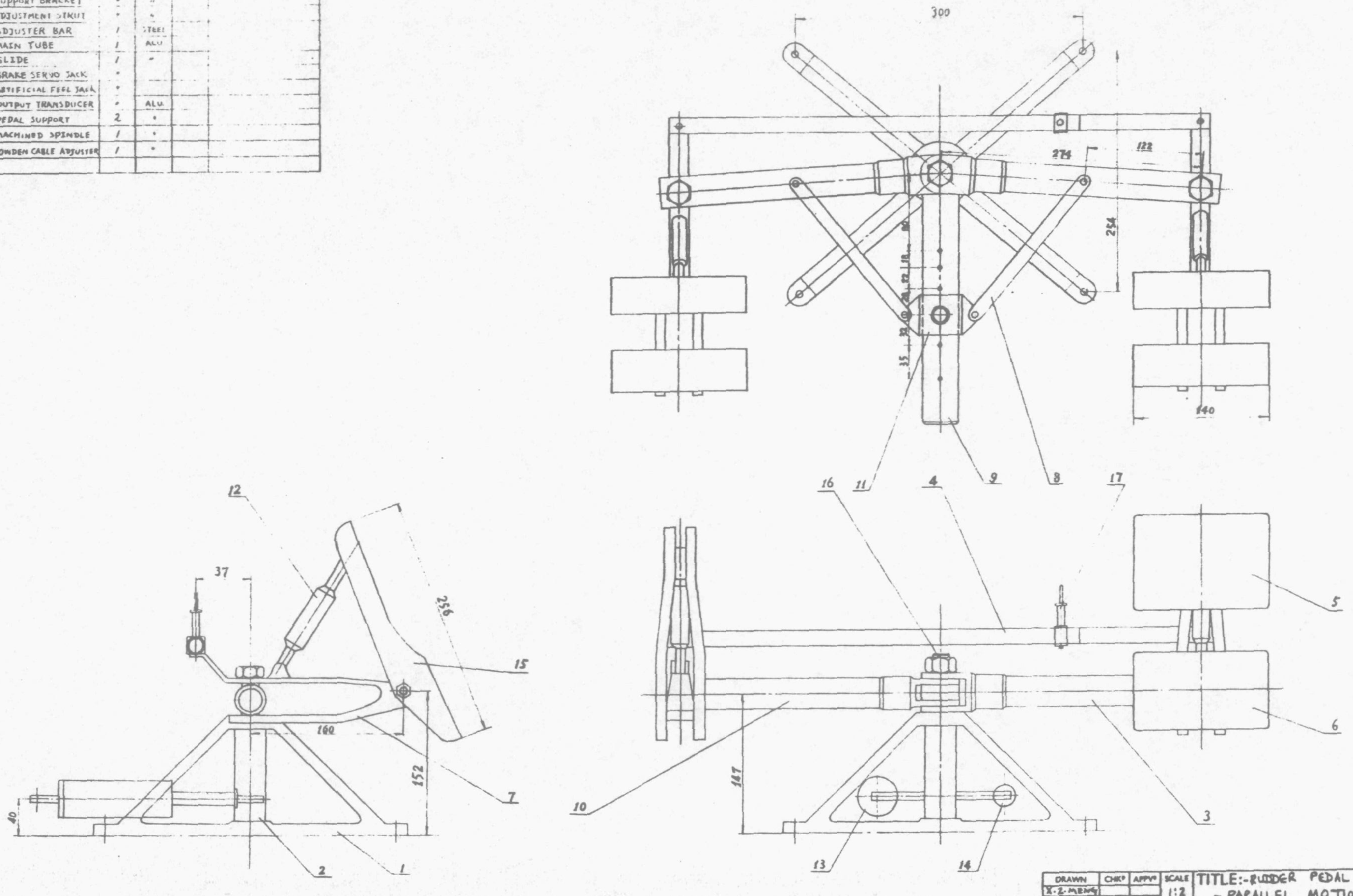

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X. Z. MENG			1:2	

COLLEGE OF AERONAUTICS
CRANFIELD INSTITUTE OF TECHNOLOGY
CRANFIELD.

DRAWING No. 10

SHT 1 OF 1 SHEETS

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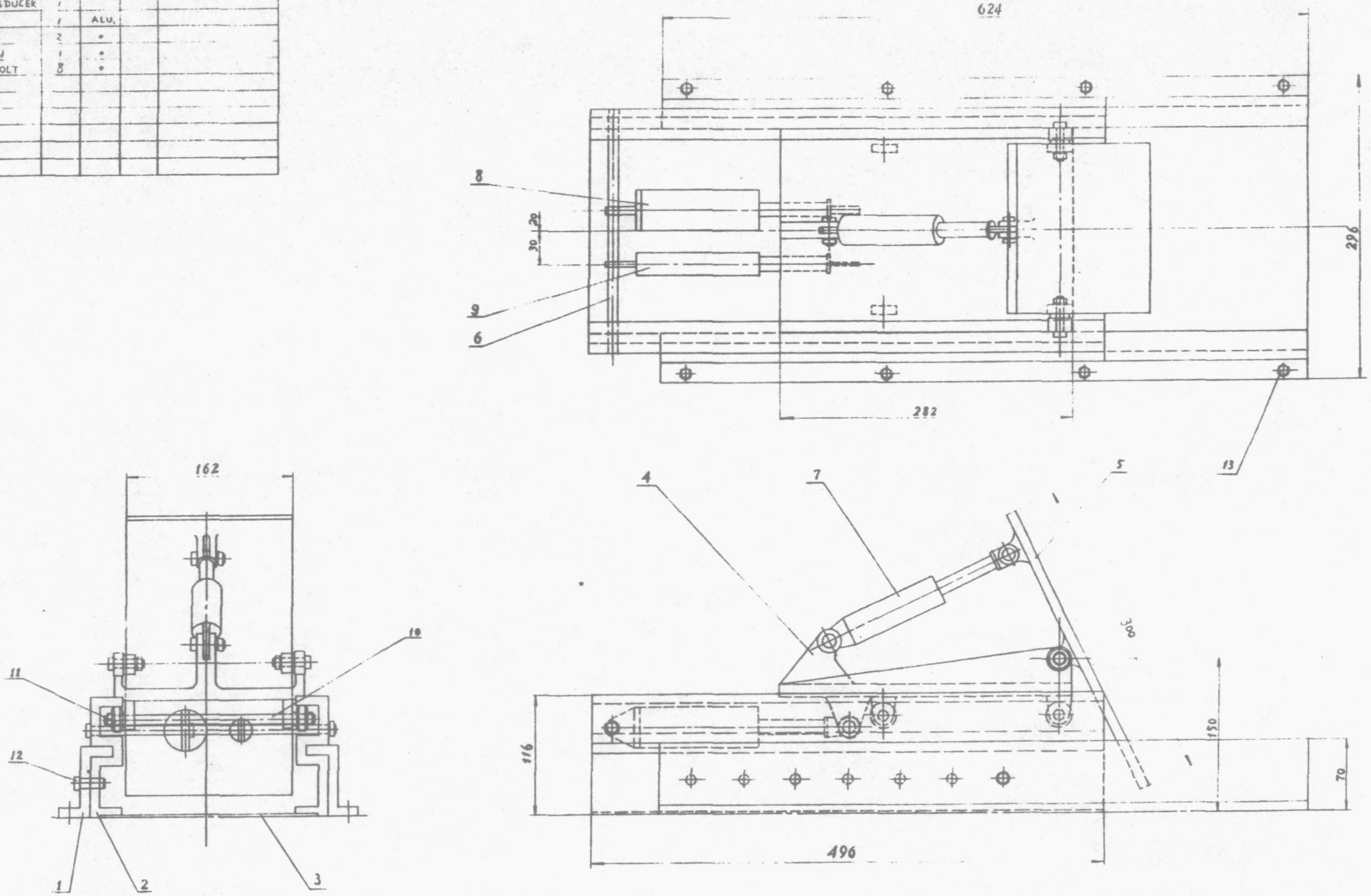
4		SLIDE BLOCK	1	"		
5		PEDAL	1	"		
6		MOUNTING ROD	1	STEEL		
7		BRAKE SERVO JACK	1			
8		ARTIFICIAL FEEL JACK	1			
9		OUTPUT TRANSDUCER	1			
10		SHAFT	1	ALU.		
11		WHEEL	2	"		
12	,	ADJUST PIN	1	"		
13		MOUNTING BOLT	8	"		



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Y. Z. MENG			1:2	
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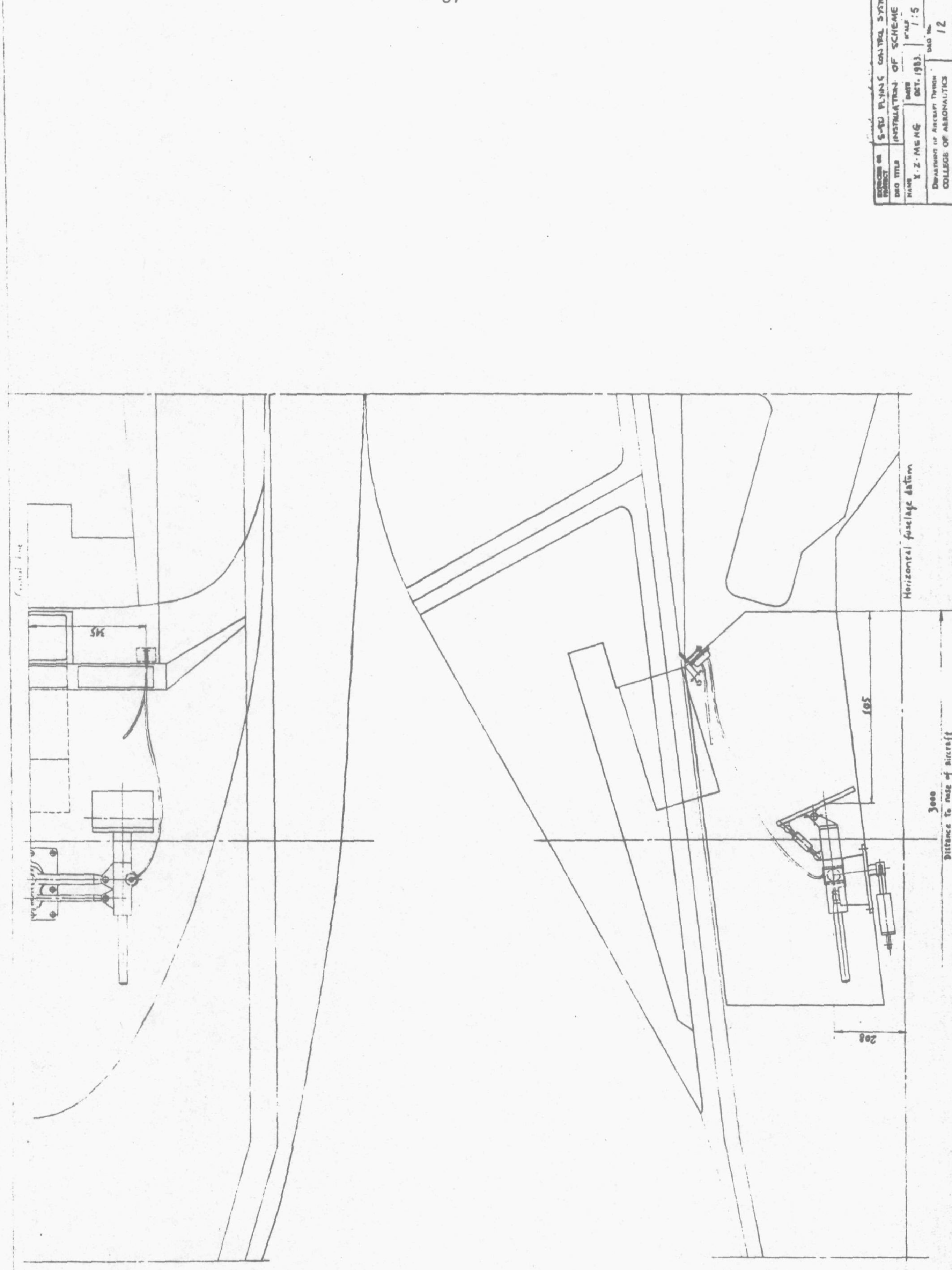
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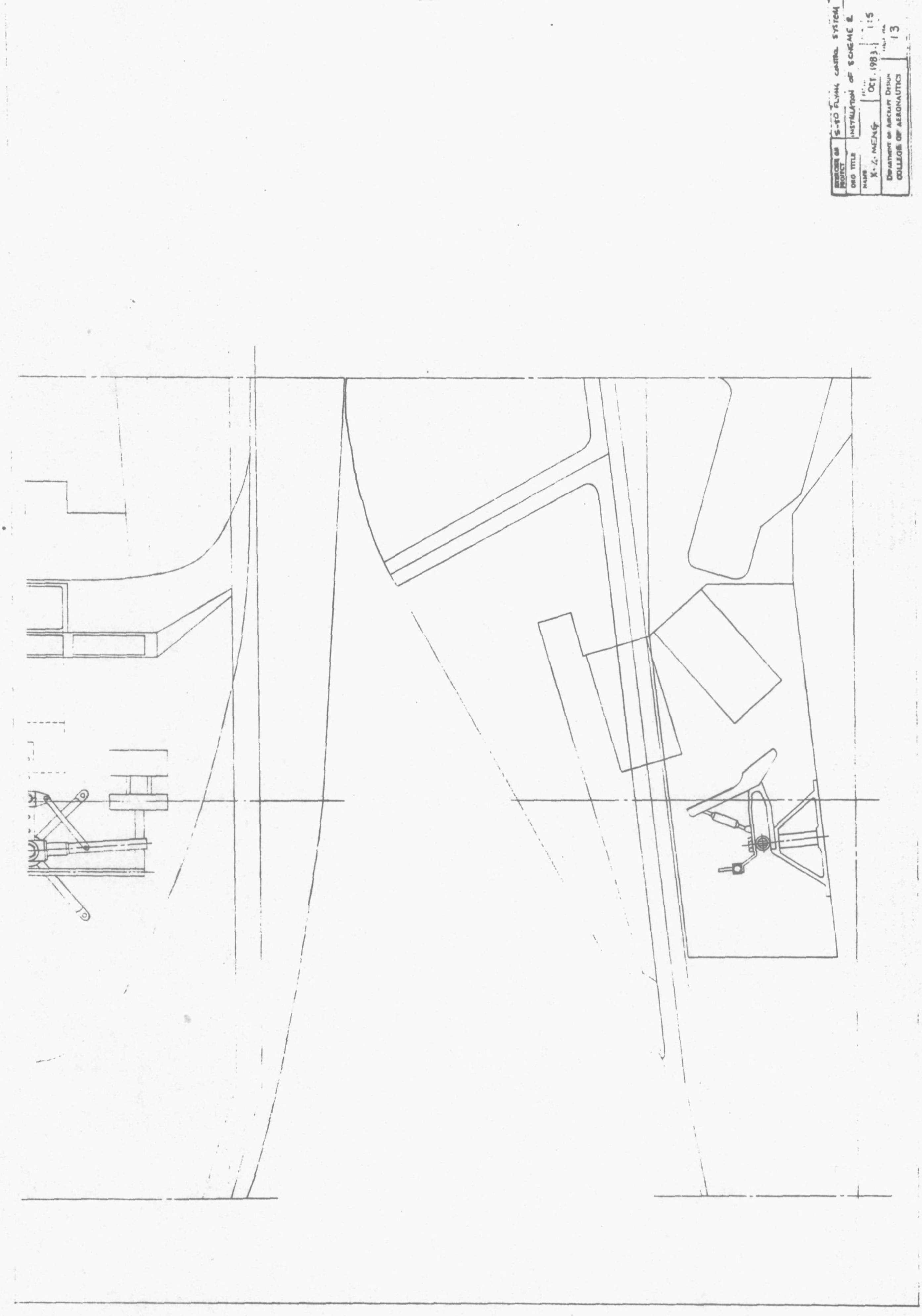
INSTALLATION OF SCHEME 1
NAME
Y. Z. MENG
DATE
OCT. 1983.
SCALE
1:5
DRG No.
12
COLLEGE OF AERONAUTICS
Horizontal fuselage datum
505
3000
Distance to nose of aircraft
208
516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S-80 FLYING CONTROL SYSTEM
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NAME
X. Z. MENG
OCT. 1983.
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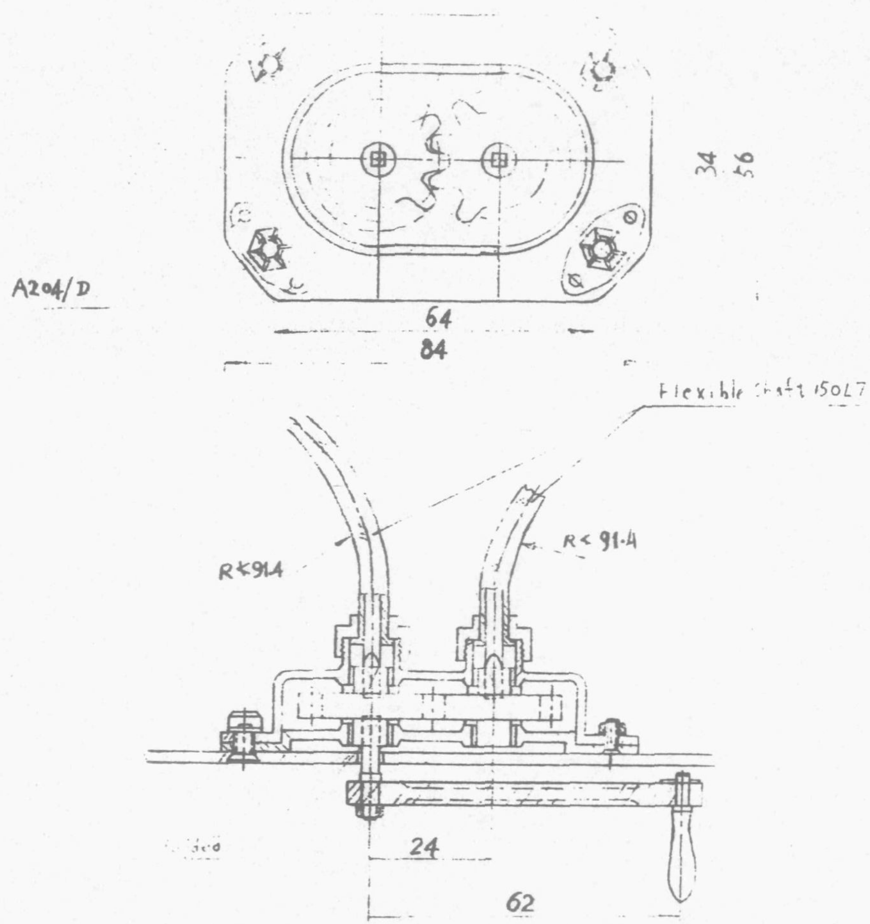
DEPARTMENT OF AIRCRAFT DESIGN
COLLEGE OF AERONAUTICS



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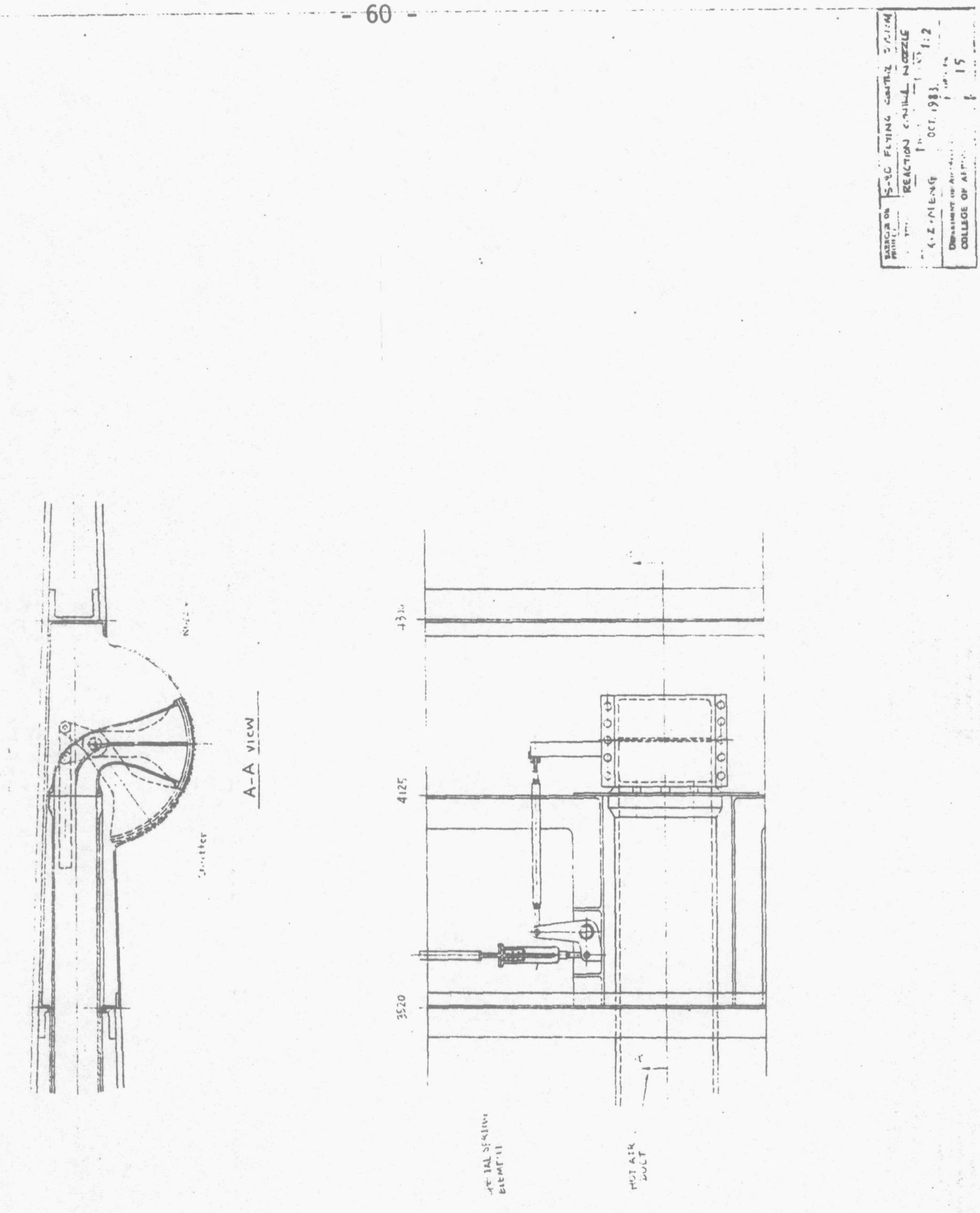
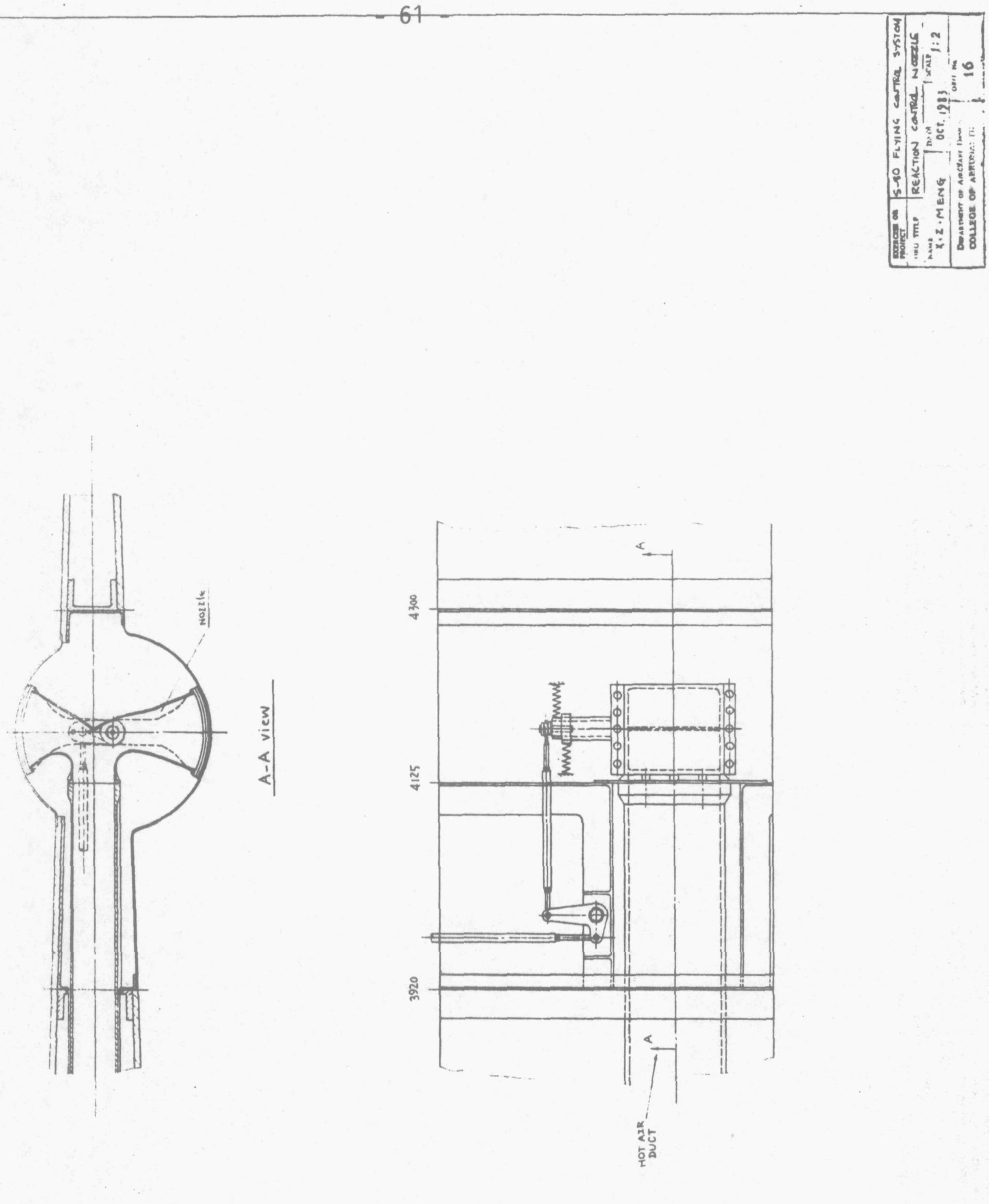
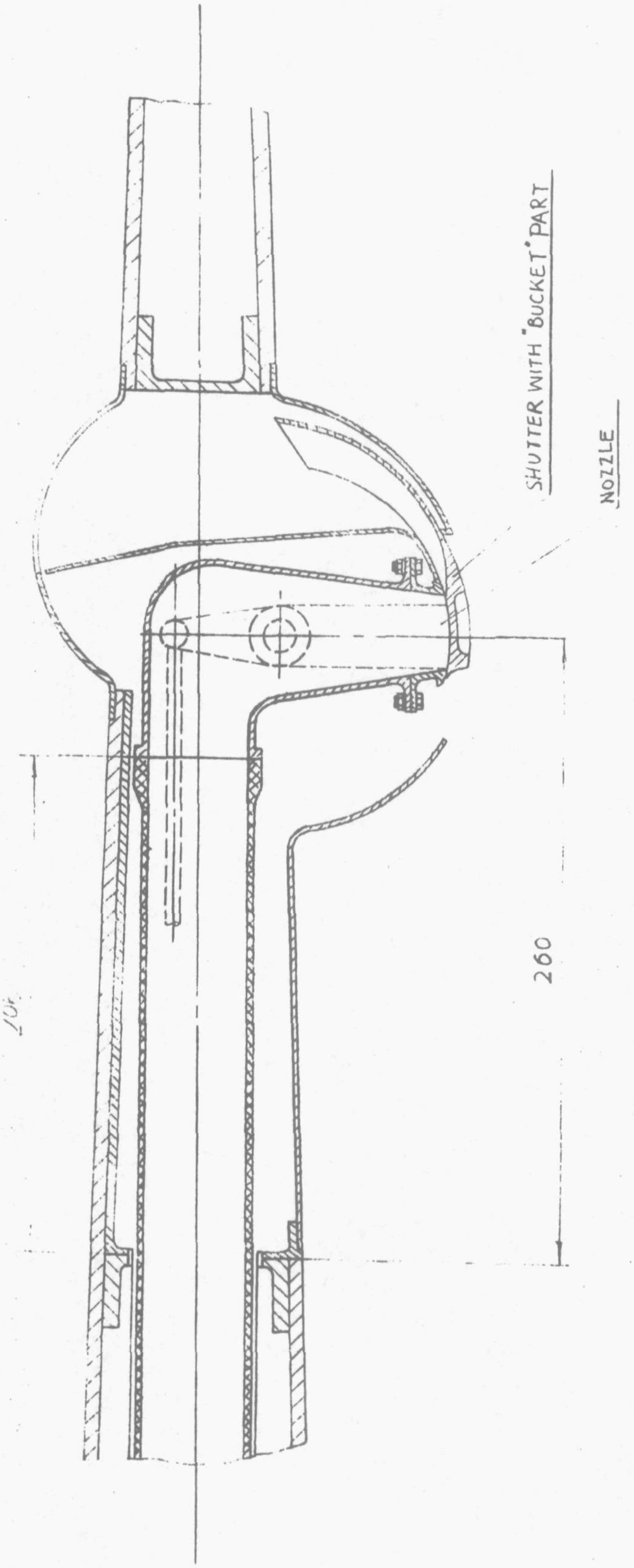
NAME X·Z·MENG	DATE OCT. 1983. SCALE 1:1
	DRG No 14


REACTION CONTROL NOZZLE
1:2
OCT. 1983
15
A-A VIEW
4125
3520


S-80 FLYING CONTROL SYSTEM
REACTION CONTROL NOZZLE
X.Z. MENG
OCT. 1983
1:2
16
COLLEGE OF AERONAUTICS
NOZZLE
A-A view
4300
4125
3920
A
A
HOT AIR DUCT



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NAME X. Z. MENG	DATE OCT. 1983. / SCALE 1 : 2
DEPARTMENT OF AIRCRAFT DESIGN COLLEGE OF AERONAUTICS	DRG. No. 17